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PROCEEDINGS OF THE 1989 STRUCTURAL INTEGRITY PROGRAM CONFERENCE

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April 1990

5-7 December 1989

Hilton Palacio Del Rio Hotel
San Antonio TX

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FOREWORD

This report was compiled by the Materials Integrity Branch, Systems Support Division, Materials Laboratory, Wright Research & Development Center, Wright-Patterson AFB, Ohio. It was initiated under Task 24180704 "Corrosion Control & Failure Analysis" with Thomas D. Cooper as the Project Engineer.

This technical report was submitted by the editors.

The purpose of this 1989 Conference was to bring together technical personnel in DOD and the aerospace industry who are involved in the various turbine engines, airframes and other mechanical systems. It provided a forum to exchange ideas and share new information relating to the critical aspects of durability and damage tolerance technology for aircraft systems. The Conference was sponsored by the Aeronautical Systems Division Deputy for Engineering and Materials Laboratory of the Wright Research & Development Center, Wright-Patterson AFB, Ohio. It was hosted by the Air Force Logistics Command's San Antonio Air Logistics Center.

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5-7 DECEMBER 1989

Hilton Palacio Del Rio Hotel
San Antonio, Texas

SPONSORED BY:

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WRDC/Materials Laboratory

HOSTED BY:

San Antonio Air Logistics Center
Directorate Material Management
Fighter/Tactical/Trainer
Systems Program Management
Division (SA-ALC/MMS)

AGENDA

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0830-0900	Research on Multiple Site Damage P. Tong, O. Orringer, S. G. Sampath, and S. N. Bobo - U.S. Department of Transportation
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1100-1130	Force Management Technology Application to Aging Aircraft J. Cochran and D. Morcock - Lockheed Aeronautical Systems Company Dr T. F. Christian and D. O. Hammond - WR-ALC
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1400-1430	Probabilistic Structural Analysis Methods (PSAM) for Aerospace System Components H. Burnside and T. Cruse - Southwest Research Institute
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1530-1600	Analysis of Cracking in the 66 Percent Wing Spar of the T-38 Aircraft K. Shrader, H. Burnside and W. Sparks - Southwest Research Institute
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1700-1730	Probabilistic Failure Assessment M. Creager - Structural Integrity Engineering N. Moore and D. Ebbeler - JPL
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0830-0900	Correlation of the F-16 Block 30 Finite Element Models (FEM) K. Wilson - ASD/YPEF
0930-1000	F-16 C Full-Scale Airframe Durability Test C. A. Babish, IV - ASD/YPEF
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1000-1030	Dynamic Fatigue Testing of the F-15 Vertical Tail to Simulate the Buffet Environment R. Milliere - McDonnell Aircraft Company J. Resnicky - ASD/VFEF
1030-1100	Rapid Repair DTA Technology for F-16 Aircraft J. P. Gallagher and P. C. Miedlar - University of Dayton Research Institute V. Juarez - General Dynamics
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1400-1430	NDE for Engine Structural Integrity Programs W. Herron - General Electric Company
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1530-1600	New Magneto-Optic/Eddy Current Inspection Methods for Aging Aircraft G. L. Fitzpatrick and D. K. Thome - Physical Research, Inc.
1600-1630	High Cycle Fatigue Endurance Limit of Titanium Materials Capt V. Johnson - ASD/YZEE
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THURSDAY, 7 DECEMBER 1989

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1030-1100	ENSIP Aeromechanical Testing - Where Do We Go From Here O. Davenport and W. D. Cowie - ASD/YZE
1100-1130	Retirement for Cause Performance - Predicted vs Actual M. VanWanderham and J. A. Harris - Pratt and Whitney
1130-1330	LUNCH AND PRESENTATION The USAF-ALC Damage Tolerance Assessment Program H. N. Abramson, Executive Vice President - Southwest Research Institute

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1400-1430	An Update of SOFI: Navy's Structural Online Fatigue Information System S. Kadiyala, S. Moon, M. Mitter and S. Neriya - Aerostructures, Inc. B. Filar - Systems and Electronics, Inc. C. Garber - NAVAIR

1430-1500	Risk Analysis and Probabilistic Design S. Jackson - LTV
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1530-1600	C-130 Loads Environment Spectra Survey (LESS) - Transition to the Standard Flight Data Recorder (SFDR) System T. C. Bell - Lockheed Aeronautical Systems Company W. O. Greenhaw - WR-ALC/MMCRA
1600-1630	A-10A Service Life Monitoring Program H. Axelrod - Grumman Aerospace Corporation K. McPhaul - SM-ALC/MMSRD
1630-1700	Improvement in Structural Integrity Data Validity Utilizing Airborne Parameter Reasonableness Techniques T. W. Mayhan - Smith Industries
1700	ADJOURN

INTRODUCTION

This report contains the proceedings of the 1989 USAF Structural Integrity Program Conference held at the Hilton Palacio Del Rio Hotel in San Antonio, Texas from 5-7 December 1989. The conference, which was sponsored by the ASD Deputy for Engineering and the WRDC Materials Laboratory, was hosted by the San Antonio Air Logistics Center Material Management Directorate Fighter/Tactical/Management Division (SA-ALC/MMS)

This conference, which in recent years has been held annually, has evolved from government only participation to a conference where government and industry personnel meet to exchange views on how to improve the structural integrity of military weapon systems. It has evolved from a conference where the emphasis was on issues derived from the Aircraft Structural Integrity Program (ASIP) to a conference where the Engine Structural Integrity Program (ENSIP) and the Mechanical Subsystems Integrity Program (MECSIP) are a vital and integral part of the program. The agenda for this conference reflects this change in emphasis. The comments received from the attendees indicate they believe this change has been beneficial to the overall quality of the conference.

The sponsors are indebted to their hosts for their support of the conference. The sponsors are also indebted to the speakers for their contributions. In particular, thanks are due to luncheon speakers for their contributions to the program. Mr Jim Smith from LTV provided an extremely interesting view of the structural integrity of the A-7D aircraft, Mr Dirk Spiekhout from the National Aerospace Laboratory in the Netherlands gave an informative paper on their use of the F-16 and Dr Norman Abramson from the Southwest Research Institute briefed the conference on the results of the USAF Scientific Advisory Board review of the organic capability for damage tolerance assessments in the Air Force. Much of the success of the conference is due to the efforts of Jill Jennewine and her staff from the Universal Technology Corporation. Their cooperation is appreciated. We are also grateful to Marianne Ramsey of WRDC/MLSA for compiling the Proceedings and preparing them for publication.

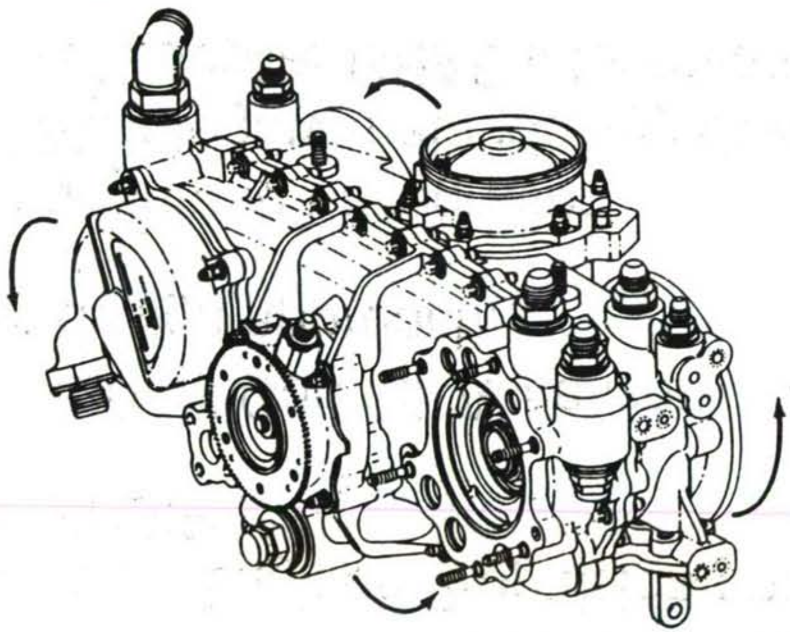
John W. Lincoln

John W. Lincoln
ASD/ENFS

Thomas D. Cooper
WRDC/MLSA

Engine Integrity Programs for Controls, Accessories, and Externals

An Integrated Design Approach



P.A. Domas, GE Aircraft Engines

1989 United States Air Force
Structural Integrity Program Conference
San Antonio, 5-7 December, 1989

S-48215-111489

Overview

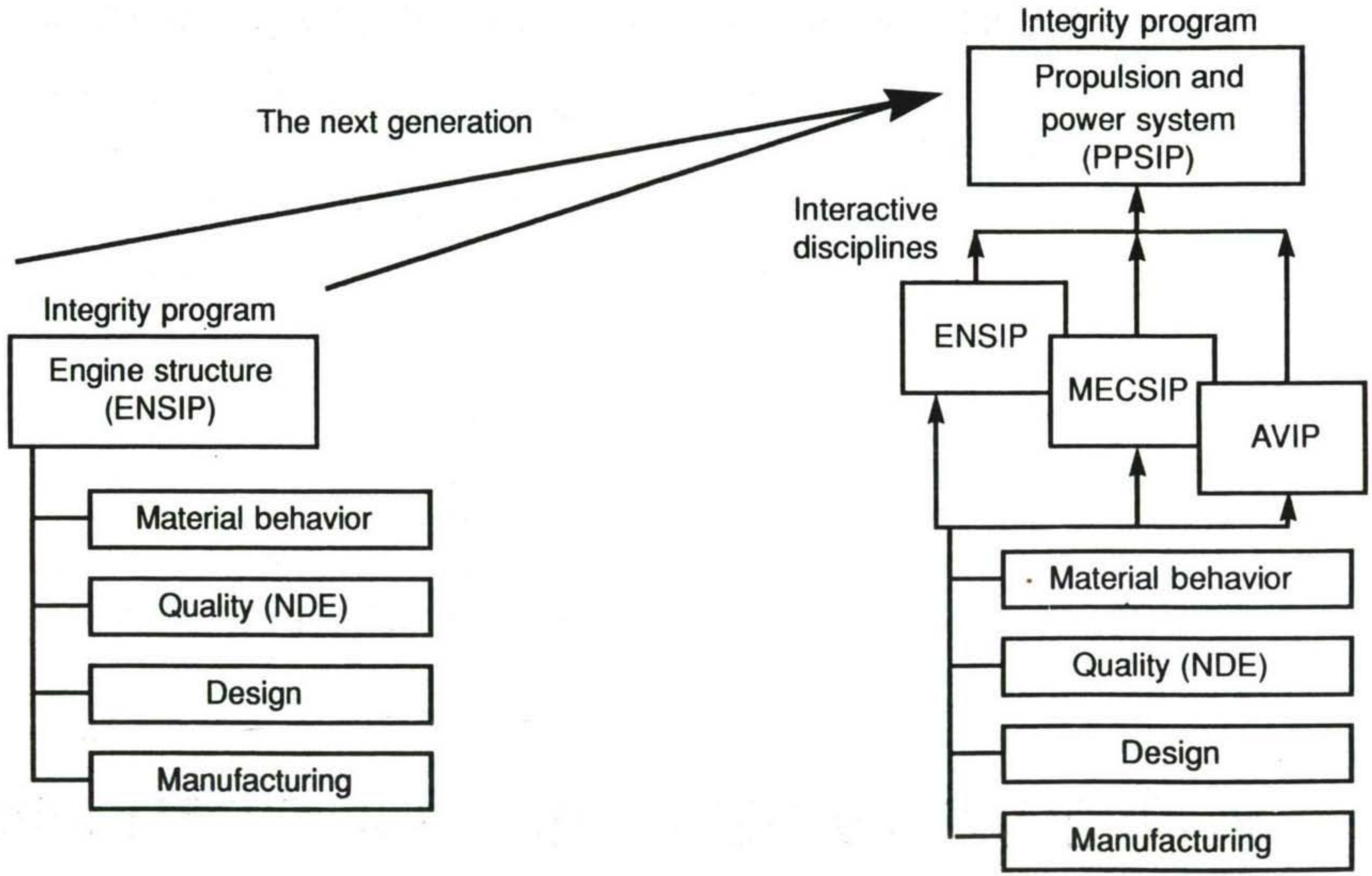
- Brief recap of GE Aircraft Engines Propulsion and Power System Integrity Programs (PPSIP).
- The challenge for a unified Control, Accessory, and External (CA&E) component design approach.
- Meeting the challenge through integration:
  - Air Force One CA&E integrity methodology development.
  - Ongoing evolution of supplier/GEAE interaction.
- PPSIP, TQM, CIP, DEMING – bringing quality, reliability, and maintainability to the forefront.

S-48216-111489

Brief Recap of PPSIP at GE Aircraft Engines

- Advanced Tactical Fighter Engine (ATFE) PPSIP includes:
  - Turbomachinery (ENgine Structural Integrity Program)
  - CA&E Components (MEChanical Equipment and Subsystem Integrity Program)
    (AVionics Integrity Program)
- Maintains and expands the requirement for integration among supporting organizations both internal and external (suppliers).
- In last year's conference described the GE Aircraft Engines ATFE PPSIP plan:
  - Stressed integration among customer, contractor, and suppliers.
  - Outlined a "Vendor Involvement Plan (VIP)".
- GEAE aggressively implementing the ATFE vendor involvement plan.

Expanding the Integrity Influence



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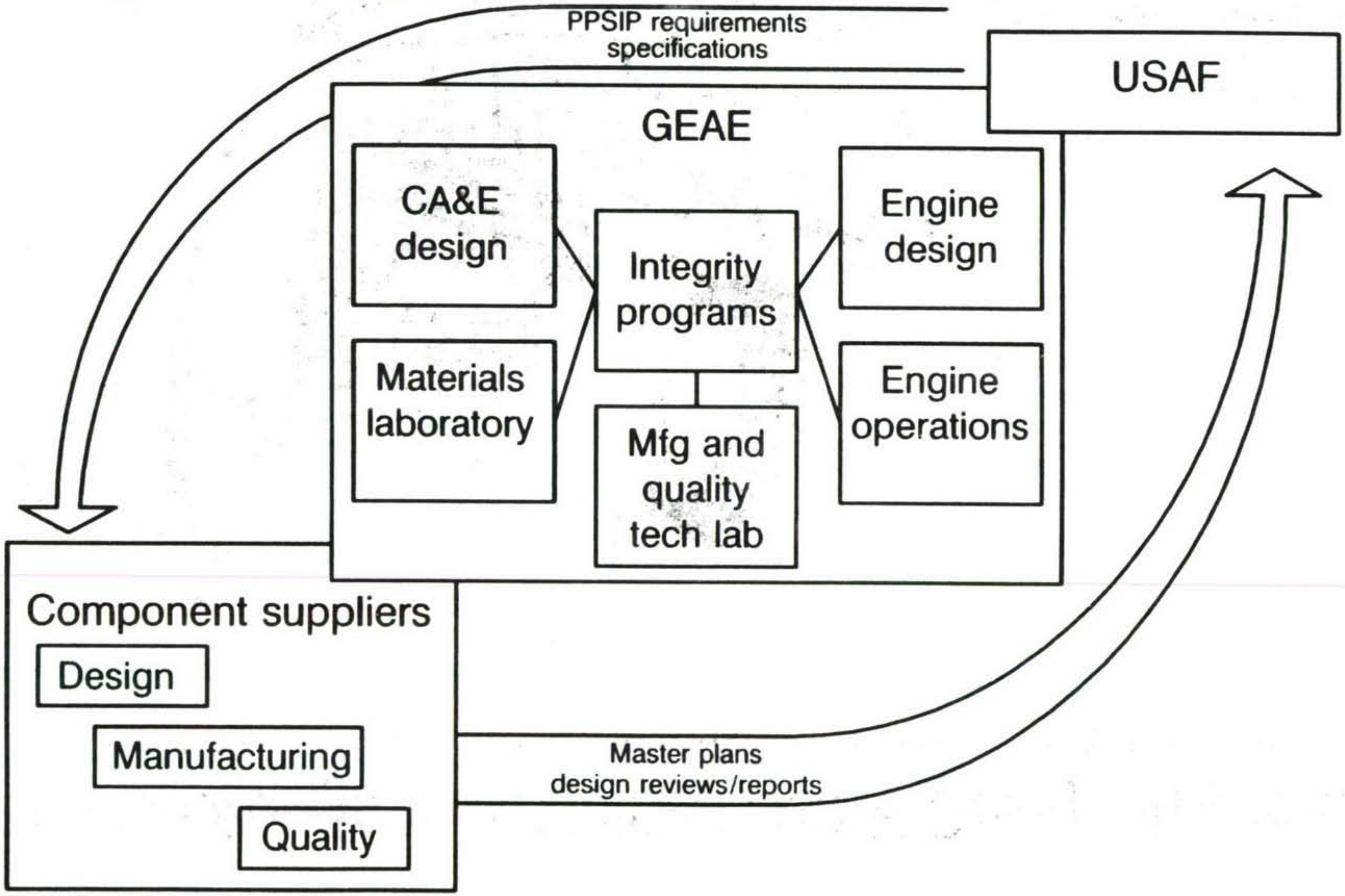
Integrated Organization Structure



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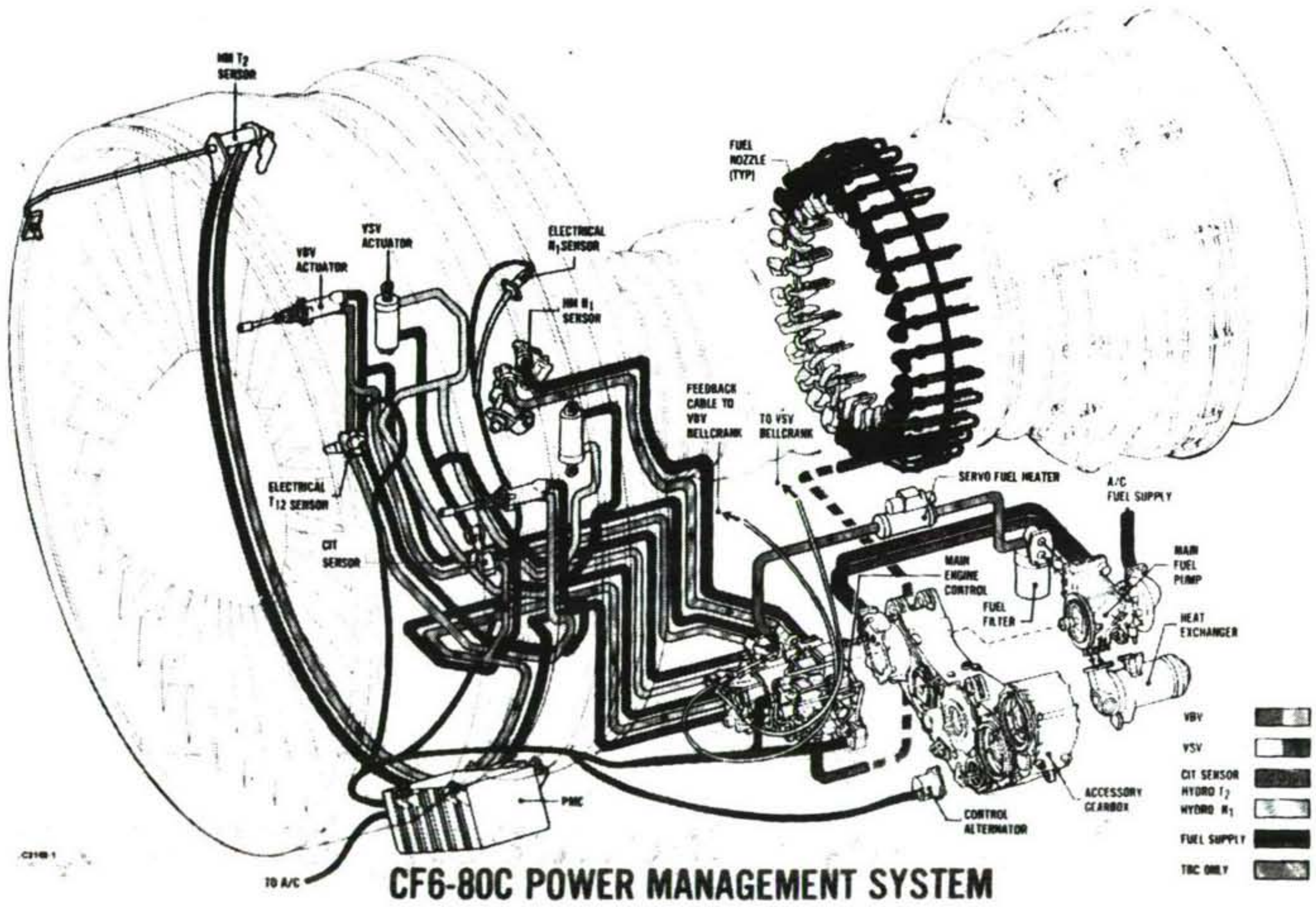
The Challenge for a Unified CA&E Design Approach

Multiple Suppliers, Multiple Components, Multiple Functional Requirements



Typical
turbine engine
power management
system

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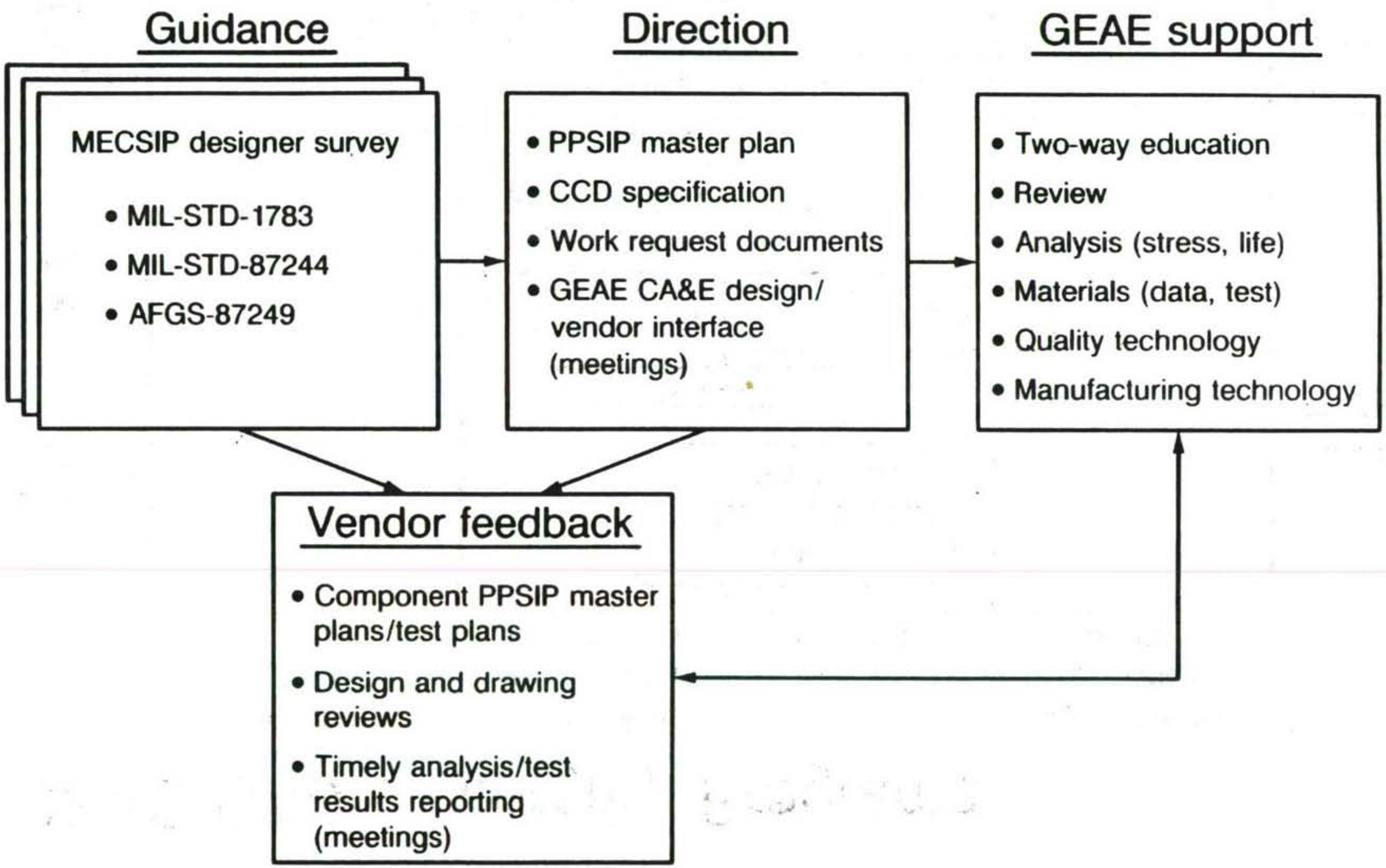
VIP (Vendor Involvement Plan)



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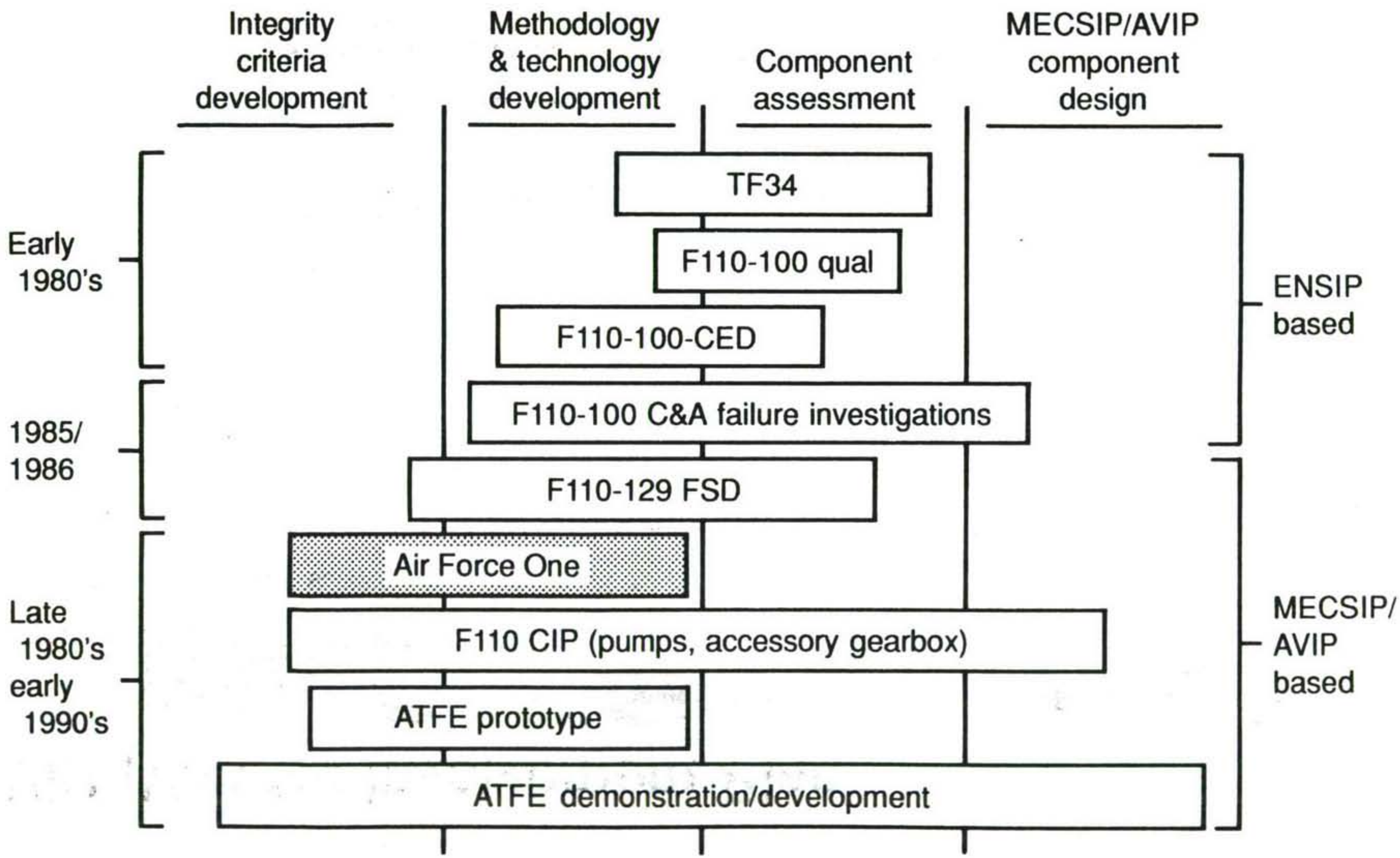
GEAE CA&E Intergrity Programs



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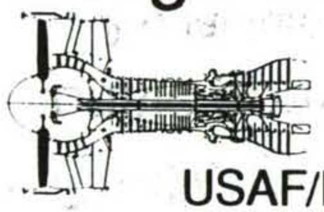
Bridge the "Gap" Between Concept and Practice



USAF/Engine contractor Integrity concepts/Criteria

- Diverse engineering resources
- Material behavior database
- Analytic tools/ techniques
- ENSIP/PPSIP experience
- Computing resources
- Diverse test facilities
- Material research laboratory
- Quality technology laboratory

CA&E Environment (real world)

- Focussed engineering resources
- Component specific specialty skills
  - Manufacturing
  - Design
  - Materials
- Focussed material database
- Unique test facilities
- Unique experience base
- Unique Quality control methods

* Common purpose - continuous product improvement
* Combination of broad vision and base support coupled with uniquely valuable skills, data, experience is the target status

S-48221-111489

GEAE CA&E PPSIP Philosophy Established

- Lead the effort rather than being driven
- Listen to internal CA&E designers and external suppliers – use their input to implement PPSIP intelligently, efficiently, and cost effectively
- Define PPSIP adequately and educate everyone involved
- Challenge the high quality people involved to achieve the basic goal
  - A quantum leap in reliability of CA&E parts

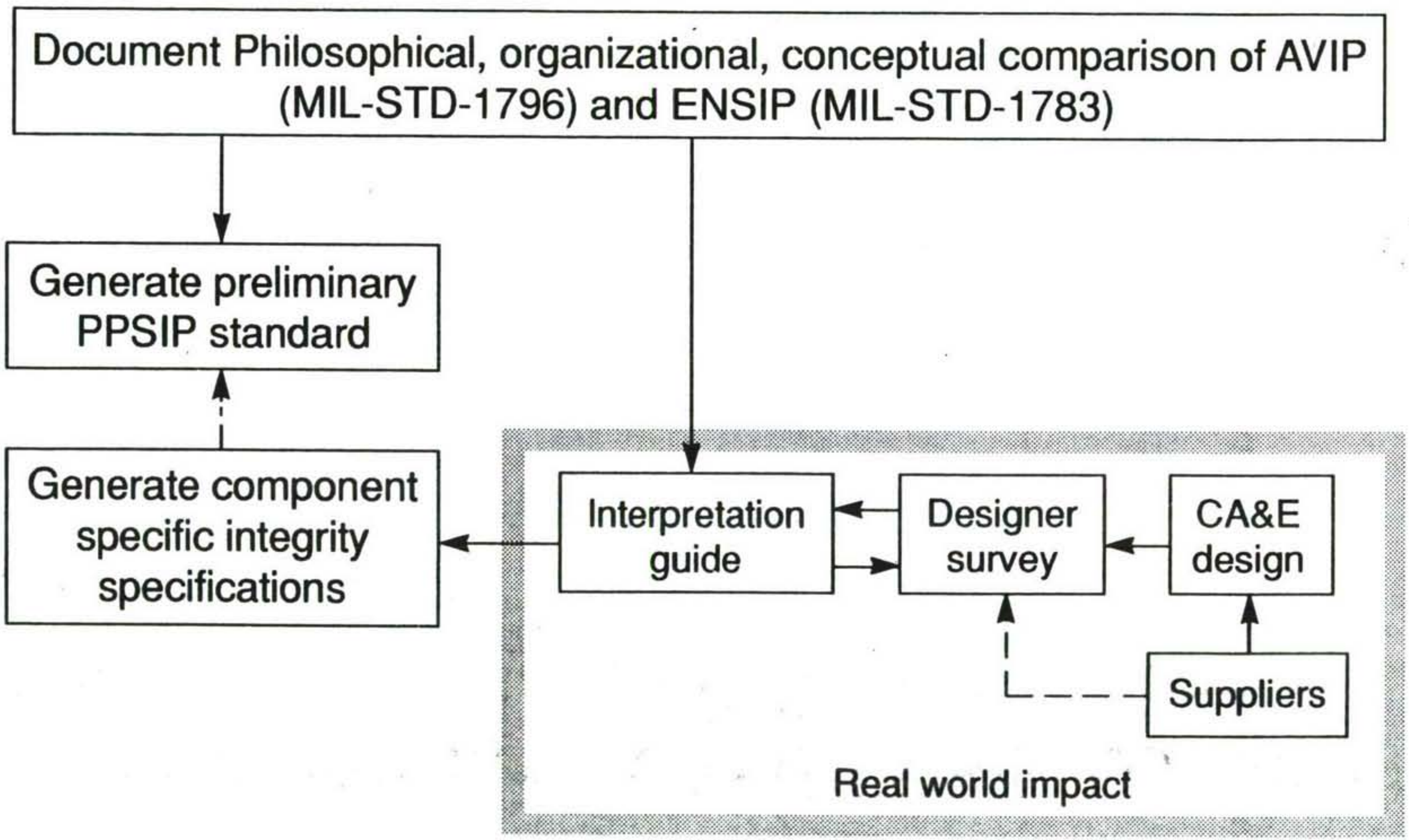
S-48222-111589

Generated Prototype CA&E PPSIP Documents for Improved Control and Communication

- To understand the plan and requirements:
  - Documented comparison of generic AVIP approach and turbomachinery specific ENSIP approach
- To help facilitate implementation:
  - Generated preliminary PPSIP standard and an example component specific specification
  - Initiated collection of C&A analysis guidelines
- To incorporate "real world impact" for cost effectiveness and viability:
  - Designer survey
  - Interpretation guide for translating survey results into appropriate specifications

S-48223-111589

Air Force One PPSIP Criteria/Methodology Development



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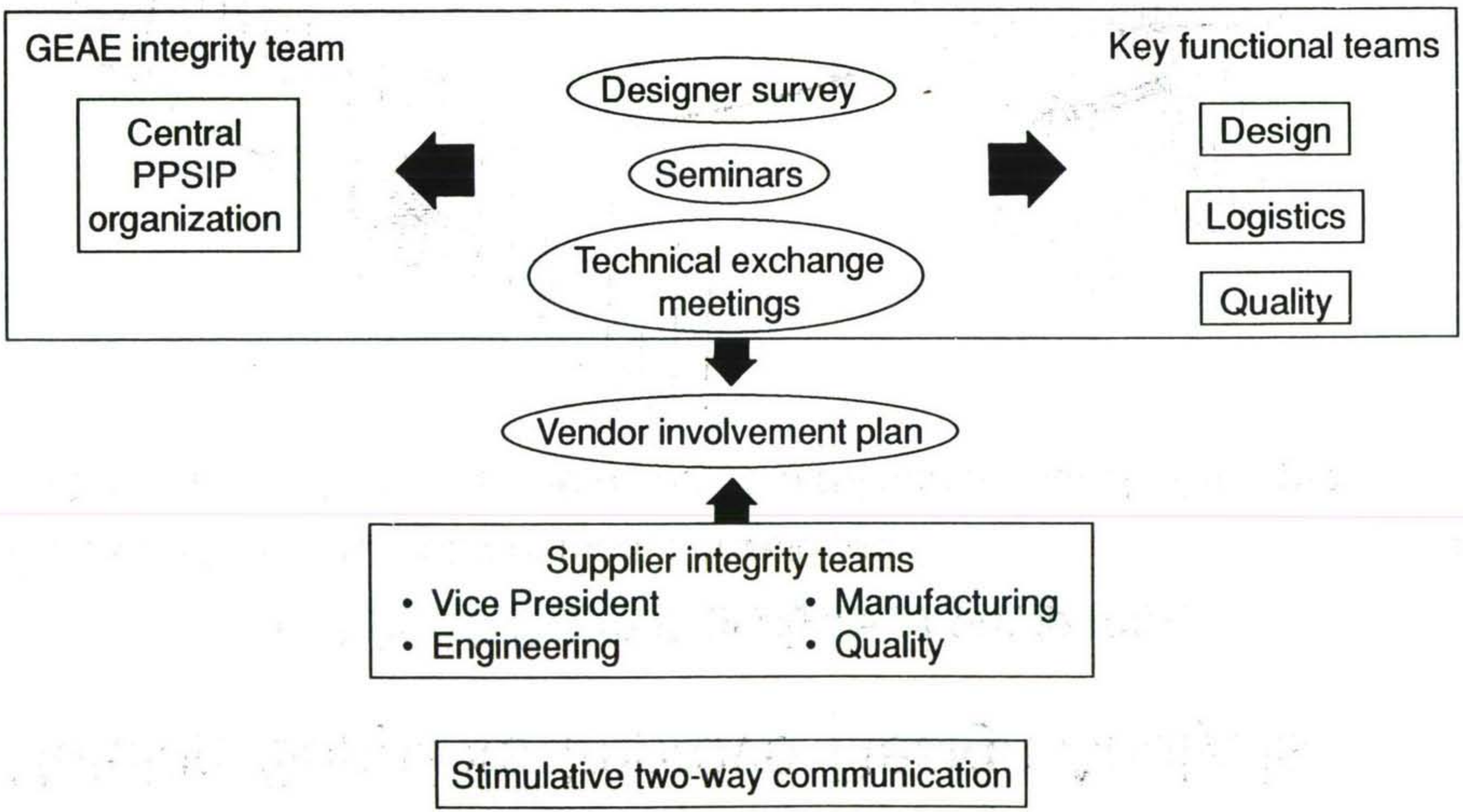
Breaking Down Barriers (Continuous Improvement Initiative)



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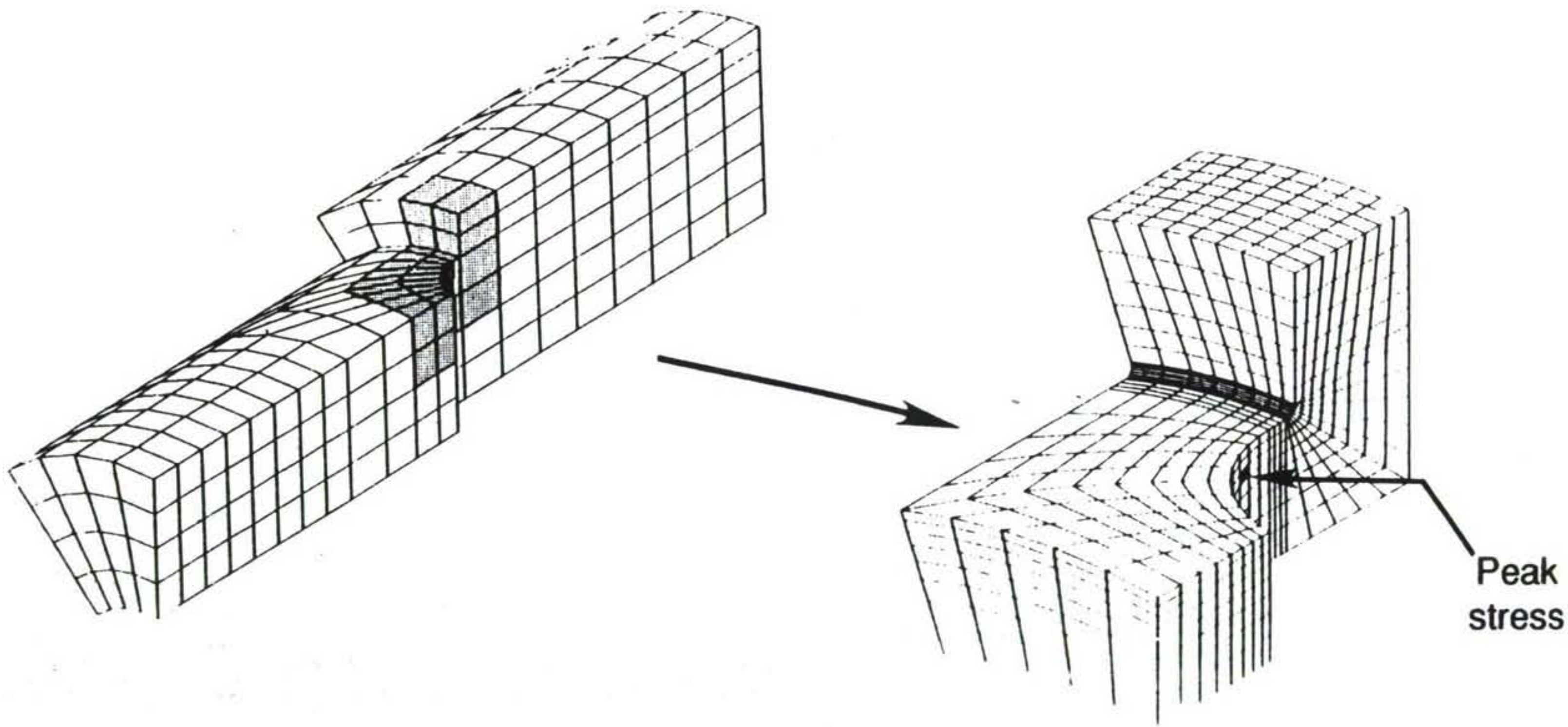
Helping Suppliers Improve Design Analysis

Detailed 3-D stress analysis of pump rotor

- Coarse 3-D finite element model developed
- Maximum peak stresses obtained from refined model of fillet region



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CA&E Designers Utilizing Computer Tomography

- GE medical systems computer tomography (CT) machine
- X-rays produce two dimensional cross sections
  - Check for “thin-wall” potential
  - Check for potential porosity and shrink regions
- Design iterations before full casting run commitment

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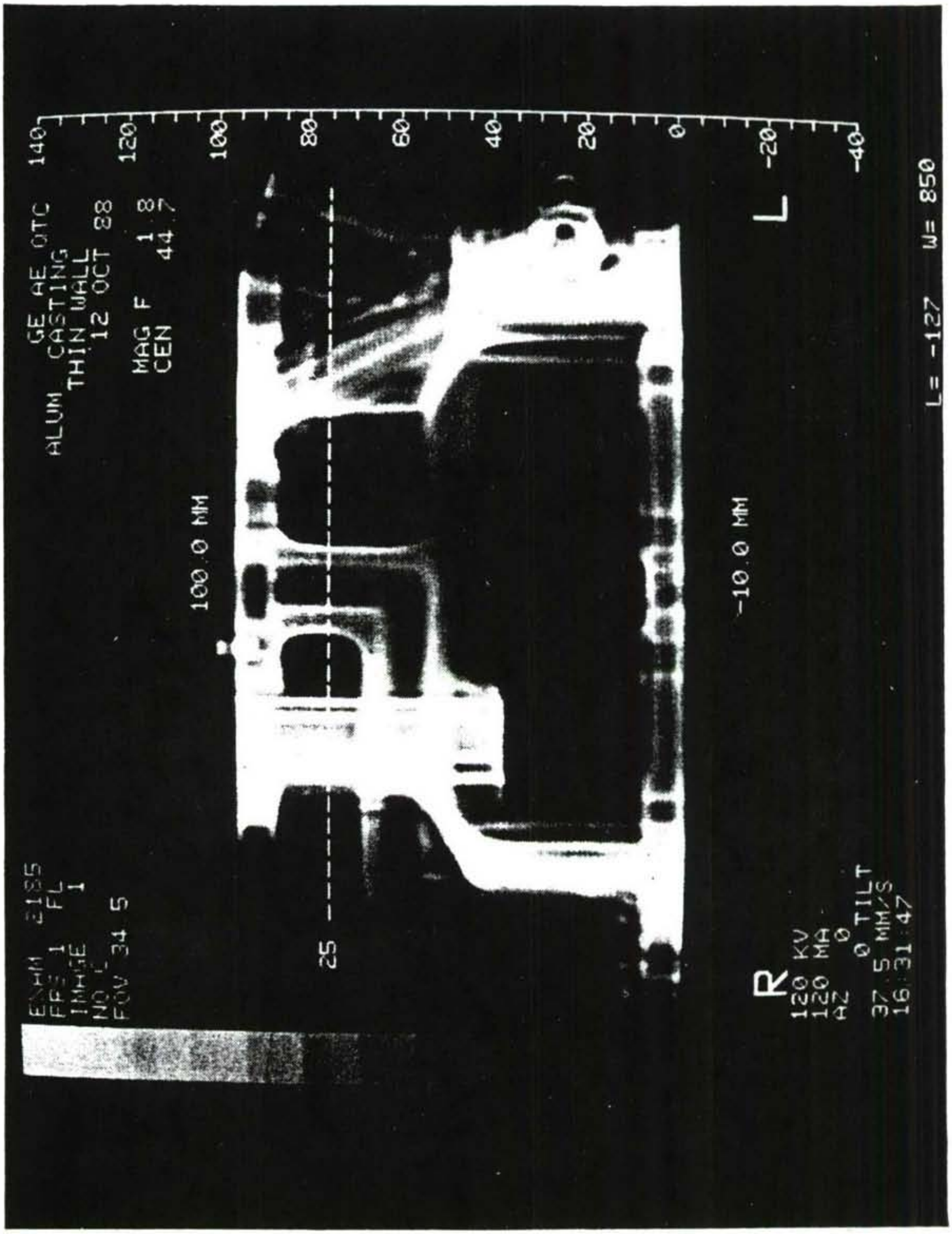

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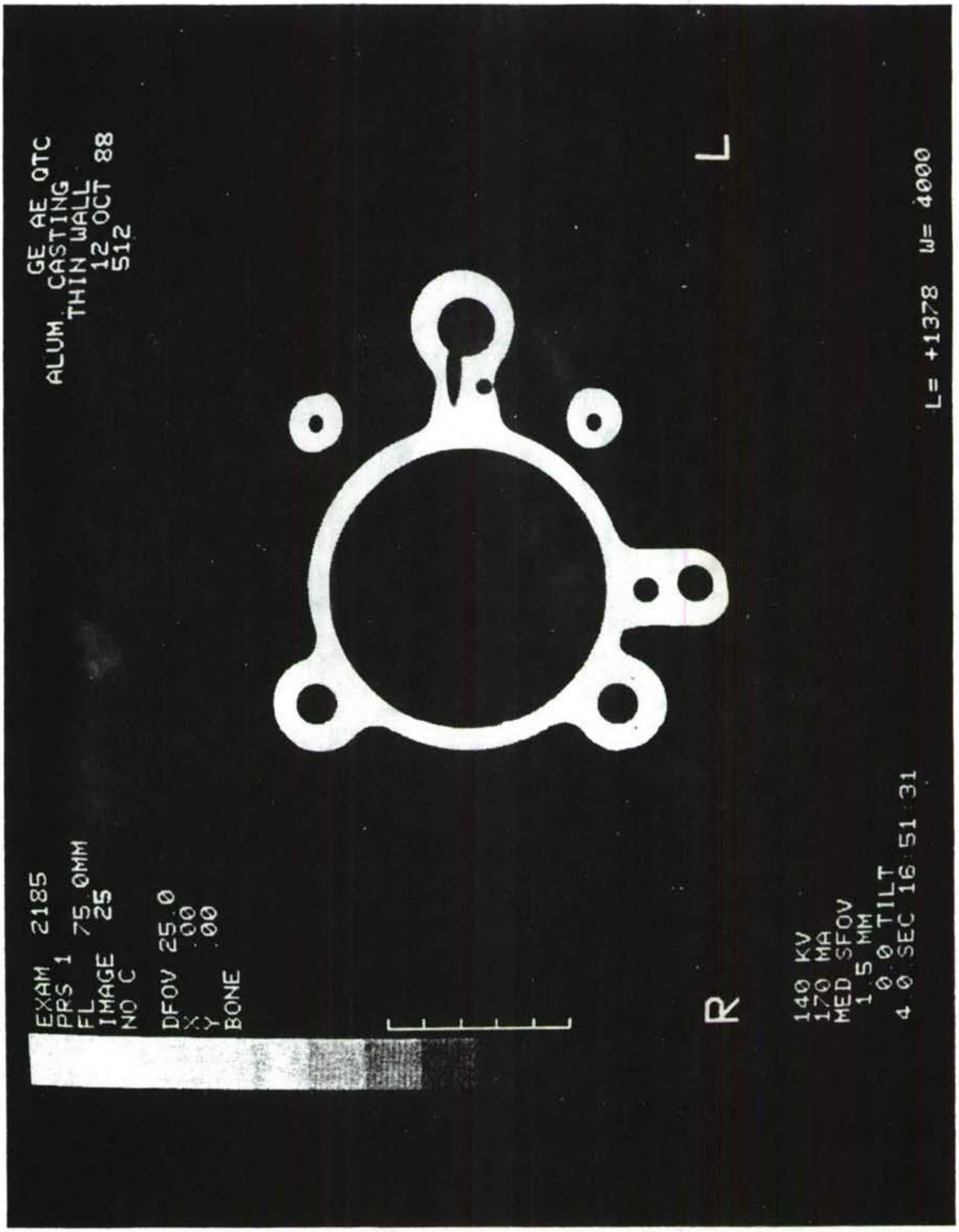

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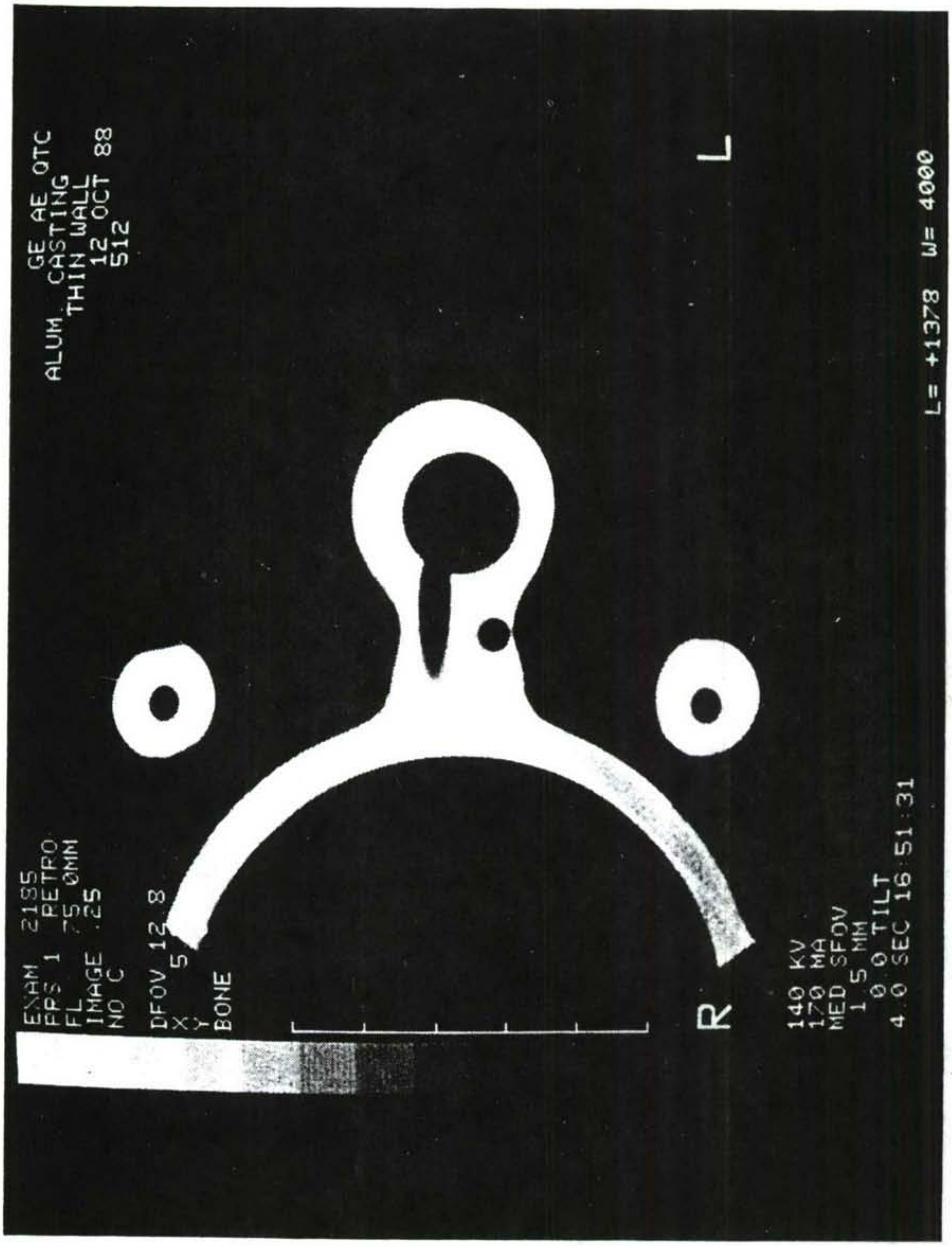

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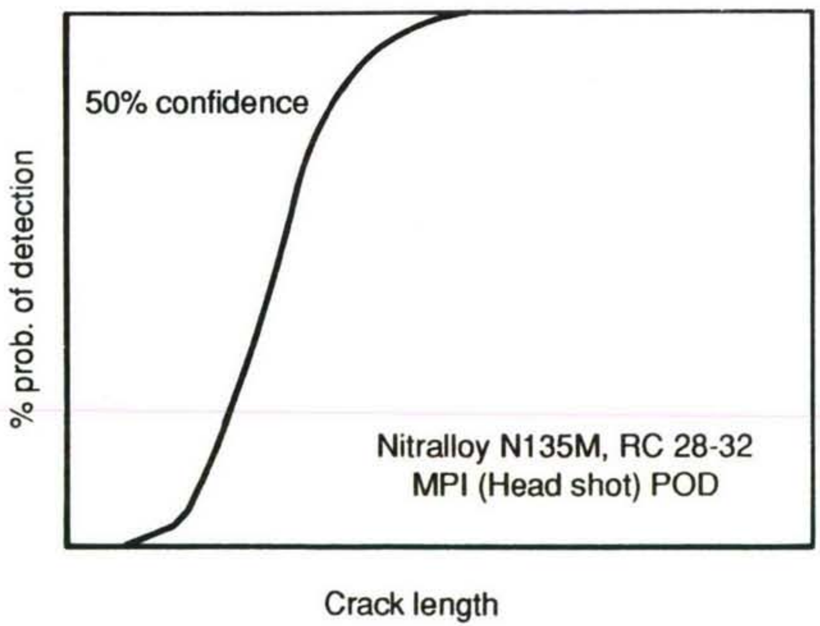
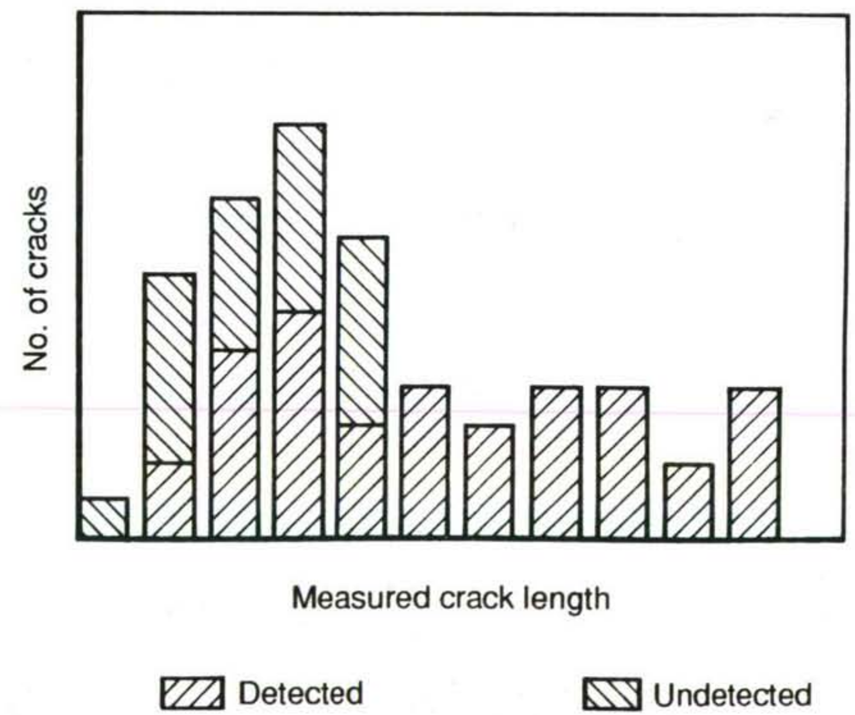
Taking Nondestructive Evaluation Quantification Into the CA&E World

Nitralloy/magnetic particle inspection detection capability

Probability of detection



Detection success/failure data



S-48228-111589

Taking Action to Accomplish Transformation (The DEMING Management Method)

- Supplier excellence program (GEAE initiative)
- Total Quality Management (TQM) (USAF philosophy)
- Continuous Improvement Programs (CIP)
- PPSIP Vendor Involvement Plan (VIP)

Vigorous programs – consistent message
Quality, reliability, and maintainability to the forefront

S-48229-111589

Summary

- PPSIP has forced re-evaluation of CA&E design process
- New approach, with more extensive contractor/supplier interaction and coordination of resources is evolving
- The integrated design approach initiated in the Air Force One program and being implemented through vendor involvement efforts is consistent with current USAF and industry thrusts for improved quality, reliability, and maintainability
  - “It makes sense”
- Extensive positive designer and supplier feedback
  - “It’s working”!

S-48230-111589

RESEARCH ON MULTIPLE SITE DAMAGE *

P. Tong, O. Orringer, S.G. Sampath, and S.N. Bobo
U.S. Department of Transportation
Transportation Systems Center
Cambridge, MA 02142

BACKGROUND

Eighteen months ago at the June 1988 Aging Airplane Conference [1], the Federal Aviation Administration (FAA) and the aeronautical industry made commitments to carry out the research and development necessary to maintain the structural integrity of older transport category airframes which continue in service. The private and public sectors both went quickly into action, a response at least in part to the March 1988 in-flight structural failure of the fuselage of N73711, a Boeing 737 owned and operated by Aloha Airlines. This paper addresses three major elements of the FAA program: research to understand the behavior of multiple site damage (MSD); how to find MSD during airframe inspection; and how to avoid MSD in future designs. Under each of these headings, a summary will be given of what has been done and what is planned.

For the purpose of this paper, MSD will be defined somewhat loosely as a group of small cracks that appear in an airframe at about the same time and which originate from similar design details located in a common area. After some period of slow fatigue crack growth, the MSD may consist of small cracks of similar size or a large lead crack growing toward a group of small cracks. Neither situation is specifically addressed in the current damage tolerance requirements for either military [2] or transport category [3] airplanes. Those requirements consider only the isolated cracks which constituted the airframe service fatigue experience base up to the mid-1970s.

Post-accident photographs of the N73711 fuselage show the presence of MSD in locations adjacent or similar to the location where the in-flight failure originated (Figure 1), and the findings of the National Transportation Safety Board [4] confirm that MSD contributed to the accident. Special inspections following the N73711 accident have revealed MSD in other airframes, including models other than the 737 and makes other than Boeing aircraft. Also, retrospective consideration of the 1983 failure of a special repair in a Japan Airlines 747 aft pressure bulkhead and earlier assessments of widespread cracking limits in Air Force transport wings suggest that MSD can occur in significant structural items

* This work is sponsored by the Technical Center, Federal Aviation Administration.

other than the fuselage pressure shell. Thus, we must assume that MSD may have the potential to appear anywhere in the nation's older fleets.

UNDERSTANDING MSD BEHAVIOR

Like tolerance of isolated cracking, MSD tolerance can be described in terms of detectable crack size[1], critical crack size[2], and the number of flights or flight hours of slow crack growth between these limits. However, all three factors can have values for MSD quite different from the range of values generally associated with isolated cracking. There is also a need to identify those aircraft and structure details which have a significant potential for MSD. Several parallel research efforts on these topics are in progress.

Fleet Database

The best way to get early warning about MSD in existing aircraft fleets is by conducting detailed inspections of high-time airframes. To be practical, however, such inspections should fit into the airlines' maintenance programs. A fleet database will help the FAA to prioritize the special inspection efforts and integrate them with the maintenance schedules. It is also hoped that analyses of the database might shed some light on the potential for MSD in different types of design details.

To date the research has concentrated on the approximately 5,000 transport category airplanes operated by the nation's scheduled air carriers. A current inventory has been produced by aircraft type, serial number, N-number, utilization history, and owner/operator history. These are the basic elements of data required to group the aircraft in age categories. For example, the database shows that almost half the DC-9 fleet has exceeded its flight cycle design goal of 45,000 landings (Figure 2), and that seven out of nine transport types have high-time aircraft beyond their respective goals (Figure 3).

Additional data elements will be needed to further group the aircraft in sub-categories relating to MSD potential. These elements include model and sub-model (down to the level of significant differences in the primary structure), implementation of major structural Airworthiness Directives (ADs) and Service Bulletins (SBs) by individual airplane, and major repair history (including

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1 For present purposes, detectable size is loosely defined as the largest crack likely to escape detection. Detectable size depends on the inspection method, procedure, inspector skill level, and other human factors such as boredom when doing repetitive tasks.

2 Critical size is defined as the smallest crack size which would precipitate unstable fracture when subjected to expected flight-load stresses.

repair of corroded structure) by individual airplane. Other existing databases such as the Service Difficulty Report (SDR) system and the Aviation Safety Analysis System (ASAS) are being searched for these kinds of data. Information on MSD occurrence patterns will be sought from the extended database. For example, are certain types of repairs followed by appearance of MSD, either in the repaired area or in adjacent structure? The FAA also participates on the Air Transport Association (ATA) communication working group subcommittee on maintenance data format and analysis.

Over the next year, the database development will be extended to the commuter aircraft fleet. Preparations are now underway for a preliminary survey of the commuter fleet by a team of experts. Over the next six to nine months, the expert team will visit manufacturers to review approximately 35 aircraft types for fleet age status, SDR history, and MSD potential.

Tear Strap Effectiveness

The ability of tear straps to contain fuselage fractures precipitated by MSD linkup is of great concern. The transport fuselages in service today are damage-tolerant, in the meaning of the term as established by two decades of design practice [5]. Specifically, these structures are able to contain a one-bay fracture (two bays in some airframes) at 110 percent of design maximum cabin pressure differential plus aerodynamic pressure differential effects. This damage tolerance requirement [3] originally came from the need to protect against the consequences of a failed propeller blade striking the cabin [5]. Thus, it is not surprising that the scenario assumed for design focused on isolated damage, and that it has been carried over to the jet age by application to isolated fatigue crack damage.

The objective of the present research program is to determine how effectively tear straps can contain a fracture in the presence of MSD. The general scenario assumes a fracture resulting from linkup of a group of MSD cracks. The fracture in this case lies along a skin splice, rather than in the mid-bay position usually assumed in the present design and test practices. Also, the MSD fracture may be advancing toward adjacent bays which contain additional (but as yet unlinked) MSD cracks.[3]

The subject of tear strap effectiveness will be investigated by means of full-scale component tests. A special fixture has been constructed to accommodate curved panels 66 inches on the circumference by 120 inches axially, with allowance for a range of 70 to 75 inches in the panel radius of curvature. The fixture is a shallow pressure box which accommodates the test article by means of floating seals (Figure 4). Hoop stress is applied by means of

3 In broad outline, the scenario resembles the adjacent panel cracking considerations raised during Air Force structural integrity assessments of the C-5A and C/KC-135 wings [6].

the internal pressure and is reacted through lateral turnbuckles (Figure 5). The turnbuckles have a limited angular orientation range to enable the simulation of hoop bending effects, which can occur near major load transfer areas such as the wing-body junction.

The normal mode of pressurization is pneumatic to provide some degree of realism in the total system flexibility for the purpose of conducting fracture resistance type tests. Provision has also been made for hydraulic pressurization to allow rapid cycling for fatigue and slow crack growth tests.

Axial pressure stress is simulated by means of servohydraulic actuators connected to the panel ends through additional sets of turnbuckles. These actuators are commanded by a closed-loop control system which keeps the axial stress in phase with the hoop stress. The actuators can also be biased to simulate the small added tension induced in the fuselage crown by vertical-plane body bending.

Vertical-plane bending also induces shear in the fuselage skin, with the shear stress attaining its maximum in the mid-body area. Body-bending shear is believed to be the cause of the slightly inclined orientation of the N73711 MSD cracks shown in Figure 1. The panel test fixture may eventually be modified to provide for the simulation of body-bending shear together with the pressure stresses.

Figures 6 and 7 illustrate the configuration selected for the first series of test panels. The dimensions, construction details, and materials were chosen to produce a configuration similar to that found in the Boeing 737 fuselage. The test panels do not precisely match the aircraft and, therefore, the test results are not intended to apply to the 737. However, the panels will provide stress levels and structure flexibilities which do lie in the range of existing designs, and the test results will thus be sufficiently realistic for the purposes of drawing general conclusions about MSD behavior and calibrating damage tolerance estimation procedures.

The first panel, now under construction, is being fabricated with a continuous skin and will be extensively instrumented for the purpose of checking out the test fixture. A major element of the checkout phase is verification that the fixture influences on panel stress and flexibility are confined to the edge zone, leaving a central region of two to four bays in each direction with insignificant fixture effects.

The remaining panels in the first series will be fabricated with two skins joined at an axial lap splice along the panel crown. The initial fracture resistance tests will investigate single long cracks for the purpose of comparing results with similar experiments that aircraft manufacturers have conducted in the past. Following those tests, the program will focus on panel tolerance of two situations: small MSD cracks approaching linkup size; and a long lead crack with adjacent MSD. Results from the first test series

are expected to be available by June 1990. Later test series are planned to extend the investigation to butt-joint type splices and repair designs.

Improved Analysis Methods

Fracture stability analyses of stiffened flat panels have been used to correlate test data and predict damage tolerance for over two decades [5]. The models most widely used for these analyses are based on the displacement compatibility method, in which the effects of load transfer through fasteners are represented by displacement influence functions in the skins and stiffeners. Recent work has shown that advanced finite element methods can provide equal or better results more efficiently by incorporating the skin stress influence functions directly in a hybrid element [7]. Applications to date have involved only the isolated crack situation, but have also made use of another special hybrid element which incorporates the stress field around a crack tip [8]. However, a new formulation for a hybrid element containing MSD has been completed [9]; numerical implementation is now in progress, and results are expected in about six months. An element combining MSD and fastener load transfer will then be developed and is expected to be ready for application in about eighteen months.

The ability to model the curvature and bulging effects of fuselage panels also requires further research to make the extension to MSD. It is well established, through the results of comparative tests, that curved panels behave differently from flat panels, and that the curvature also interacts with cracks to affect properties such as damage tolerance. Flat-panel models must, therefore, be empirically calibrated by comparison with curved-panel tests in order to predict damage tolerance for fuselage structure. Flat-panel model correction factors have been developed for the prediction of tolerance to isolated damage, but MSD correction factors have not been developed. The research approach in this case is to develop better analysis methods which account for curvature effects based on established principles of mechanics and thus require fewer validation tests than the purely empirical approach.

Basic Fracture Resistance

A better understanding of basic fracture resistance properties is also required in order to predict the tolerance of panels to MSD. Panel fracture predictions are currently made by means of R-curve methods, which account for the fact that a large isolated crack in a thin ductile skin can undergo stable extension at stress levels below the fracture strength. The data from which material R-curves are derived come from tests of center-cracked panels or "compact" type specimens with either pin or wedge loading [10]. In all cases the specimen dimensions and length of the single crack are large if the material under test has appreciable ductility. The results of such tests can be applied to conventional damage tolerance assessments.

Conversely, calculations based on the N73711 post-accident observations suggest that conventional R-curves give unconservative estimates for critical crack length. With assumptions of 1.15-inch rivet pitch and skin bypass stress corresponding to design maximum pressure, a critical crack length of 1.1 inches (tip-to-tip) is predicted from a 2024-T3 R-curve, whereas a retired 737 airframe tested by Boeing appears to suggest that the critical crack length is only 0.66 inch. Thus, there is a need to obtain coupon type data which characterizes material fracture resistance in the presence of the joining details with which the MSD is associated. In fact, it is by no means clear that the R-curve approach is applicable to MSD linkup, which may be controlled by local plastic collapse (net-section type failure).

A program of laboratory coupon testing has been started to investigate the MSD linkup mechanism. The first series of test specimens were single-ply 2024-T3 tensile coupons 0.04 inch thick and 4 or 8 inches wide, with three open holes of 0.16 inch diameter at 1-inch pitch and colinear saw cuts simulating MSD (Figure 8). Load versus crack extension was recorded manually, with crack extension measured by means of an optical microscope. Loading to failure was conducted both from the as-fabricated state (blunt saw cuts) and after pre-cycling at low load to initiate a sharp fatigue crack from the central notches.[4] Stable crack extension was observed, but there appeared to be no significant difference in the behavior of the two types of specimens.

Preliminary analysis of the data suggested that the R-curve for the simulated MSD depends on initial crack length as well as extension, i.e., there is apparently no "master" curve independent of initial crack length, such as is found in conventional R-curve tests. Furthermore, it appeared that the results might be correlated by a net-section failure line based on remote stress. However, most of the data was apparently located above a 65 ksi (ultimate strength) net section line, suggesting the need for a better definition of the term "net section" as applied to the testing of MSD simulation coupons.

The simple definition of the net-section line might have been made invalid by the transfer of load to the uncracked ligaments between the side holes and the coupon edges as the interior cracks extended. Consequently, an 8-inch wide specimen with 7 holes was tested to reduce the edge ligament influence, and the result for this coupon did fall on the ultimate-strength net-section line. Figure 9 summarizes the test results to date. The 8-inch/7-hole coupon series is being continued to provide the data required for a better interpretation of what is meant by "net section" in a test coupon (versus MSD in a structure). Additional series representing lap and butt splice details will be investigated later. Sufficient results for the characterization of lap-splice details are expected in about one year.

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4 In this case, the side holes were notched after fatigue.

Even if the R-curve approach is ultimately found to be inapplicable to MSD, it is still useful to have load versus crack extension data for the purpose of calibrating a plastic collapse model. However, observing crack extension through the optical microscope has thus far proved to be a difficult task because the extension increments are quite small. The raw data in most cases contains apparent jumps in crack length, a phenomenon which is believed to result from the difficulty of optically resolving small extensions and/or crack tunneling in the specimen bulk underneath the surface cladding.

In view of these difficulties, investigation has been started into the AC potential drop (ACPD) method as a possible alternate for measuring crack extension. The DC potential drop (DCPD) method is a well established technique for making real-time measurements of crack length in standard laboratory specimen fatigue tests but is not suited for dealing with MSD. The ACPD method is not yet a proven technique but does show promise of providing the resolution and focus required for making MSD crack extension measurements.

In an initial trial, ACPD measurements were correlated with microscope measurements in a slow crack growth experiment to establish a calibration factor and resolution of 0.6 mil/mv, using one of the 4-inch/3-hole coupons. The coupon was then loaded to failure, and a load versus ACPD extension curve was recorded on an X-Y plotter while microscope measurements were also made. The load versus ACPD extension curve (Figure 10) exhibited a large anomaly in the ACPD signal from the beginning of the test up to a load of 4,300 lb but then appeared to behave properly. At 4,800 lb the first visible extension (2 mils) was observed. The test was continued up to 5,550 lb (30 mils visual extension), at which point linkup occurred. Comparison of the ACPD and microscope measurements indicated that the ACPD signal generally followed the visual observations in the 4,800 to 5,550 lb load range, but both bias and calibration errors were present (Figure 11). The ACPD data is suspect because of the initial anomaly (which remains unexplained), and further trials must be conducted in order to evaluate the technique. Experimentation with ACPD procedures will be continued in parallel with the coupon test program.

HOW TO INSPECT FOR MSD

Others have said that there is no reason why a modern airframe should not continue to be flown indefinitely, provided that it is properly maintained and inspected. The second part of this statement is the key phrase for aging airframes with MSD potential. Preliminary tests and calculations based on the N73711 experience and subsequent tests by Boeing suggest that MSD must be detected at quite small crack lengths and in much shorter time than an isolated crack, if the MSD is to be found and repaired ahead of linkup and fracture.

The detection requirements for MSD preclude reliance on visual inspection. The only alternative which has been reduced to practice in airline maintenance shops and repair stations involves the use of hand-held eddy current probes. The eddy current method is technically reliable but tedious to apply, leading to excessive downtime and human factors problems. Semi-automated eddy current inspection procedures are available to reduce these problems to some extent (with some sacrifice of technical reliability) but are at best imperfect solutions.

Better Nondestructive Inspection

Better nondestructive inspection (NDI) methods must be sought to arm the airlines with procedures which are both proper for MSD and economical to apply. The FAA and NASA are pursuing complementary research programs in the NDI area. The NASA program focuses on the long-term goal of advancing NDI technology. The FAA program focuses on the short-term objective of surveying existing alternatives and reducing the most promising candidates to practices which can be applied to airframes. Thermal imaging, laser holography, and magneto-optic field imaging are the current short-term candidates.

A simple laboratory demonstration has shown that the infrared (IR) signature of a 1-inch through crack in a 0.1-inch thick 7075-T6 coupon was easily detected by a videocamera after the coupon surface was treated to increase its emissivity and a mild heat source was applied to establish a steady flux along the surface and across the crack plane. Whether this technique has the sensitivity to detect MSD remains to be seen. The basic requirements for enhanced surface emissivity and heat flux could be met in practice by means of water-soluble paints if necessary and external heat sources, both of which could be temporarily applied to an airframe during a scheduled inspection. The IR videocamera appears to be a highly efficient inspection tool that can provide a 20 square foot field of view at a 0.1-inch resolution. Consideration of the procedure outline suggests that the thermal imaging method requires foreknowledge of where to look, and thus that its practical application is likely to be limited to specific inspection requirements such as those in the recent ADs issued on certain 727, 737, and 747 lap splices.

Laser holography and magneto-optic imaging are emerging methods which have the promise of unrestricted coverage of the structure. Laser holography requires application of load to the structure, but in principle the requirement can be met within the limit load envelope. The holo-topographic or magneto-optic interference patterns can be used to identify suspect local areas requiring more detailed eddy current NDI. It remains to be seen whether either of these methods will prove to be practical for detecting MSD in transport fuselages.

The FAA Technical Center has recently completed detailed eddy current NDI on the fuselage of N40, a Boeing 727 in the FAA test aircraft fleet. N40 is now being used as a test bed for trials of the candidate alternate NDI methods. About three months of work will be required to make preliminary evaluations and develop the initial procedures. Some of the curved panel test articles in the tear strap effectiveness program may also serve as test beds for refinement of the procedures, and the FAA is considering the acquisition of a retired airframe to serve as a test bed for full-scale procedures evaluation.

In parallel with the NDI development, a program has been started to collect and document a permanent archive of test specimens containing various types of damage (corrosion, disbonds, isolated cracks, and MSD). The FAA plans to make this archive available to other organizations for evaluation of new methods and procedures.

Proof Testing

About one thousand 727 and 737 aircraft currently under ADs following the Aloha Airlines accident are now in periodic eddy current NDI programs to protect against in-flight MSD linkup. In mid-1988 a number of the independent experts who advise the FAA on structural integrity suggested that proof testing be considered as a backup option, to be implemented if the NDI program should prove to be ineffective, until the terminating actions specified by the ADs have been taken on those airframes. The proposal recommended a pressure test to limit load, i.e. 1.33 times the design maximum pressure differential (1.33P).

The proof test is an appealing concept because it appears to eliminate the uncertainties of NDI by establishing a precise upper limit on existing crack size, relative to the critical size in flight. The Air Force successfully applied proof tests to deal with a specific problem in the F-111 fleet and (coupled with payload and flight restrictions) to extend the life of the aging wing structure in the B-52D fleet [6]. However, neither of these experiences prove that a pressure proof test of transport fuselage structure is either feasible or practical.

The proof test proposal was evaluated by means of a damage tolerance analysis [11]. The Boeing 737 fuselage structure was selected for this study because of the availability of MSD crack growth data pertinent to the type of situation for which the proof test had been proposed. Preliminary R-curve properties derived from the FAA laboratory test program data were used to estimate the critical crack lengths corresponding to proof and maximum service pressures. A range of proof pressure was studied, not only for the effect on post-test safe crack growth interval, but also to investigate the potential for stable crack extension during the test itself.

The study author recommended 1.5P as the optimum proof pressure, the lowest level at which he estimated MSD would be either unextended or driven fully to obvious partial failure (one or two bay crack with flapping).[5] However, the safe inspection interval corresponding to a 1.5P test was estimated to be no more than 600 flights.[6]

After reviewing the study results, the ATA and the manufacturer concluded that proof testing should not be considered as a backup option because of the impractically short safe inspection interval and the fact that the recommended pressure would exceed limit load. The FAA has abandoned further research on the question of fuselage proof testing, based on the condition that further testing and evaluation would be necessary, should the proof test option be seriously considered in the future.[7]

HOW TO DESIGN FOR MSD RESISTANCE

Besides responding to the problems associated with MSD potential in existing fleets, the FAA research program also has the objective of fostering improvement of design practices to avoid MSD in the future. This goal applies to repairs as well as new designs.

Approach

A conceptual model has been developed to rank the MSD potential of alternative design details [12]. The model combines fatigue and damage tolerance analyses in a practical engineering tool which yields a quantitative risk parameter. The parameter is $\alpha\Delta T/\beta e$, where α and β are respectively the shape and scale parameters

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5 At lower pressure, such as 1.33P, he found an unacceptable level of risk that a proof test could drastically shorten post-test life by causing a stable crack extension of less than one bay, a non-obvious damage state.

6 This interval would be equivalent to requiring a proof test about every three months, based on typical figures for 727 and 737 flight hours per year and landings per flight hour. In principle, a longer interval could be set by using a higher test pressure, but the author's analysis also suggested that the risk of uncontained fuselage failure would become unacceptable at pressures not much greater than 1.5P.

7 The requirements essentially follow the FAA plan for similar testing of "generic" structures. Tests of curved panels or full scale components would be required in sufficient quantity to establish: (1) the critical MSD size corresponding to the proposed proof pressure; (2) the critical MSD size corresponding to limit load in service; and (3) the safe growth life of MSD from just below test-critical size to limit-load-critical size, when the damage is subjected to one proof pressure cycle followed by simulated flight cycles at limit service pressure.

of a Weibull model of the detail fatigue life distribution, and ΔT is the slow growth life to propagate a crack from adjacent critical to primary critical length.[8]

While the model statement is quite simple, its application requires extensive replicate testing of realistic details subjected to realistic stress environments to provide for appropriate estimates of the fatigue life parameters. A flat-panel test program is planned for this purpose. The first series of flat panels will complement the first series of curved panels in the tear strap effectiveness program in order to allow for validation of fidelity in the flat-panel stress fields.

The curved panel design (Figures 6 and 7) has been stress analyzed, with the results for mid-bay hoop stress as shown in Figure 12. The mid-bay hoop stress has the greatest axial variation and consequently poses the biggest challenge to the designer of flat test panels. Figure 13 illustrates four alternative flat-panel designs which have been evaluated by means of comparative stress analysis (Figures 13 through 17). A fatigue test program intended to characterize the life distribution in a mid-bay splice would consist of several replicates of design "A" (to establish the characteristics of the individual fastener/skin detail as a function of stress level) and several replicates of design "D" (to establish the behavior of the bay detail as a whole and to provide guidance for cascade scenarios). The actual test program is planned to investigate an axial splice over a stringer, where the variation of hoop stress is less than at mid-bay. Later test series will keep pace with the investigation of other types of splice details in the tear strap effectiveness program.

Repair Practices

Many major repairs to airframes are designed and installed by airline maintenance or repair station personnel. The repair designs generally possess static strength equal to the original airframe static strength. Such repairs can become sites for later fatigue damage in the repaired or an adjacent area, unless damage tolerance as well as static strength is considered in the repair design. However, most maintenance organizations are not equipped to conduct elaborate damage tolerance assessments.

The research objective in this area is to provide a manual for damage-tolerant repair design. Fracture mechanics analyses of typical doubler-type repairs will be used to identify critical design variables and their damage sensitivities. The analyses will

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8 These two crack lengths are loosely defined as the limits for a group of MSD cracks which may participate in a linkup cascade. The primary crack is the largest crack in the group. The adjacent crack is the smallest crack that will join the cascade as a result of load transfer during linkup.

be validated by means of flat and curved panel tests and will then be used to produce design charts for the manual. The target date for completing the manual is October 1992.

A part of this activity will also examine the question of whether terminating actions, such as those specified in the current fuselage lap splice ADs, are truly terminating. It has been suggested that fatigue cracking might re-appear in a repaired splice (perhaps at a fastener row not involved in the original MSD), and some flat-panel simulation tests will be conducted to shed light on this question. Results are expected in about 18 months.

Interference-Fit Fastener Effects

In the long run, the resistance of new designs to MSD will depend to a large degree on the condition of panel skin in the vicinity of holes filled with interference-fit fasteners. The ability of such details to resist fatigue or corrosion damage depends upon the interaction of residual stresses (established for beneficial purposes by the interference) with flight stresses. Although a great deal of empirical work has been done on fastener effects, there has never been a systematic study of the stress interactions, especially with a view to rating the MSD potential. Flat-panel tests will be conducted for this purpose, with the goal of providing basic information for the manual in about 36 months.

CONCLUDING REMARKS

The FAA research program on MSD in aging airframes is fast-paced and realistic. It addresses major issues which were raised in the aftermath of the N73711 accident. We hope that the industry will study the program results, as they are developed, and take the final step of reducing those results to practice for specific designs and maintenance actions.

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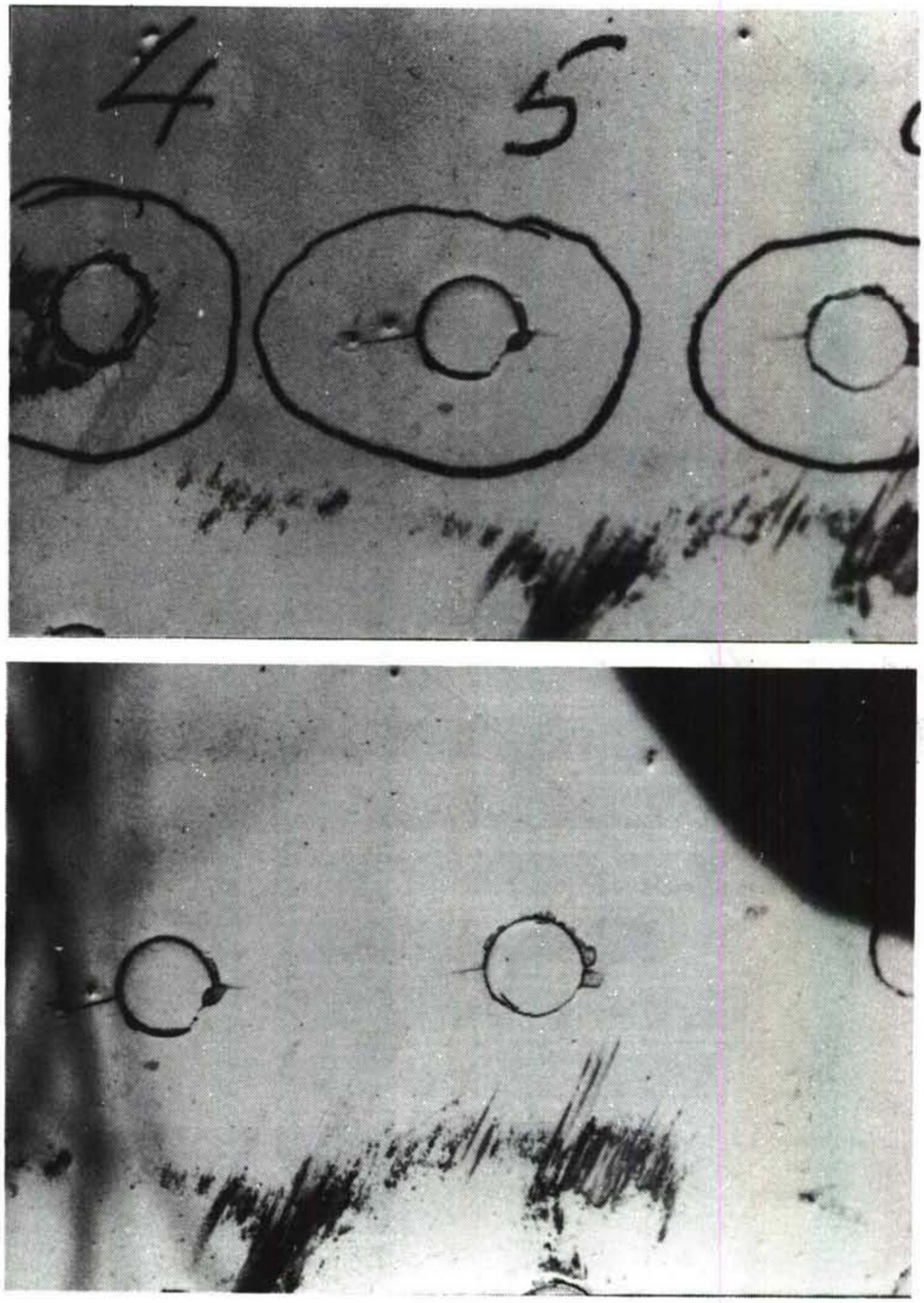

FIGURE 1. MULTIPLE SITE DAMAGE IN N73711

Photographs of several rivet details in two areas of the N73711 fuselage adjacent to the in-flight failure area. Several MSD cracks are visible. (Photographs courtesy of T. Swift, FAA.)

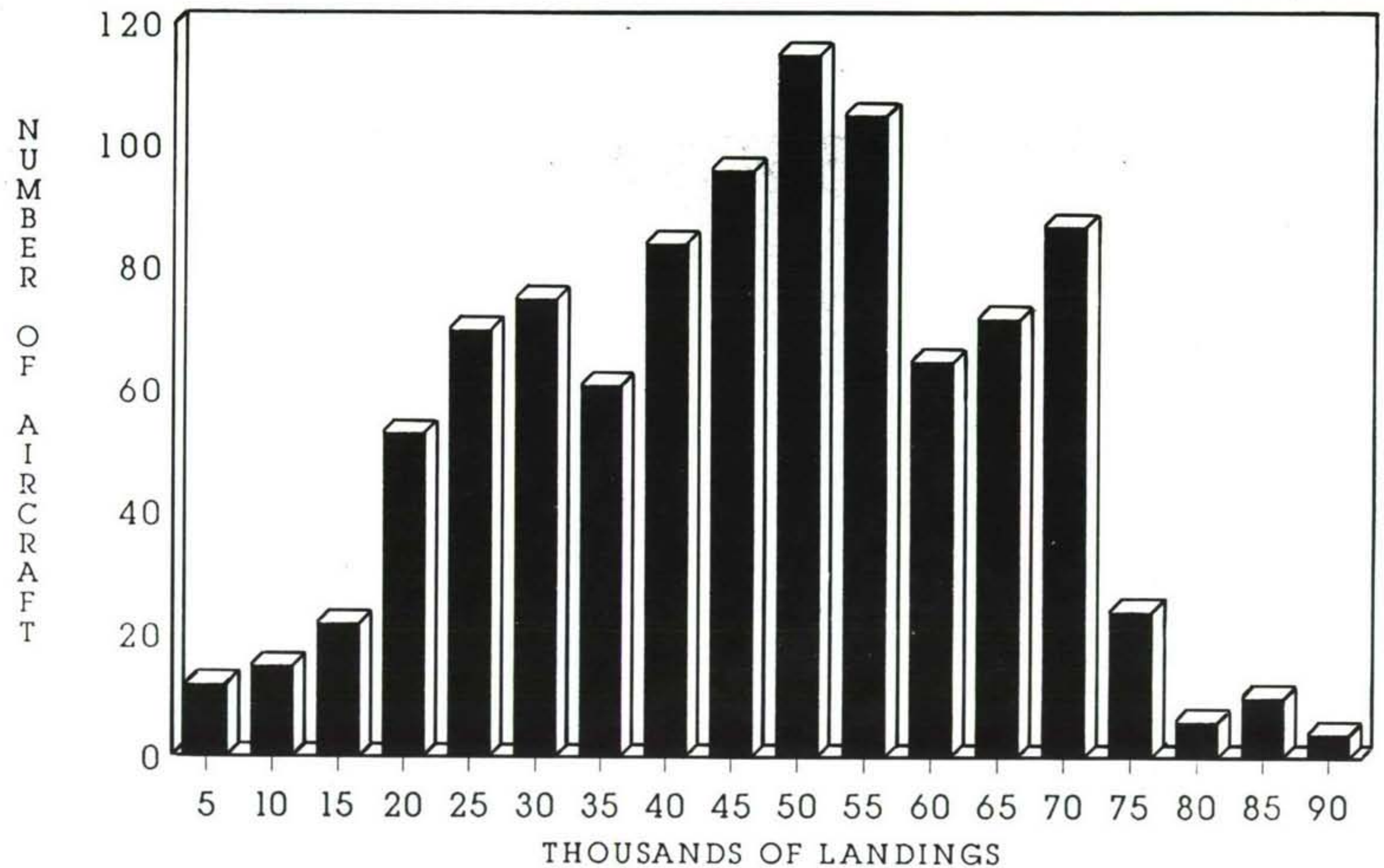

FIGURE 2. DC-9 FLEET AGE STATUS

The data reflect the fleet as of October 31, 1988. The DC-9 design goal was 45,000 landings, each assumed to represent a maximum service pressure cycle on the fuselage.

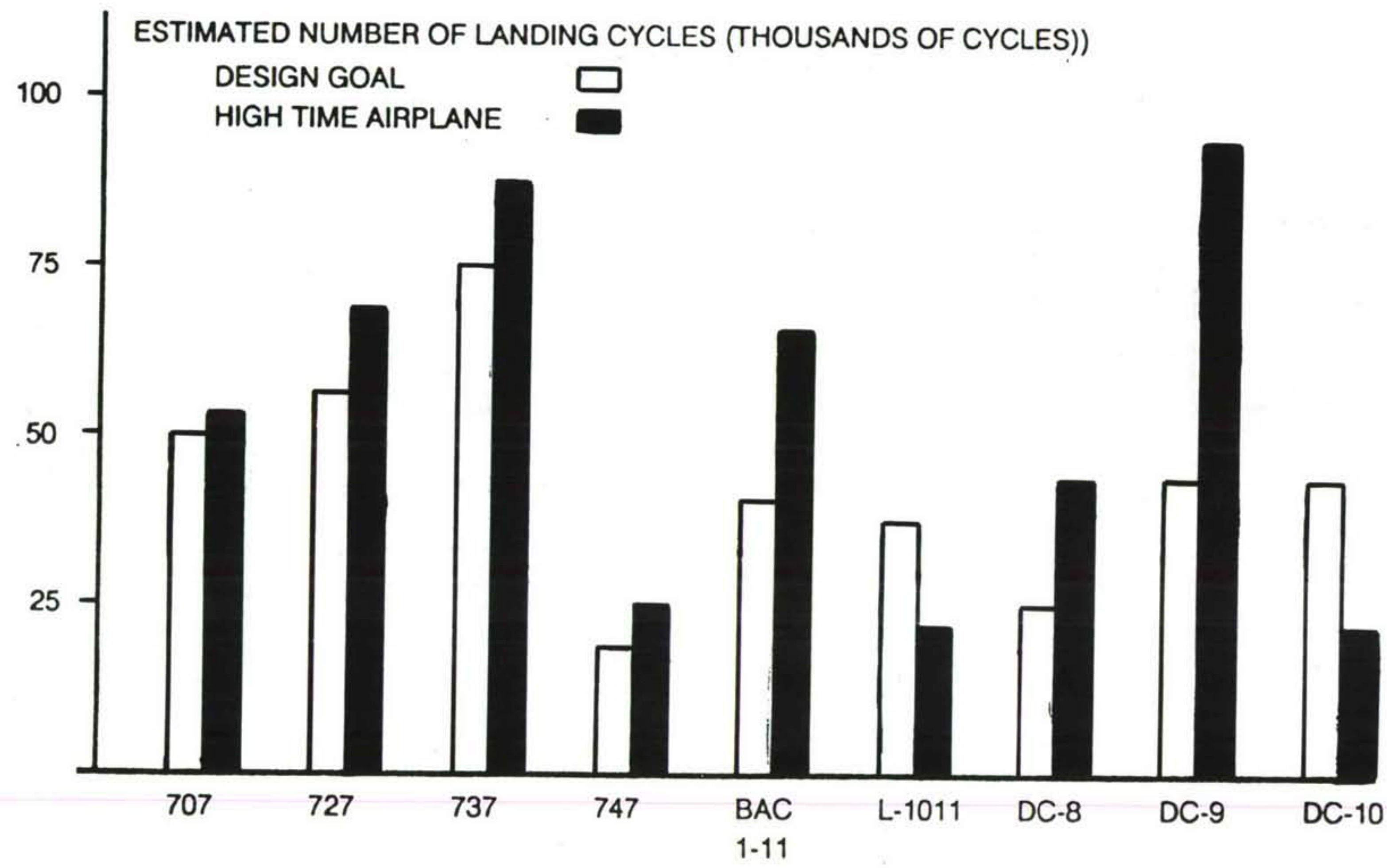

FIGURE 3. COMPARISON OF HIGH-TIME AIRCRAFT WITH DESIGN GOALS

As of October 31, 1988, the high-time aircraft has exceeded the design goal for all models shown except the L-1011 and DC-10.

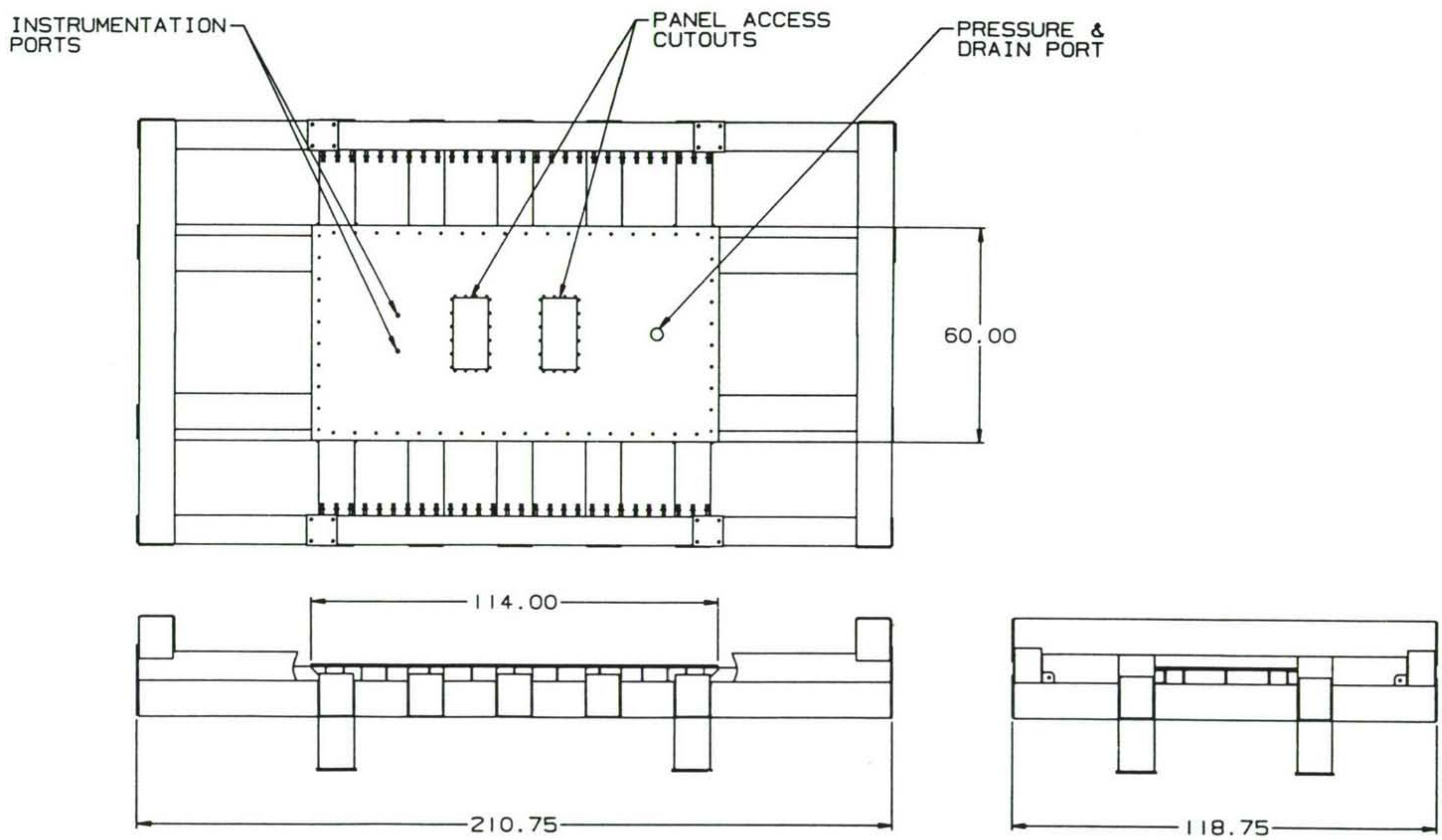

FIGURE 4. CURVED PANEL TEST FIXTURE SCHEMATIC

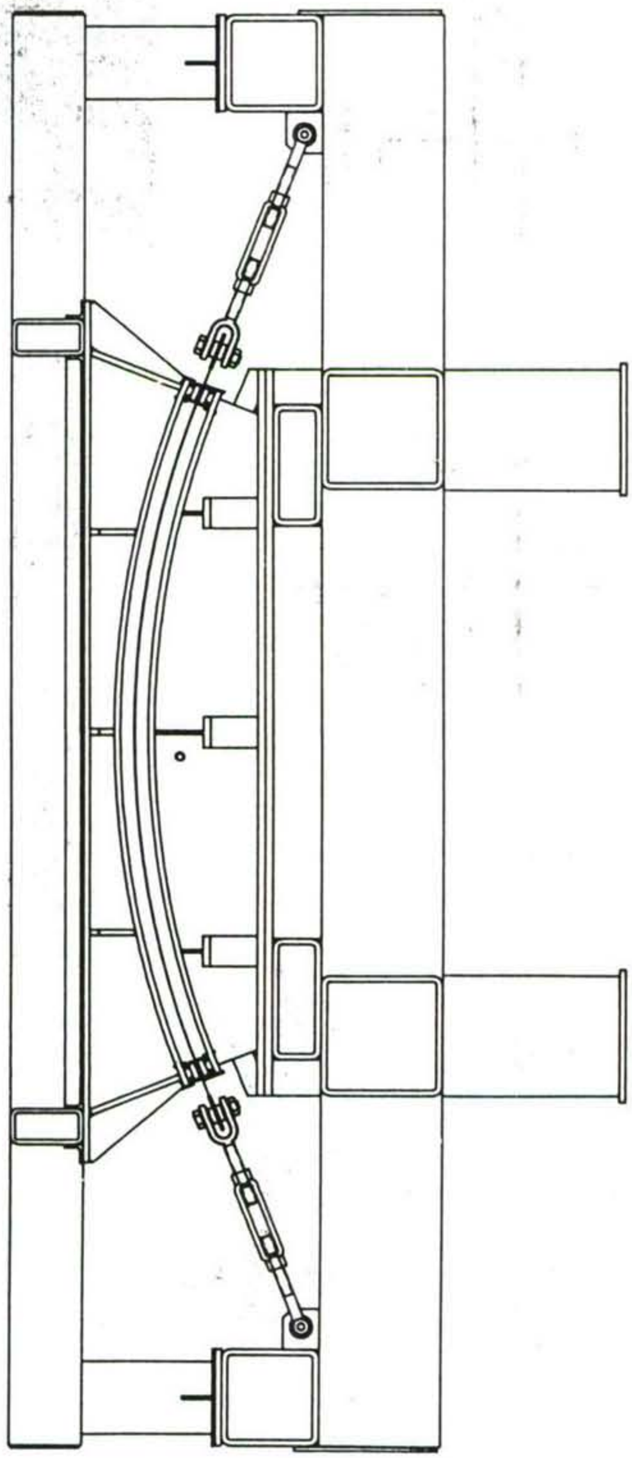

FIGURE 5. END-VIEW DETAILS OF TEST FIXTURE

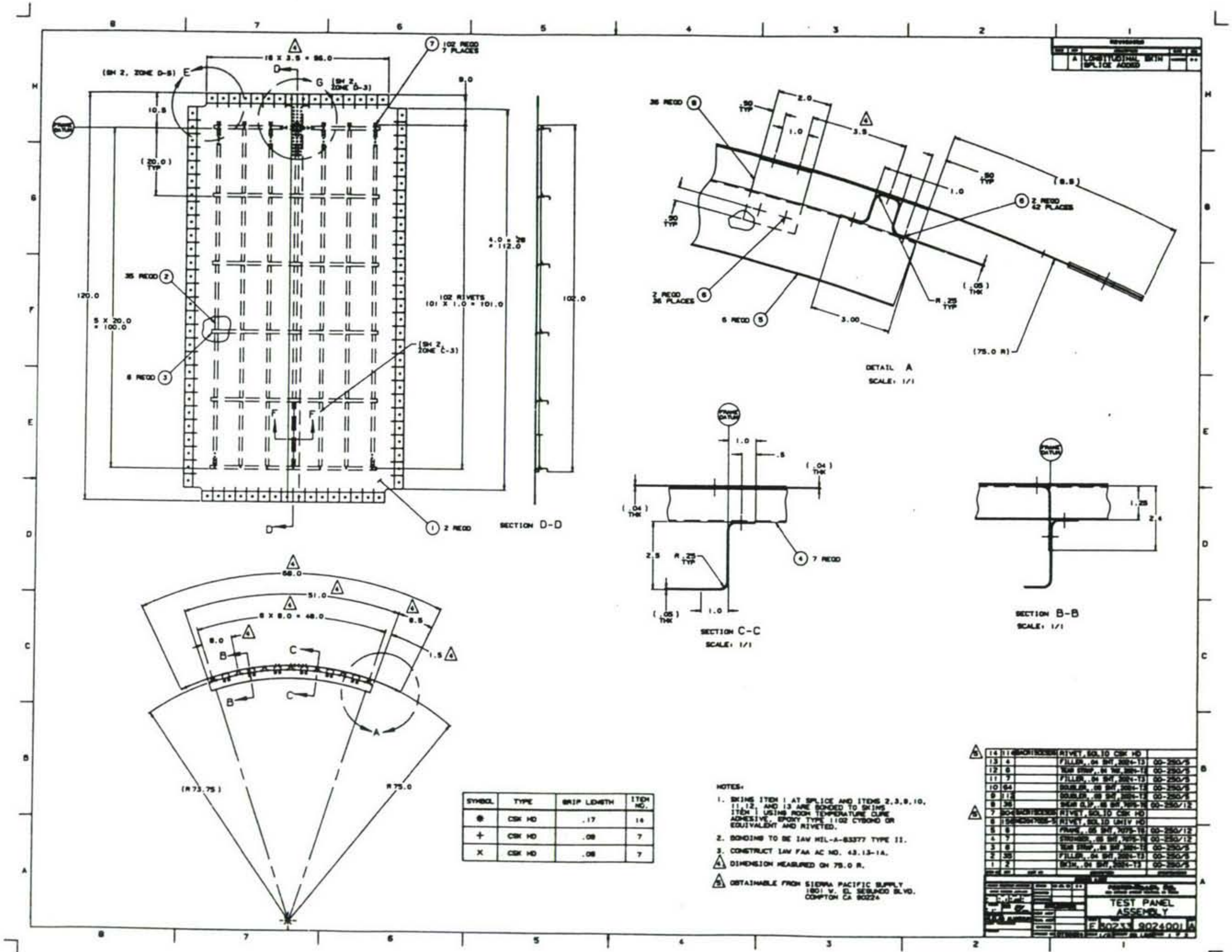

FIGURE 6. DESIGN FOR FIRST PANEL TEST SERIES

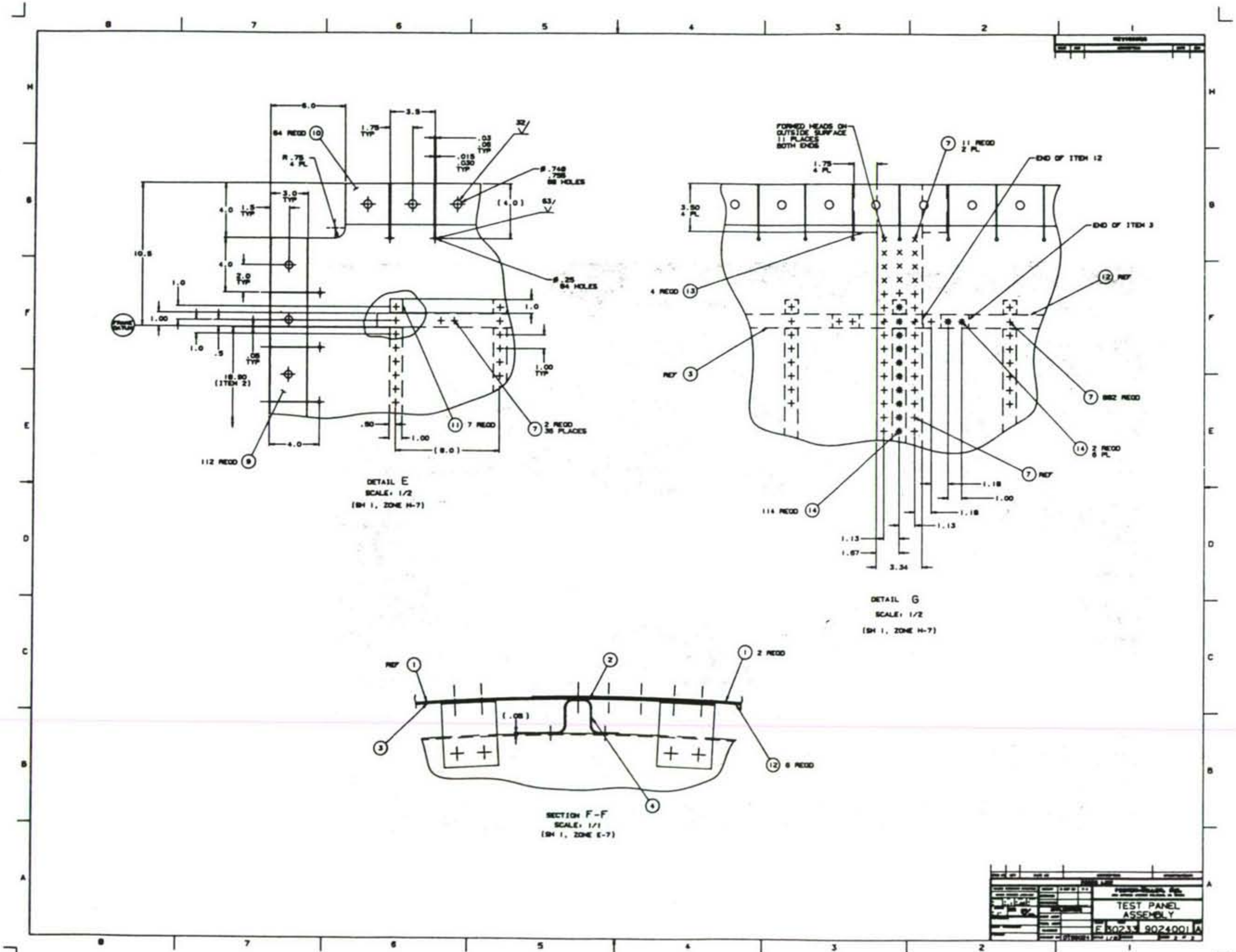

FIGURE 7. PANEL DESIGN DETAILS

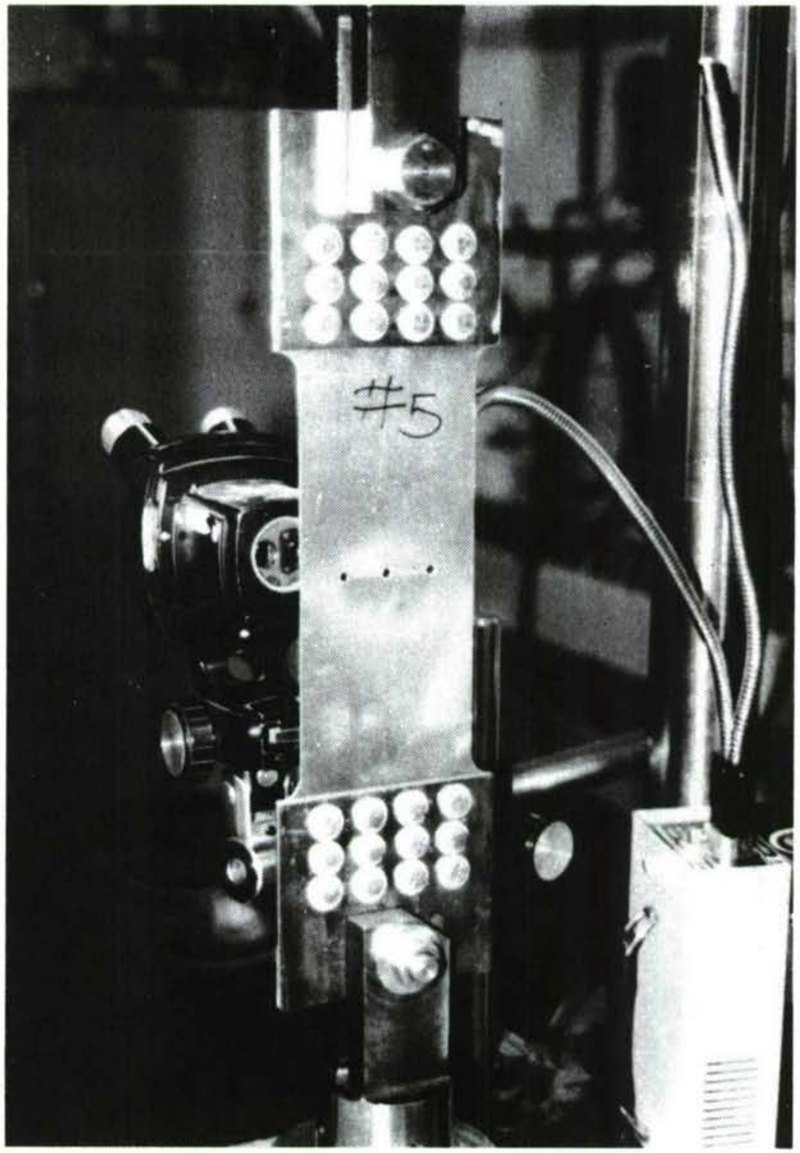

FIGURE 8. SIMULATED MSD COUPON SPECIMEN

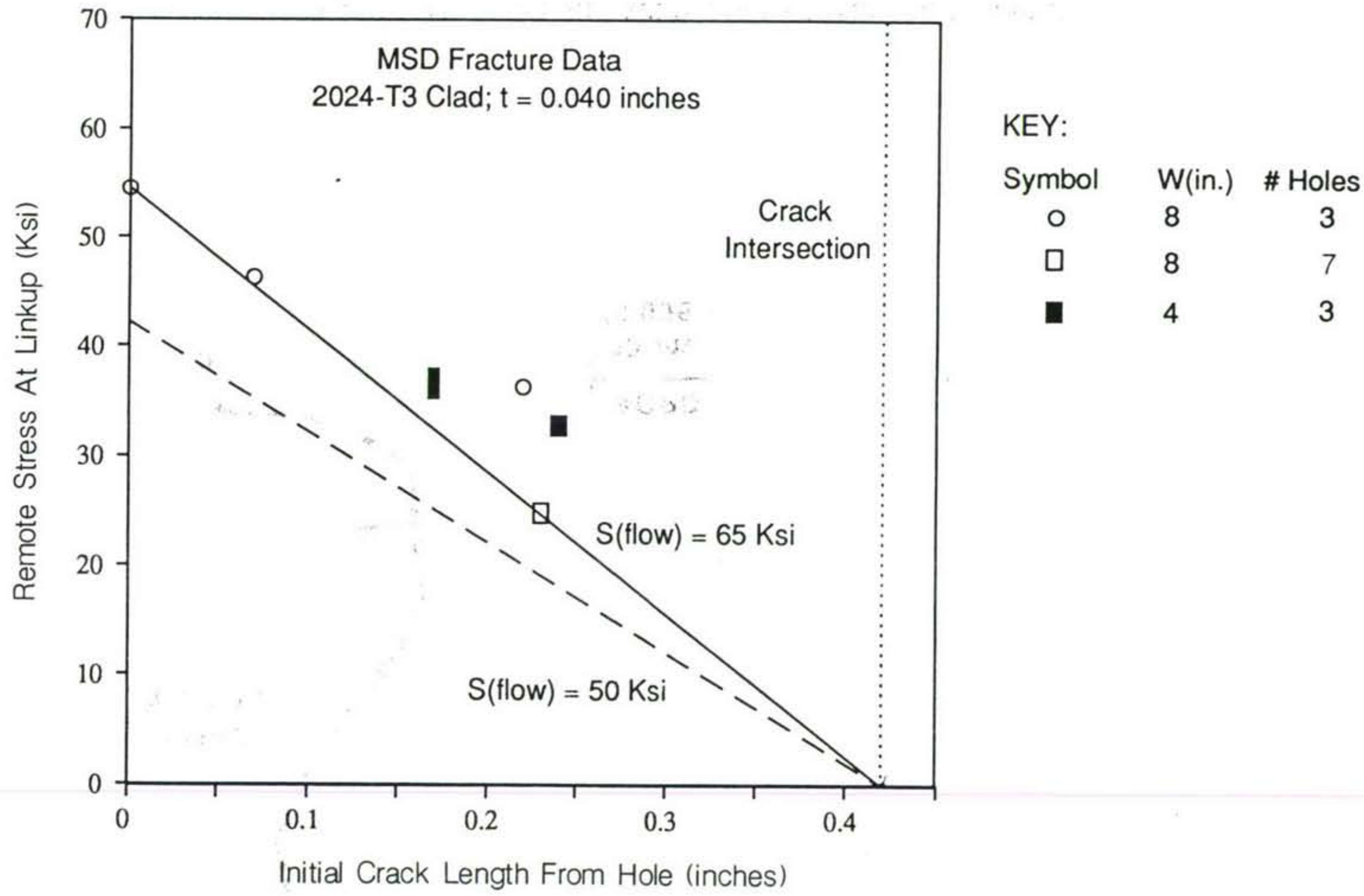

FIGURE 9. COUPON TEST RESULTS

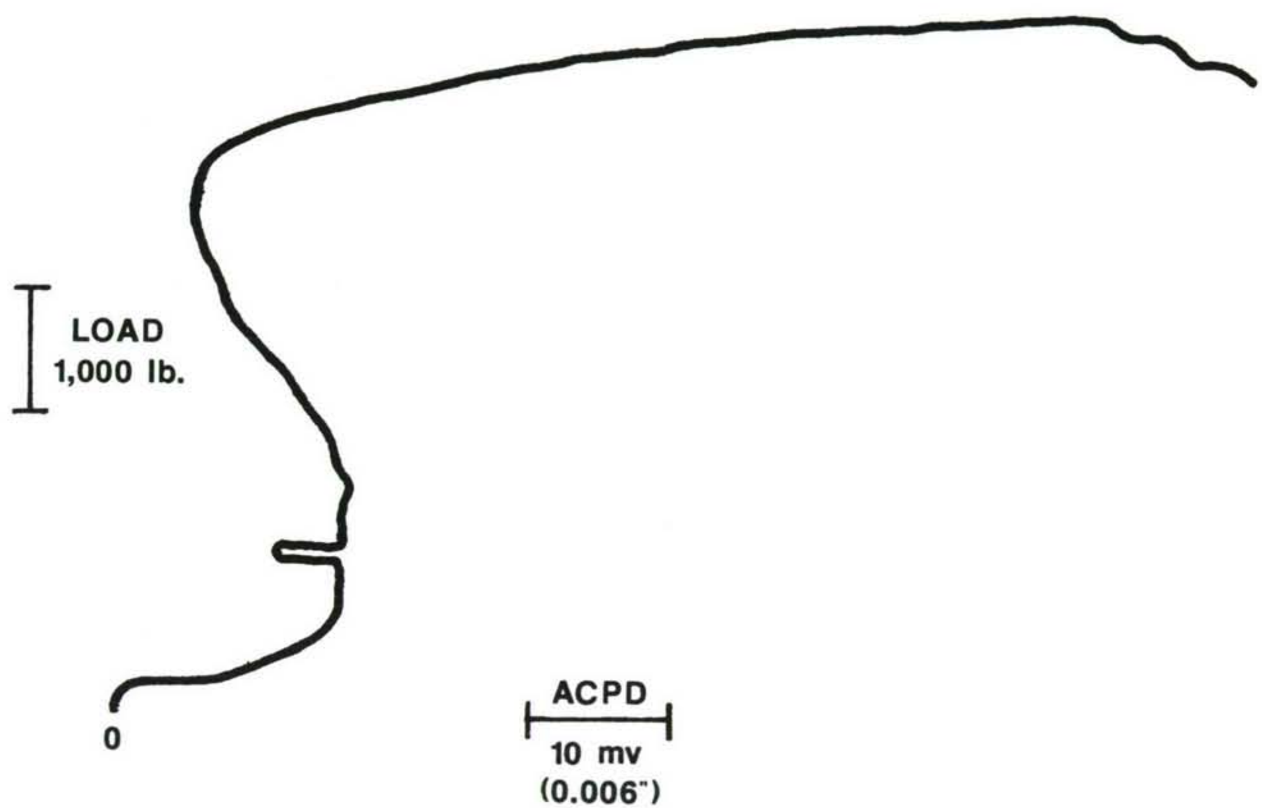

FIGURE 10. TRIAL ACPD LOAD-EXTENSION CURVE

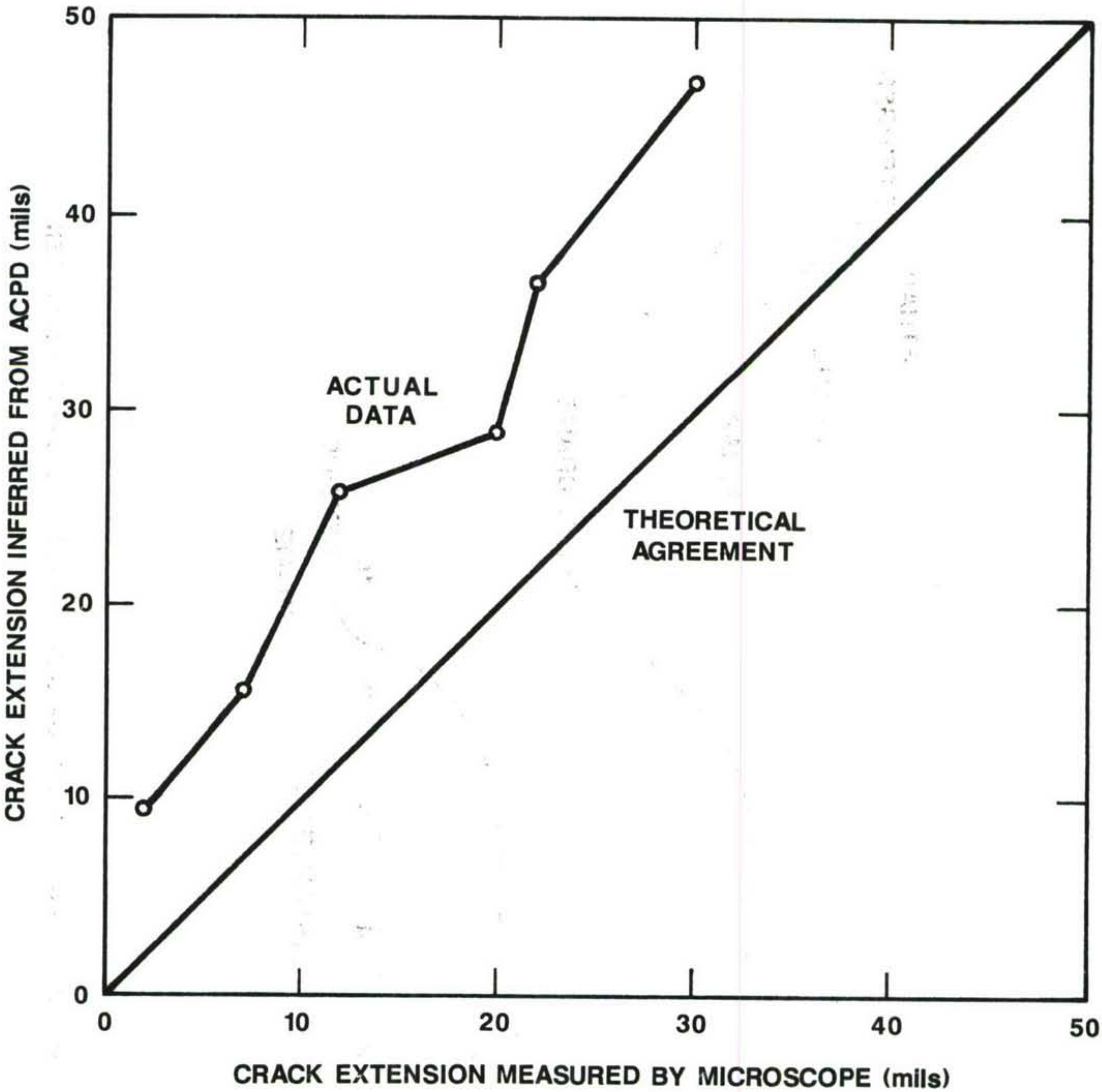

FIGURE 11. COMPARISON OF EXTENSION MEASUREMENTS

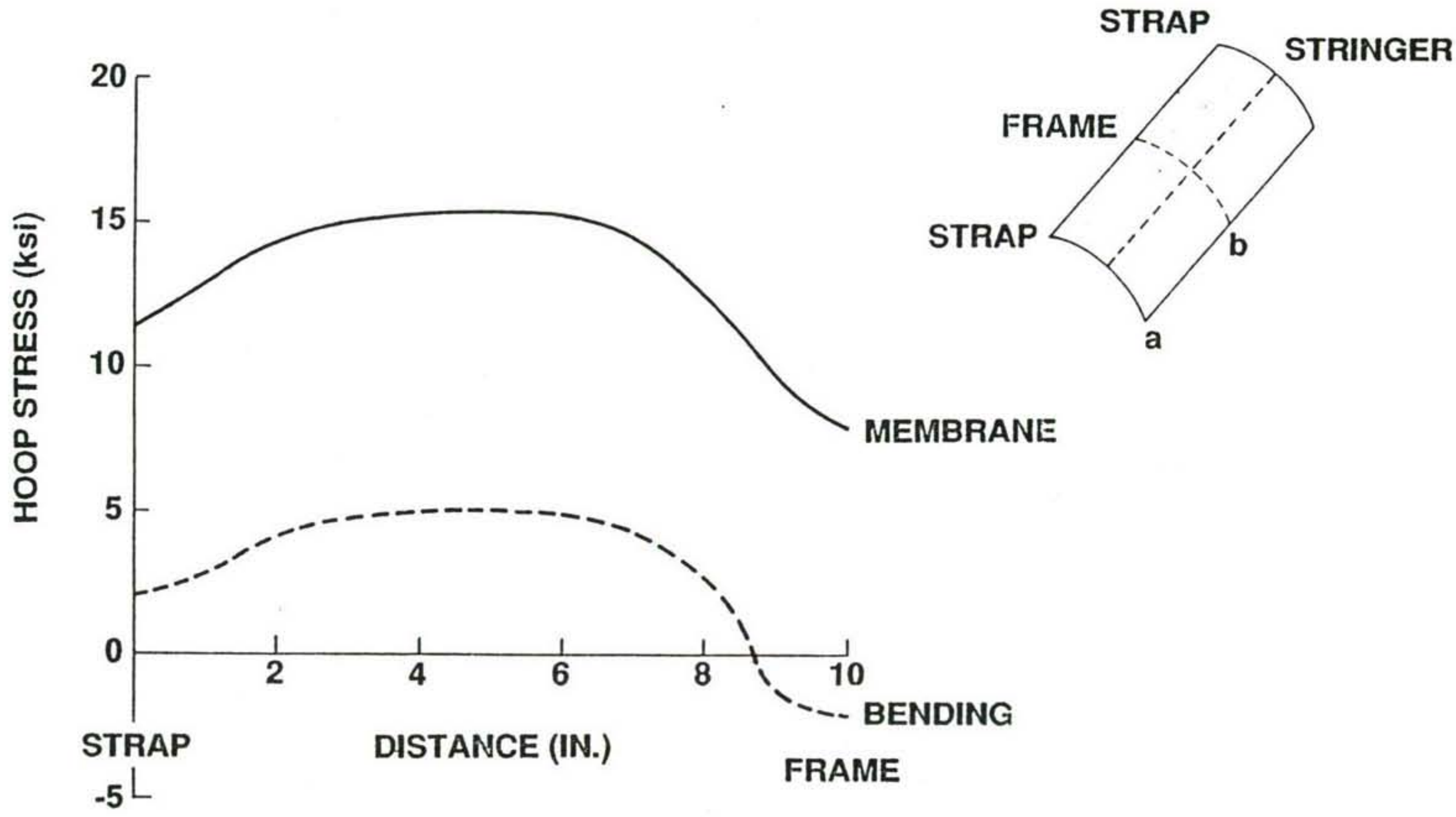

FIGURE 12. PANEL SKIN HOOP STRESS ALONG MID-BAY AXIAL LINE

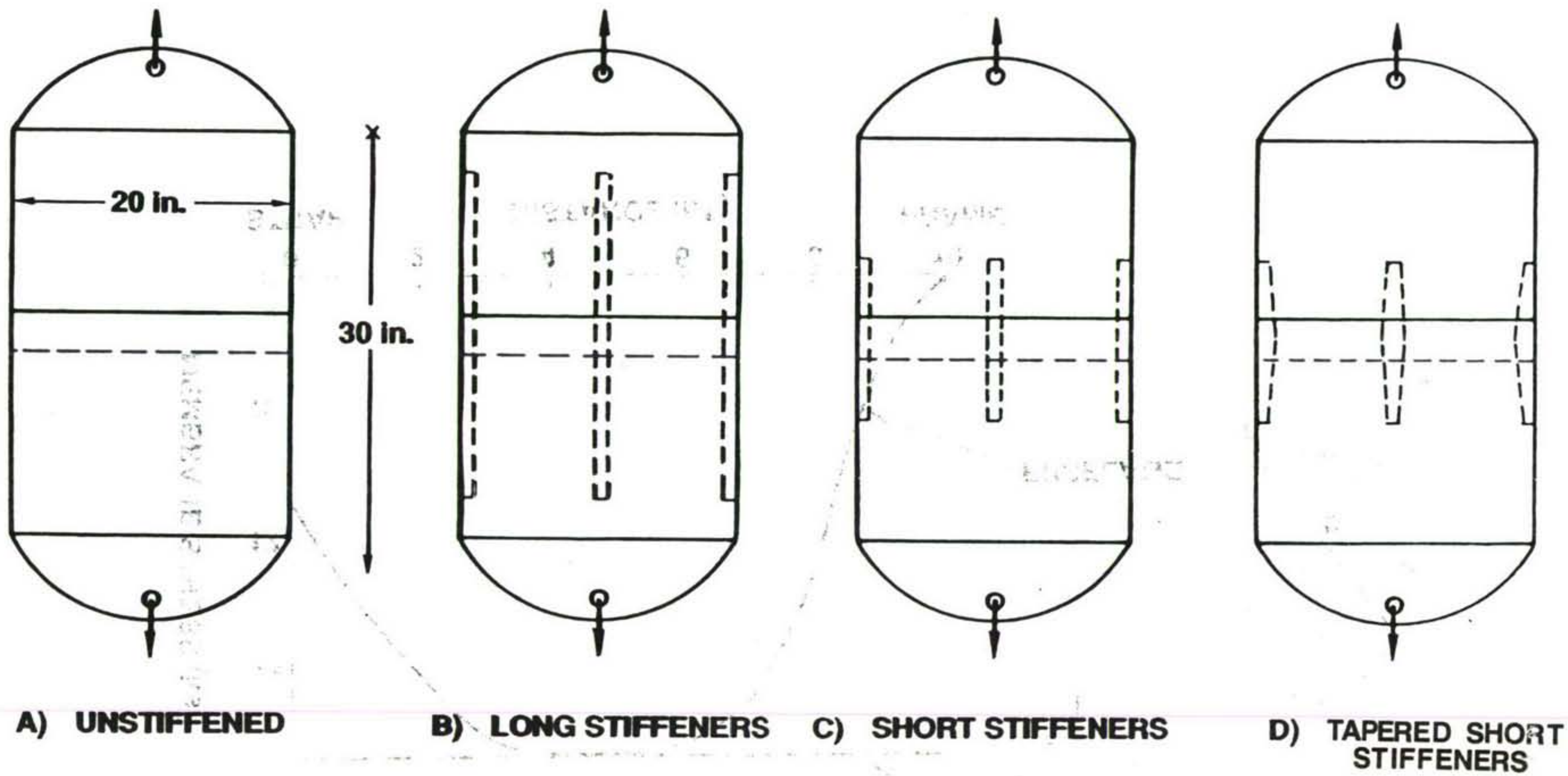

FIGURE 13. FLAT PANEL DESIGNS

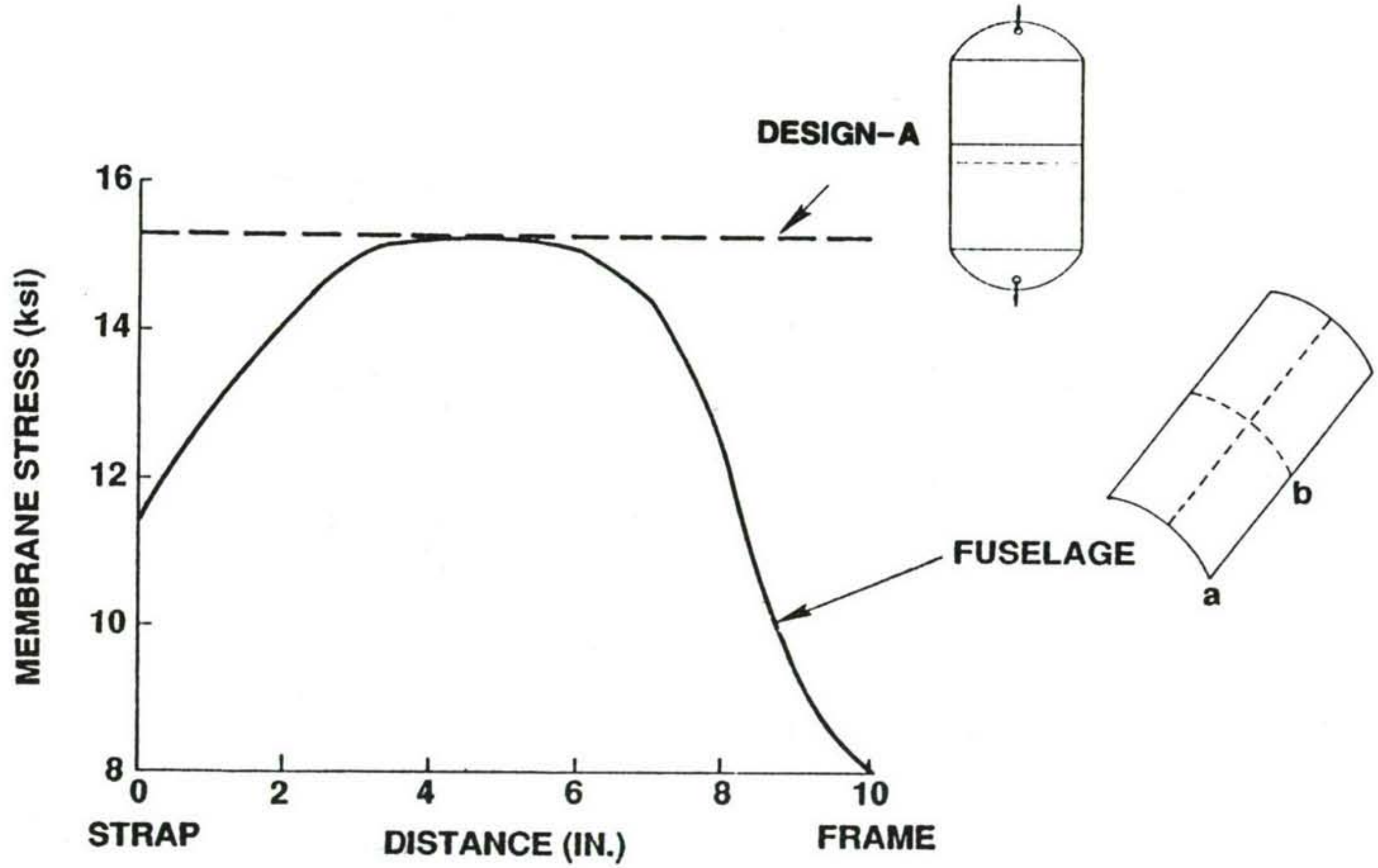

FIGURE 14. PERFORMANCE OF DESIGN "A"

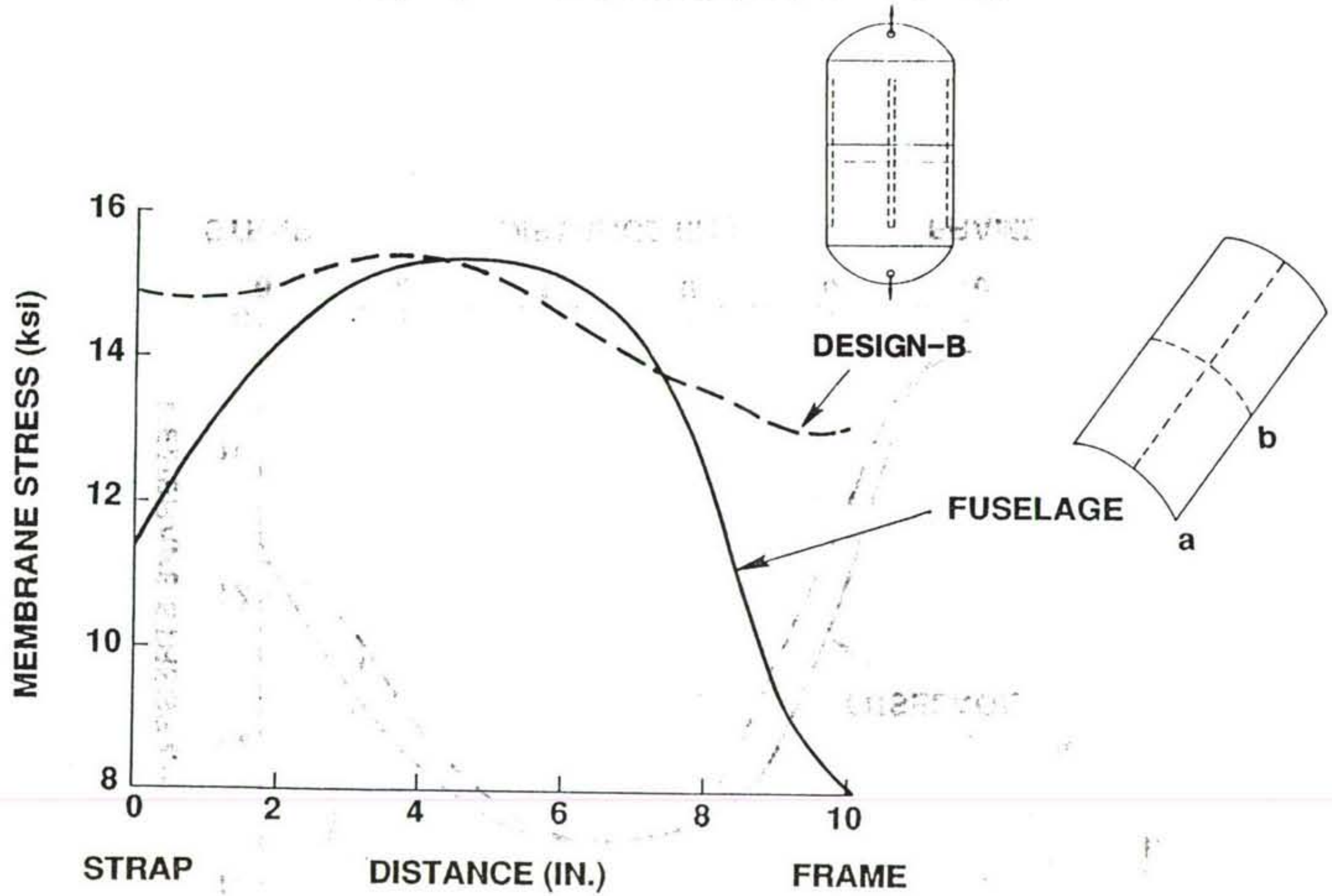

FIGURE 15. PERFORMANCE OF DESIGN "B"

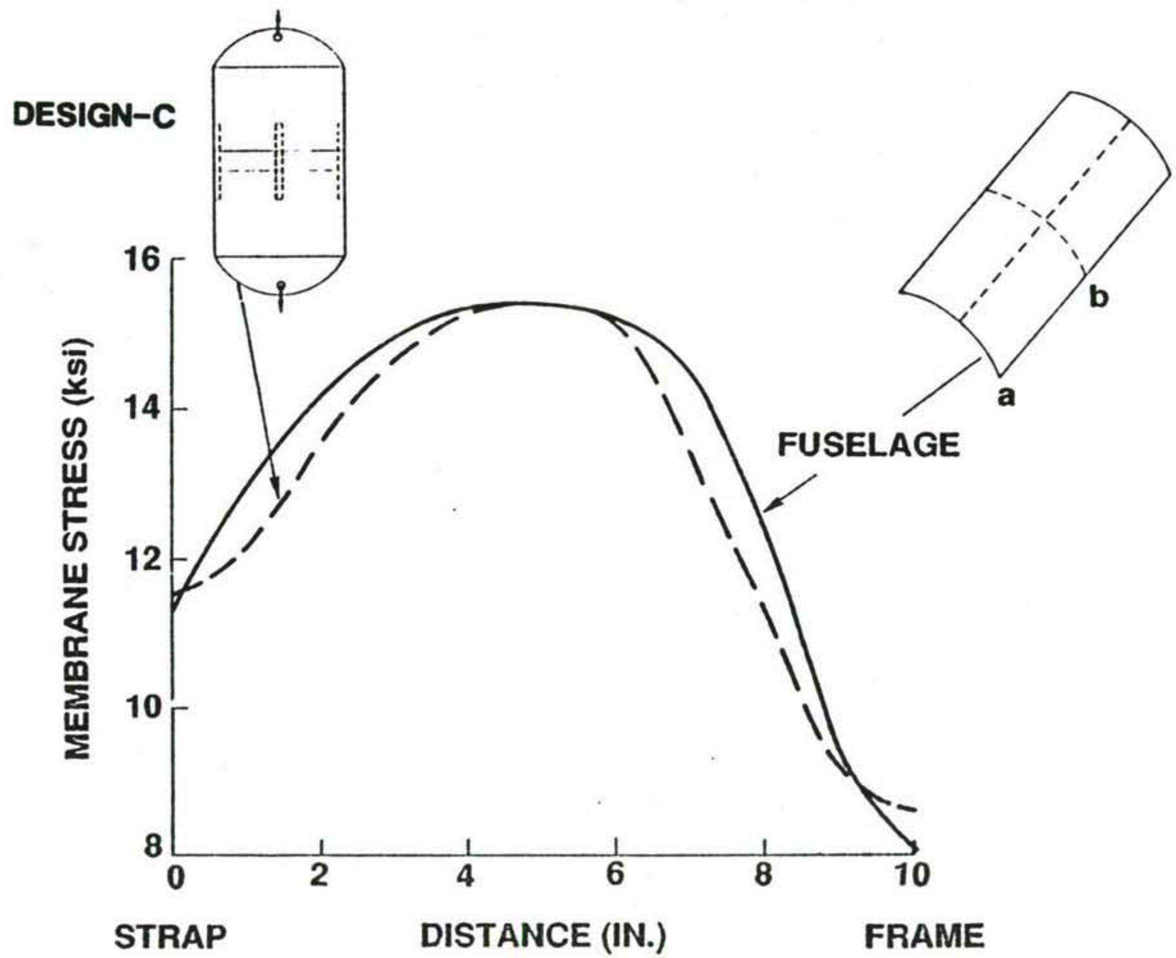

FIGURE 16. PERFORMANCE OF DESIGN "C"

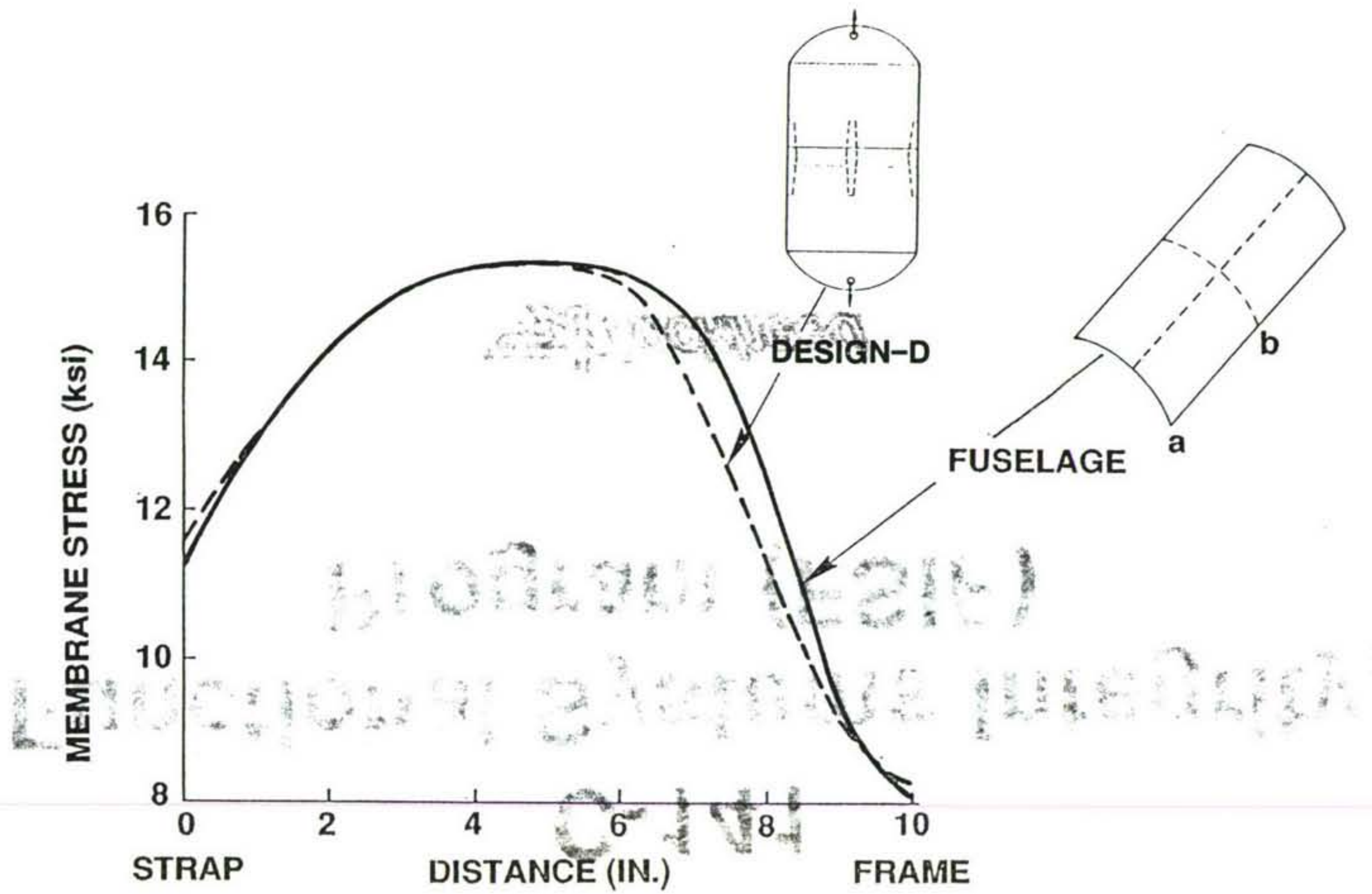

FIGURE 17. PERFORMANCE OF DESIGN "D"

C-141
Functional Systems Integrity Program (FSIP)

Lockheed

GA-8870-1
11-14-89

Acronyms

C-141 FSIP

- ACI - Analytical Condition Inspection
- AIRS - Aircraft Information Retrieval System
- ARIES - Automated Readiness Integrated Engineering System
- ASIP - Aircraft Structural Integrity Program
- CA/IP - Condition Assessment/Improvement Program
- DBMS - Database Management System
- FMP - Force Management Plan
- FSIP - Functional Systems Integrity Program
- PDM - Programmed Depot Maintenance
- PDP - Program Decision Package
- R&M - Reliability and Maintainability
- TSIP - Tailored Systems Inspection Program
- WSMP - Weapon System Master Plan

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BACKGROUND

Over the past fifteen years considerable analysis has been performed on the C-141 structure. The Aircraft Structural Integrity Program (ASIP) program is well defined and has been very successful in predicting problems relating to fatigue and corrosion in the main structural areas of the aircraft.

Recognizing that the functional systems have not been under the same scrutiny as the structure, Lockheed and Warner Robins initiated a program, called the Condition Assessment/Improvement Program, in 1986 to assess the current condition of the fleet with respect to safety of flight and mission essential functional subsystems and components.

To date fourteen aircraft have been evaluated. Conclusions and recommendations with respect to the conditions of C-141 systems are being made from the knowledge gained in the CA/IP program. A comprehensive Functional Systems Integrity Program using CA/IP experience is being developed to ensure that the functional systems can be managed at the same level as the structure.

Background

C-141 FSIP

- Aircraft Structural Integrity Program (ASIP)
- Condition Assessment/Improvement Program (CA/IP)
- Functional Systems Integrity Program (FSIP)

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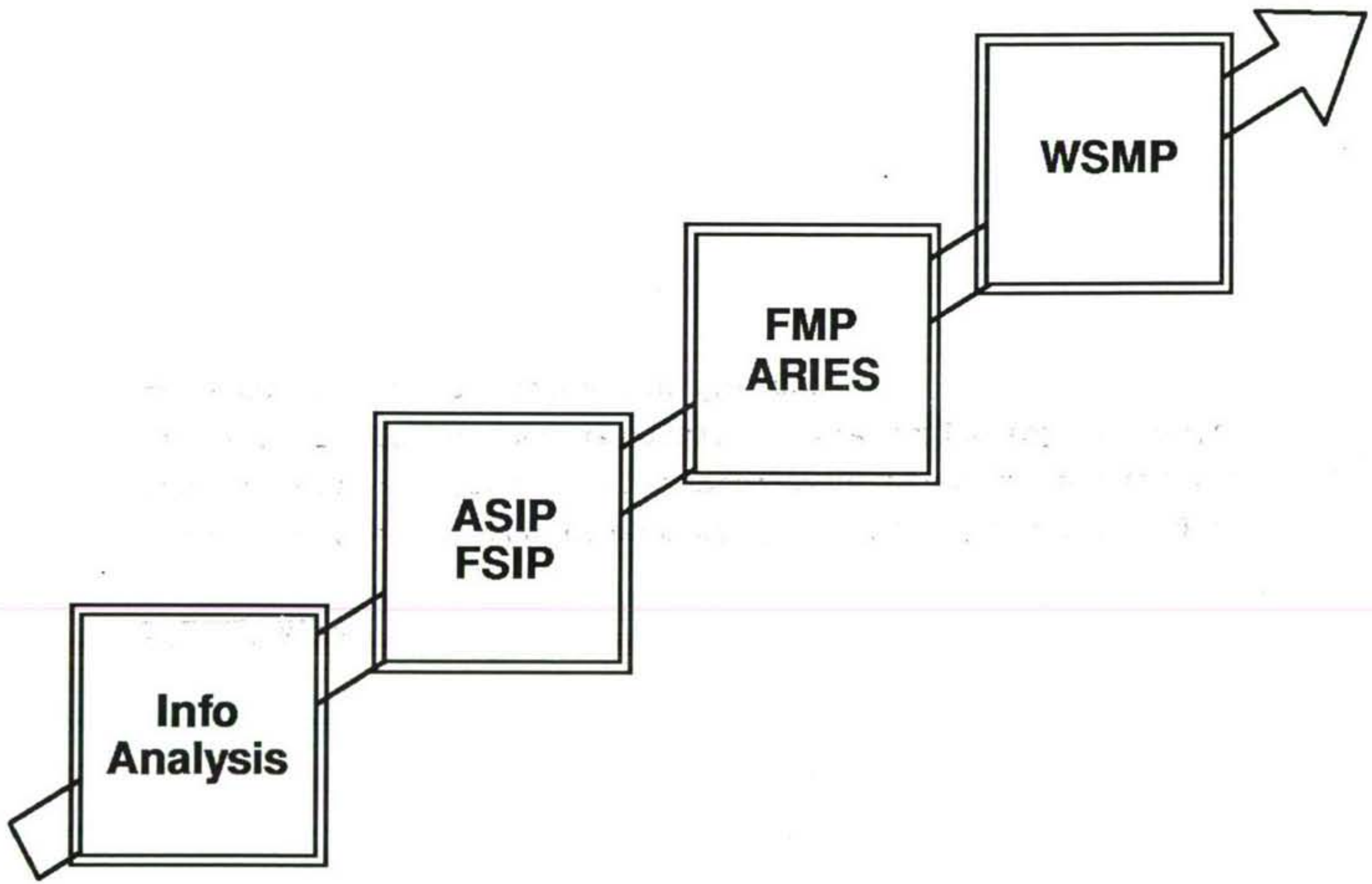
C-141 INTEGRITY PROCESS

This is an overall view of the integrity process for the C-141. It starts with the collection of information - test data, operational data, usage plans, logistics plans, and the like. Analysis of that information is directed through the Aircraft Structural Integrity Program and Functional Systems Integrity Program for recommendations and priortization. Lockheed then presents a Force Management Plan to Warner Robins which describes recommended improvements and schedules. This information becomes part of the Automated Readiness Integrated Engineering System, a database system of information the System Program Manager (SPM) and his staff use to develop the C-141 Weapon System Master Plan.

C-141 Integrity Process

C-141 FSIP



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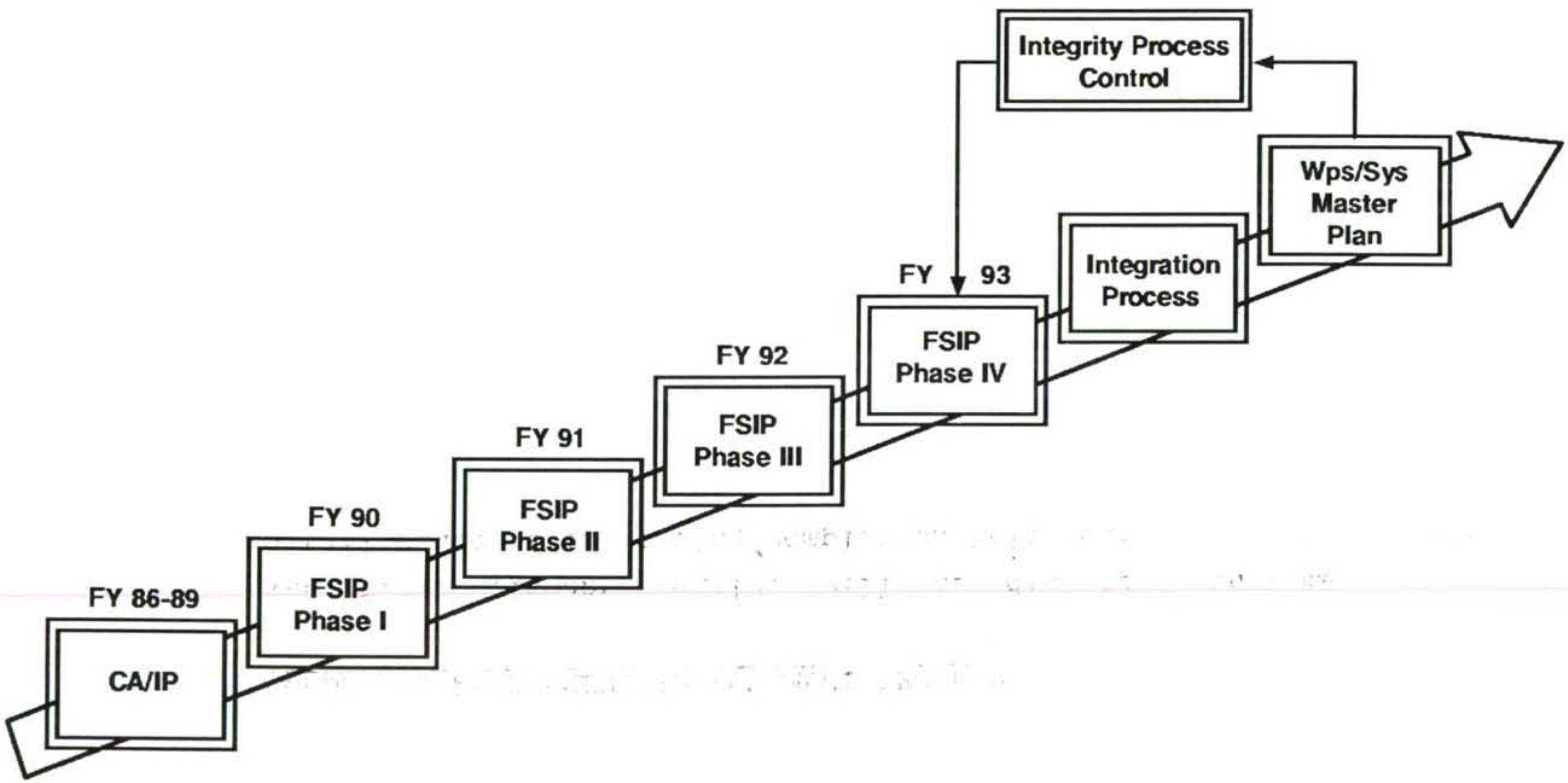
FSIP Phases

This chart shows the phases of FSIP from the beginning in the Condition Assessment/Improvement Program through the transition to a comprehensive FSIP comparable to ASIP. Each phase will be discussed further.

C-141 FSIP Phases

C-141 FSIP



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CONDITION ASSESSMENT/IMPROVEMENT PROGRAM

The first element, CA/IP, will be discussed in some detail since it is nearing completion this year.

Condition Assessment/ Improvement Program

CA/IP

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PURPOSE

The purpose of CAIP is to collect and analyze information, not available from other sources, to determine the condition of the C-141 fleet major functional systems.

Avionics and Engines are not included since both have specific programs dedicated to determine their conditions.

C-141 CA/IP

Purpose

To Obtain Information on the
Condition of the C-141 Fleet
With Special Reference to All
Major Functional Systems,
Excluding Avionics and Engines

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OBJECTIVES

The program objective is to determine the baseline condition of systems as they relate to safety of flight and mission success. With these specific conditions known, recommendations can be made that simplify maintenance tasks and provide for improved spares provisioning.

Objectives

C-141 CA/IP

- Determine Baseline Condition of Selected Systems
  - Safety of Flight/Mission Essential
- Use Data Collected To Help Simplify Maintenance Tasks
- Improve Spares Provisioning Support

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TASKS

CA/IP was initiated with a review of 66-1 data and visits to Air Force Bases to discuss systems conditions with operational and maintenance personnel. The information collected helped to identify specific problems and concerns for further study.

The aircraft inspection program began at the end of 1986 with one aircraft at the Lockheed facility. This aircraft was used to prototype inspection and data collection methods. As these methods were refined, six additional aircraft were inspected in 1988 and seven in 1989 to provide a large enough data sample to give confidence that the conditions found are representative of the fleet. The production program took place at Warner Robins.

Engineers evaluated the conditions of systems by performing functional checks and visual inspection in accordance with existing technical orders.

Components considered system critical were removed and sent to Technical Repair Center (TRC). Each component was bench tested using existing technical orders and then disassembled to allow visual inspection of its parts.

In addition to the specific information gathered through bench test and teardown, a component tracking system using maintenance and logistics data was developed.

Tasks

- Review Maintenance and Reliability Data
- Visit Three C-141 Air Force Bases
- Conduct the Program on 1 Aircraft at Lockheed (Prototype) Followed by 13 Aircraft at WR-ALC (Production)
- Accomplish Operational Checks and In-Depth Examinations of Functional Systems
- Conduct a Component Removal Program Followed by Bench Test and Teardown

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TASKS (CONT'D.

An automated data system was developed to collect & analyze CA/IP data. The inspection and teardown data was collected on portable PCs. Analysis software was developed at Lockheed to present the data in the form of graphs and reports. Once the system was fully operational it was transferred to Warner Robins for use on their VAX computer.

The conclusion and recommendations are being developed in two parts. First there are specific things that need to be improved that relate to tech data, maintenance, logistics, and design. With the completion of this years aircraft and component evaluations, improvement recommendations will be finalized and submitted to WR-ALC.

Second, recommendations are being made to the initiation of FSIP in its prototype phase. Many of the tools required for FSIP are available as a result of CA/IP and lessons learned during CA/IP will simplify tasks during the FSIP development process.

Tasks (Cont'd)

- Develop Automated Data System Compatible With Aircraft Information Retrieval System (AIRS) at WR-ALC
- Formulate Conclusions and Recommendations

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STATUS

The current status of CA/IP is that the data review has been completed; Altus, Norton, and McGuire AFBs were visited and a report has been issued detailing their problems as discussed with maintenance personnel. The prototype phase is complete and the report has been released. The CA/IP concept was developed during this prototype phase.

Status

C-141 CA/IP

- Reviewed AF 66-1 Maintenance Data
- Visited Air Force User Bases (1986)
  - Reported Released
- Phase 1 "Prototype"
  (1 A/C 1986 - 1987)
  - Report Released
  - Data Input to AIRS Computer
  - Developed CA/IP Concept

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STATUS (Cont'd.)

Phase 2 of CAIP started as a production program in 1988; during that year 6 aircraft were evaluated and a report was released for each aircraft. Several special reports were also written on the elevators, spoilers, and bleed air subsystems. An overall 1988 report has also been released.

Phase 3 production is in process with 7 A/C. All inspections have been completed and the interim reports have been released.

Status (Cont'd)

C-141 CA/IP

- Phase 2 "Production" Program
  (6 A/C Activity 1988)
  - Interim Reports Released
  - Special Reports Released
  - Final Report Released July 1989
- Phase 3 "Production" Program
  (7 A/C Activity 1989)
  - Interim Reports Released

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DIRECTION

Currently the FY'89 report is being compiled and a final report for the program will be included. When complete the condition of a sample 14 aircraft will have been determined.

Analysis needs continue to define areas where the analysis tasks can be simplified with changes to the automated data system.

The tracking/forecasting systems is in a development process. The final product will provide a means to simplify the task of forecasting where large amounts of quantitative and qualitative information is used.

C-141 CA/IP

Direction

- Continue FY'89 CA/IP Phase 3 Activity (7 A/C)
- Continue Development of Automated Data System
- Continue With Reliability & Maintainability Tracking/Forecasting System
  - Identify Current R&M Problem Areas
  - Identify Current Logistics Support Problem Areas
  - Forecast Future Problems

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DIRECTION (Cont'd.)

CAIP experience and results will guide the development of a Function Systems Integrity Program (FSIP) that is comparable with the Aircraft Structural Integrity Program (ASIP). With these programs the complete aircraft integrity can then be evaluated.

Direction (Cont'd)

- Plan for FY'90 CA/IP Related Activity As Well As Other Required Tasks for the Overall Functional System Integrity Program (FSIP)

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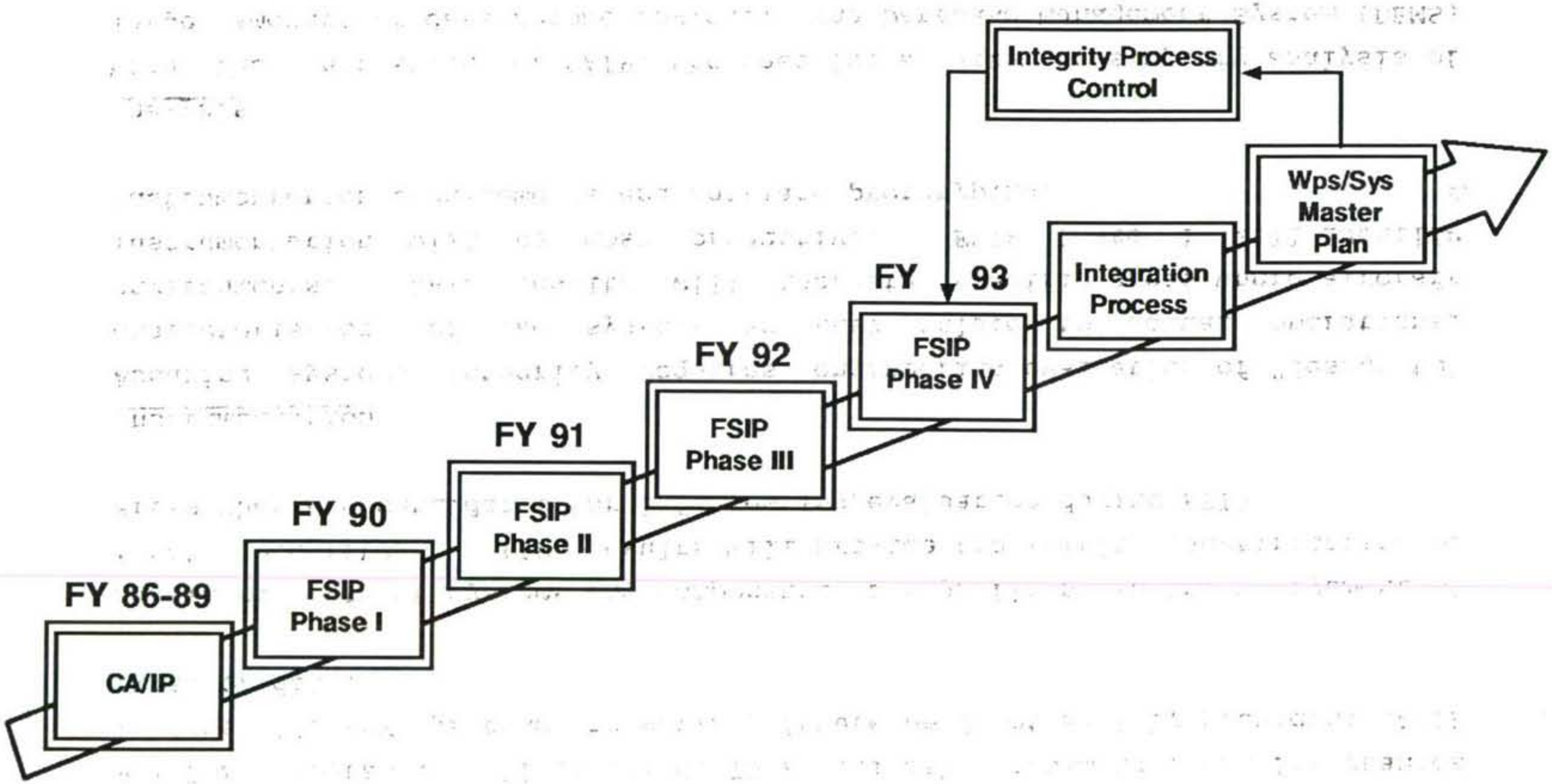
C-141 FSIP PHASES

This again is the overall FSIP; the phases following CA/IP will now be discussed.

C-141 FSIP Phases



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FSIP PHASE I

Tailored System Inspection Program (TSIP)

TSIP will continue aircraft evaluation as in CA/IP. Knowledge gained from CA/IP will be used to determine if additional data relating to systems condition is needed.

TSIP will also evaluate the compatibility of CAIP techniques with existing ACI/PDM programs. If required, TSIP will make recommendations for changes to the ACI/PDM program to satisfy future needs of FSIP by providing CA/IP types of data.

Remaining life of systems and components is significant in the management of fleet integrity. CAIP results will provide the initial identification of areas that are candidates for life testing/evaluation during FSIP.

Instrumentation

Assuring systems integrity requires engineering evaluation of "design to" characteristics of the systems as they relate to actual operational environments. CAIP results will identify specific areas where aircraft instrumentation will be most productive. FSIP Phase I will identify instrumentation requirements and initiate prototyping.

Analysis

With the initiation of CAIP, the need for a system to simplify analysis of large amounts of data become obvious. The Database Management System (DBMS) has been developed under CAIP and will be updated to satisfy needs for FSIP.

Coordination meetings with MAC are planned to assure that changes in operational use are considered as a part of the FSIP process.

The component tracking program developed during CA/IP will be expanded during FSIP development to include system level tracking and to provide for forecasting of both components and systems conditions.

FSIP Phases

FSIP Phase I
TSIP (Prototype) Aircraft Evaluation ACI/PDM Evaluation Life Testing/Evaluation **Instrumentation** Definition Prototype **Analysis** DBMS Update MAC Requirements Tracking/Forecasting Recommendations

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FSIP PHASE II

Tailored System Inspection Program (Validation)

Aircraft evaluation will continue during Phase II as guided by CAIP and FSIP Phase I. Data collected during Phase II will complete the CAIP data package.

Results of studies during Phase I will be used to validate ACI/PDM programs as the base for future FSIP information needs.

Selected critical LRU/NON-LRUs may be evaluated under accelerated life testing in a laboratory environment to validate life testing as a means to support the ability to predict hardware population remaining life.

Instrumentation

Prototyping will be completed in Phase II with the installation of a prototype recorder and the installation of transducers and sensors. Validation of data resulting from the prototype system will be completed during this phase.

Analysis

The potential of technology insertion as a means to solve previously identified hardware problems will be evaluated as a part of Phase II.

The tracking/forecasting system is evaluated as an operational system during Phase II.

FSIP Phases

FSIP Phase II
TSIP (Validation) Aircraft Evaluation ACI/PDM Evaluation Life Testing/Evaluation Instrumentation Prototype Validation Analysis DBMS Update MAC Requirements Technology Insertion Tracking/Forecasting Recommendations

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FSIP PHASE III

Phase III completes the FSIP Development process. During this phase, final adjustments are made to the program to assure its compatibility with the needs for integration of ASIP and FSIP into the integrity process. Also during Phase III, Lockheed provides support to WRALC in the transfer of responsibility for operation of the FSIP.

FSIP Phases

FSIP Phase III
TSIP (Transition) ACI/PDM Life Testing/Evaluation Instrumentation Production Data Collection Analysis DBMS Update MAC Requirements Technology Insertion Tracking/Forecasting Recommendations

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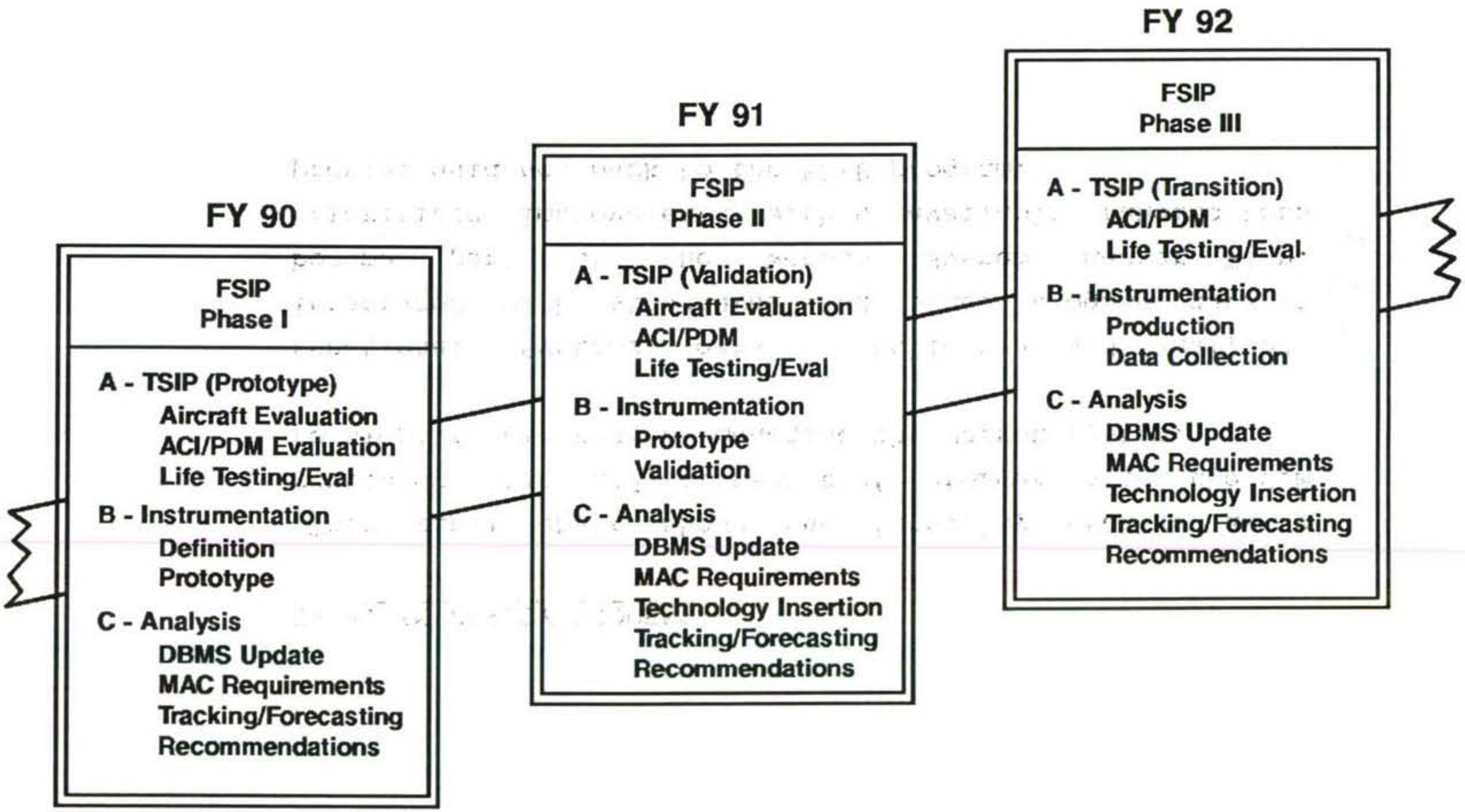
FSIP DEVELOPMENT PROCESS

The three phases just described show the FSIP development process which lead to the final program of integrity process control.

As discussed earlier, avionics and engines were not included in the CA/IP program. Information to support AVIP, ENSIP, and MECSIP will be evaluated during FSIP Phase I, and if needed, FSIP can be expanded to include these areas.

FSIP Development Process

C-141 FSIP



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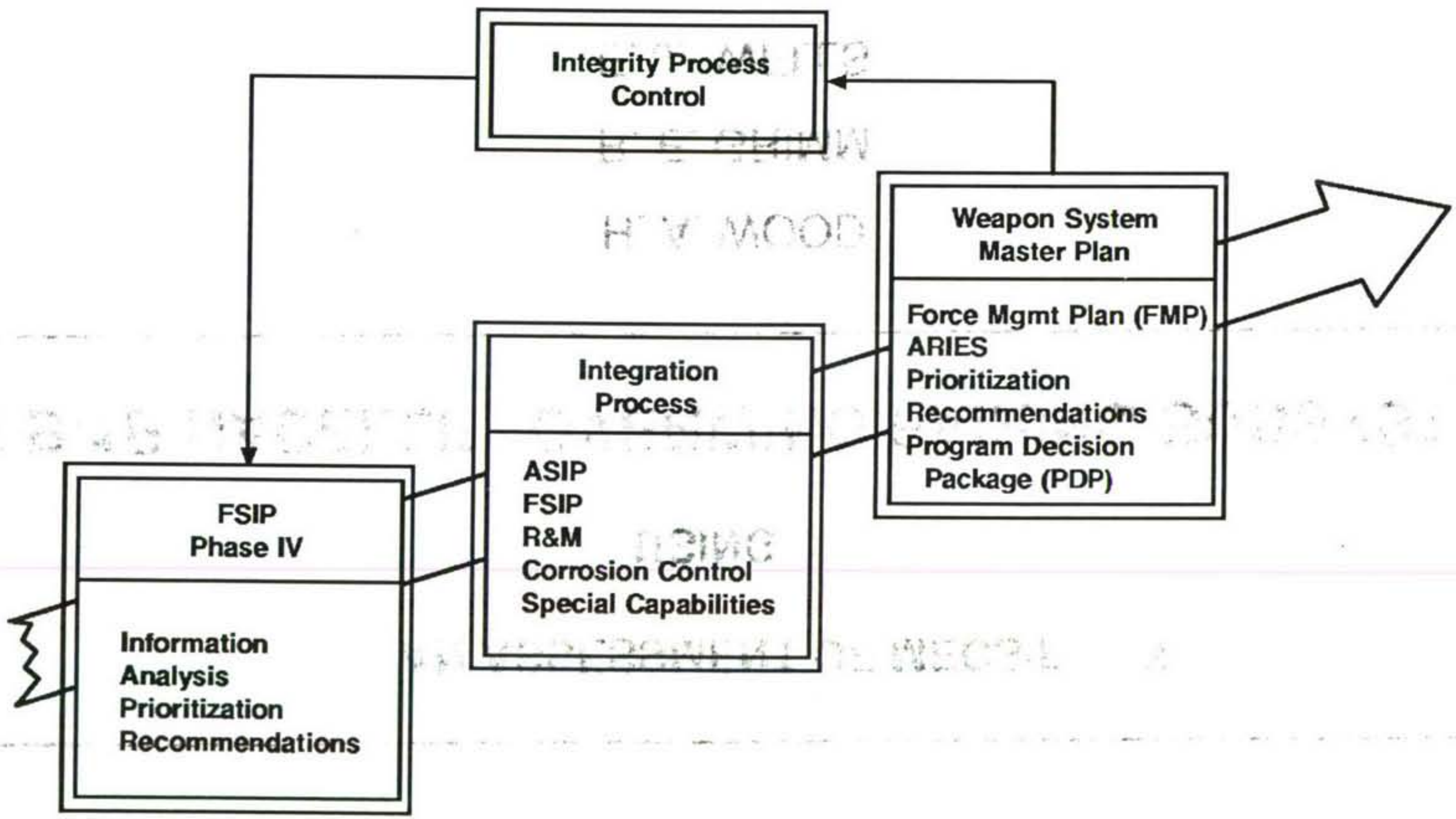
C-141 INTEGRITY PROCESS

This final chart shows the integrity process as a continuous flow and refinement of information for the SPM to simplify his task of managing the weapon system.

Functional Systems data is collected and analyzed, integrated with structural and other program data to become part of the Weapon System Master Plan. Prioritized improvements within available funding then provide guidance back to the FSIP program.

C-141 Integrity Process

C-141 FSIP



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11-15-89

AN ASSESSMENT OF MECSIP

USING

THE B-1B NACELLE / OVERWING FAIRING SUBSYSTEMS

H. A. WOOD

R. E. GRIMM

E. J. WELLS

AERONAUTICAL SYSTEMS DIVISION ASD/ENF

WPAFB, OHIO

PRESENTED AT THE USAF STRUCTURAL INTEGRITY CONFERENCE, SAN ANTONIO, TEXAS, DECEMBER 4-7, 1989

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OUTLINE

- STATUS REPORT OF MECSIP EVALUATION
- HIGHLIGHT MAJOR FINDINGS
- LESSONS LEARNED/RECOMMENDATIONS
- FUTURE ACTIVITIES

OBJECTIVES

- BETTER UNDERSTAND AND SCOPE THE IMPACT OF MECSIP COMPATABILITY WITH CURRENT PRACTICE

- ASSEMBLE DATA BASE TO USE IN FORMULATING GUIDANCE FOR IMPLEMENTING MECSIP IN FUTURE DEVELOPMENT PROGRAMS

- DEVELOP DATA FOR INTERNAL TRAINING AND COMMUNICATION

- RECOMMEND CHANGES TO CURRENT MECSIP DOCUMENTS

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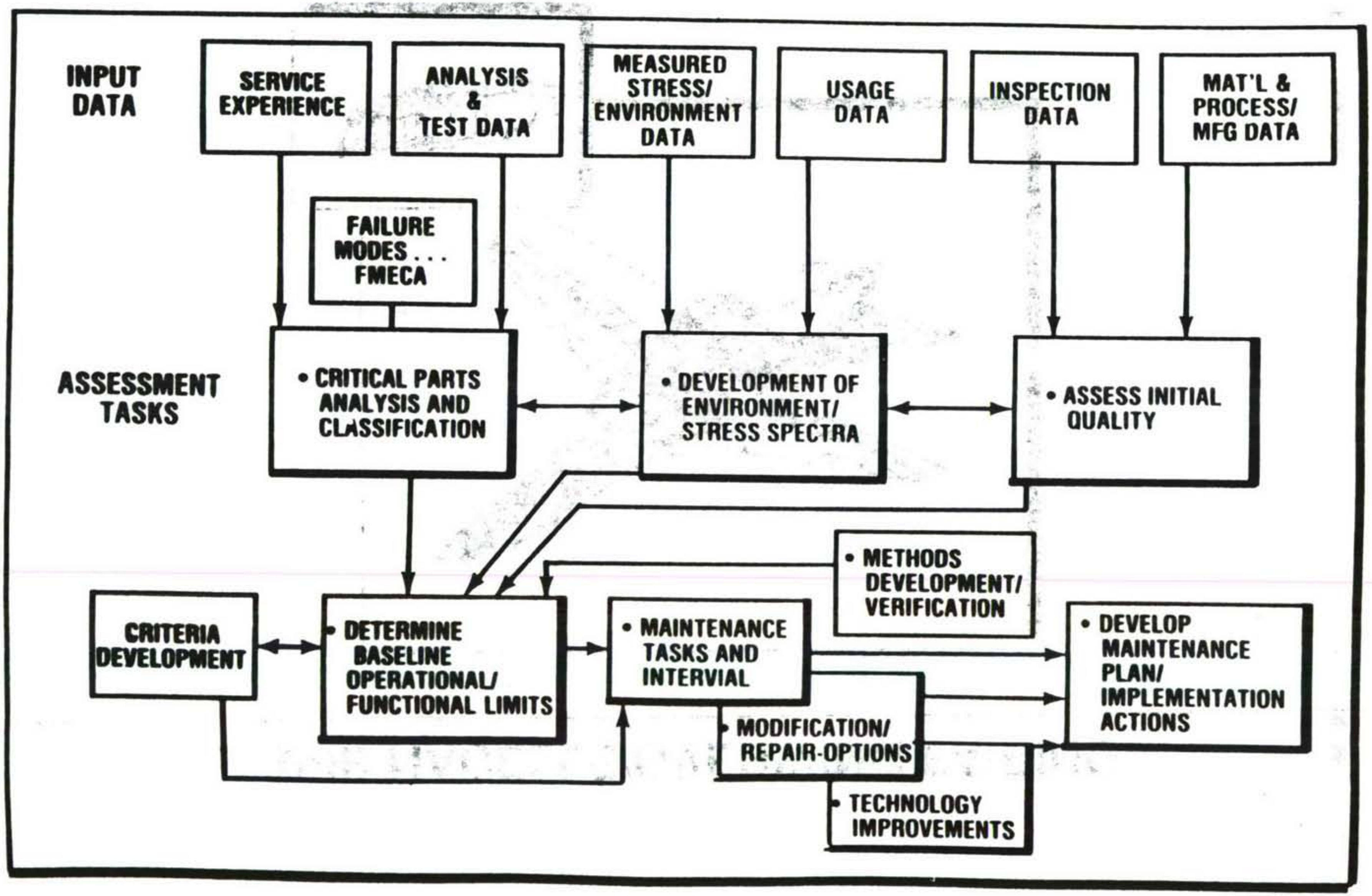
APPROACH

- IN HOUSE STUDY
- EVALUATE EXISTING, RECENTLY DEVELOPED HARDWARE
- CANDIDATE SYSTEM - B-1B
- ENGINE NACELLE/OVERWING FAIRING SUBSYSTEMS
- FOLLOW "DADTA" APPROACH
- ASSEMBLE DATA BASE WITH SPO AND CONTRACTOR ASSISTANCE

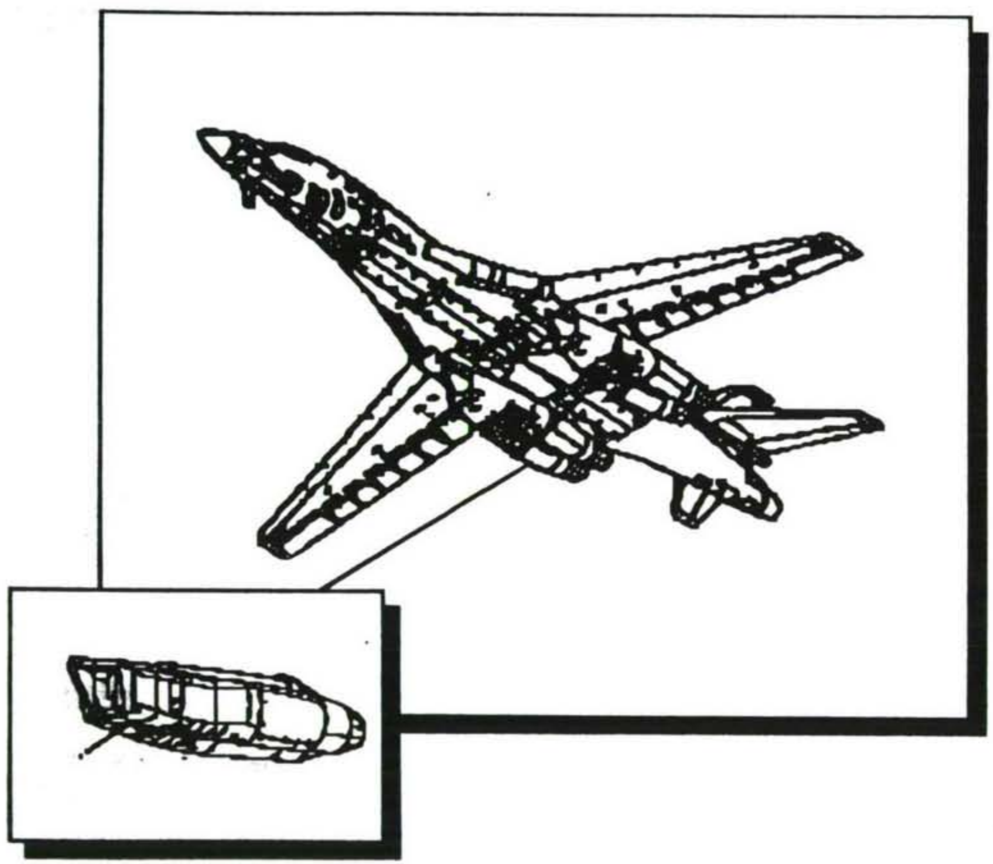
BACKGROUND ELEMENTS

- PRODUCT FAMILIARIZATION/FUNCTION/CRITICALITY
- MATERIALS/MANUFACTURING PROCESSES
- DESIGN/ANALYSIS/TESTING PHILOSOPHY AND APPROACH
- PROCUREMENT PRACTICES (SPECS,SOW, DATA, ETC.)
- VENDOR/PRIME/CUSTOMER ISSUES
- MAINTENANCE AND SERVICE EXPERIENCES

TECHNICAL APPROACH



B-IB NACELLE/OVERWING FAIRING



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DATA BASE

- SPECIFICATIONS
- DRAWINGS
- DESIGN ANALYSIS REPORTS
- QUALIFICATION REPORTS
- M&P SPECIFICATIONS
- STRUCTURAL ANALYSES
- FAILURE MODES ANALYSES (FMECA)
- RELIABILITY CENTERED MAINTENANCE (RCM) REPORTS
- VERBAL DISCUSSIONS - B-1B SPO, ROCKWELL AND VENDORS

FINDINGS - DATA BASE REVIEW

- PRODUCT DEVELOPMENT FOLLOWS "MIL SPEC"/INDUSTRY PRACTICES
- MOST COMPONENTS VENDOR SUPPLIED
- PRIME RESPONSIBLE FOR FIXED ITEMS, TUBING, DUCTS, ETC.
- SOME STRUCTURAL ANALYSES PERFORMED
- LOW OPERATING STRESSES
- LARGE DESIGN FACTORS FOR "STATIC" CONDITIONS
- LIMITED ENDURANCE TESTING
- SOME COMPONENTS QUALIFIED BY SIMILARITY
- SOME SPECIFIC M&P TASKS FOCUSED ON CRITICAL PARTS
- VENDORS DESIRE MORE OPERATIONAL USAGE DATA

FINDINGS - MAJOR CAUSES FOR LOSS OF FUNCTION

- STRUCTURAL FAILURE
- BEARING FAILURE
- IMPROPER INSTALLATION/MAINTENANCE
- WEAR
- CONTAMINATION
- LOSS OF LUBRICATION
- LOSS OF ELECTRICAL SIGNAL
- FAILURE OF CONTROLLER DEVICES

CANDIDATE ELEMENTS

SYSTEM

FUEL SYSTEM	HYDRAULIC SYSTEM	ELECTRICAL POWER	FIRE PROTECTION	SECONDARY POWER	ECS SYSTEM	OTHERS
• FUEL LINES • COUPLINGS • VALVES • FLOW METERS • BRACKETS • CONTROLS	• LINES • COUPLINGS • BRACKETS • PUMPS • RESERVOIRS • ACCUMULATORS	• GENERATOR • WIRES • CONNECTORS	• BOTTLES • DISTRIBUTION LINES • DETECTORS	• GEARBOX[ADG] • APU • PTO SHAFT • PNEUMATIC STARTER [ATS]	• LINES • HEAT EXCHANGER • COUPLINGS • BRACKETS • BLOWER	• INLET LIP ACTUATOR • OVERWING FAIRING ACTUATOR

FUNCTION

• ENGINE AND APU FEED SYSTEM	*• HYDRAULIC POWER SUPPLY* *• APU START* *• EMERGENCY BRAKING*	*• ELECTRICAL POWER GENERATION*	*• FIRE DETECTION AND SUPPRESS.*	*• ENGINE POWER TO HYDRAULIC, ELECTRICAL SYS.* *• APU POWER TO START ENGINES* *• STARTERS FOR MAIN ENGINE START*	*• ENGINE BLEED AIR DIST.* *• BLEED AIR PRE-COOL* *• BLEED AIR FOR ATS*	*• ACTUATE MOVABLE INLET LIP* *• OPEN/CLOSE OVERWING FAIRING*

SA10

COMPONENTS SELECTED

- FUEL LINES/COUPLINGS
- HYDRAULIC TUBING/COUPLINGS
- HYDRAULIC PUMPS
- STARTER ACCUMULATOR
- ACCESSORY DRIVE GEAR BOX (ADG)
- POWER TAKE OFF SHAFT (PTO)
- PNEUMATIC STARTER (ATS)
- ECS LINES/COUPLINGS
- ECS 'BABY PANTS' ASSEMBLY
- INLET LIP ACTUATOR
- FUEL FLOW METER

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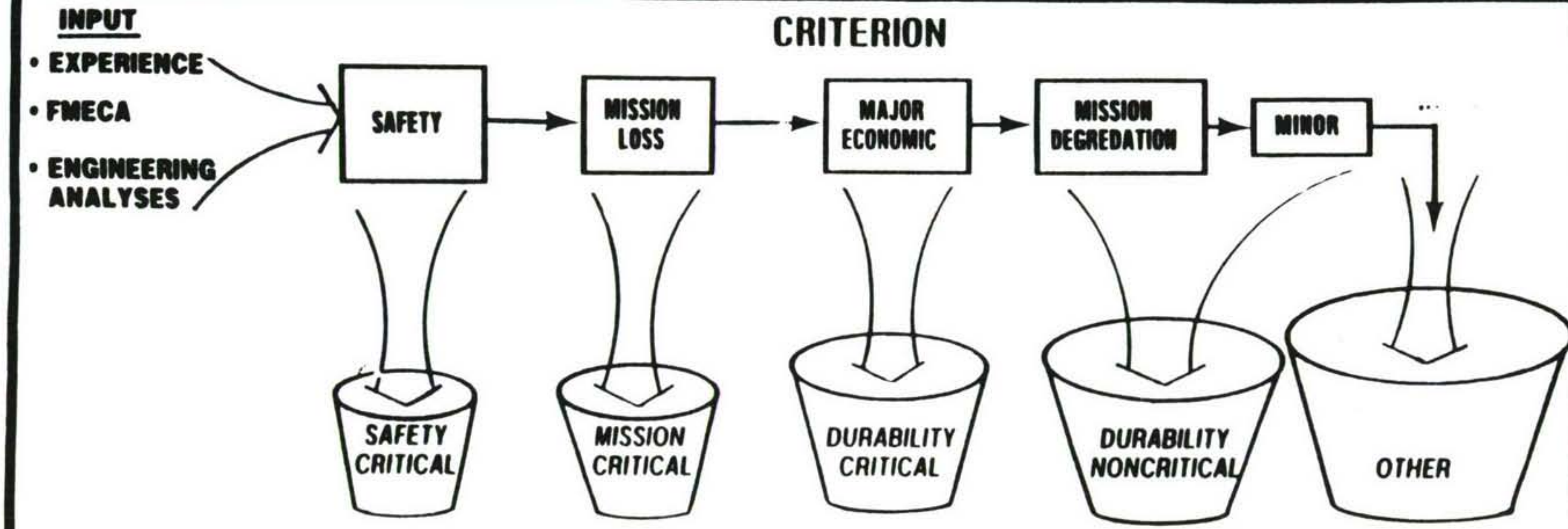
ASIP 7

COMPONENT CLASSIFICATION DECISION LOGIC



CLASSIFICATION CATEGORIES	CANDIDATE SPECIAL PROVISIONS*						
	EXPANDED DESIGN CRITERIA	TIGHTENED PROCESS CONTROL	ENHANCED QUALITY CONTROL	ACCEPTANCE TESTING	RESTRICTED REPROCUREMENT	PARTS TRACKING	PREVENTATIVE MAINTENANCE
SAFETY CRITICAL	●	●	●	●	●	●	●
MISSION CRITICAL	●	●	●	●	●	●	●
DURABILITY CRITICAL		●	●	●	●	●	●
DURABILITY NONCRITICAL				●			●
OTHER							

* THE EXTENT TO WHICH THE SPECIAL PROVISIONS ARE APPLIED IS COMPONENT SPECIFIC

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FMECA DEFINITIONS

• MISSION COMPLETION

A - NO EFFECT

B - SIGNIFICANT DEGRADATION IN PERFORMANCE

C - CANNOT COMPLETE

• FLYING QUALITIES

LEVEL 1 - CLEARLY ADEQUATE FOR MISSION FLIGHT PHASE

LEVEL 2 - ADEQUATE, BUT SOME INCREASE IN WORKLOAD, AND LOSS OF MISSION EFFECTIVNESS

LEVEL 3 - SAFE, BUT EXCESSIVE WORKLOAD, LOSS OF MISSION EFFECTIVNESS

LEVEL S - SPECIAL

• HAZARD

I - NEGLIGIBLE - NO EFFECT

II - MARGINAL - CAN BE CONTROLLED WITHOUT HAZZARD

III - CRITICAL - WILL CAUSE INJURY OR MAJOR SYSTEM DAMAGE, REQUIRE IMMEDIATE ACTION

IV - CATASTROPHIC - DEATH, INJURY, LOSS OF SYSTEM

B-1B MISSION PHASES

1. GROUND ALERT
2. ENGINE START
3. TAXI
4 .TAKE OFF
5. CLIMB
6. REFUELING
7. SUBSONIC DASH-HIGH ALTITUDE
8. SUBSONIC DASH-LOW ALTITUDE
9. WEAPON DELIVERY
10. DECENT
11. APPROACH
12. LANDING
13. GROUND OPERATION

INITIAL SCREENING CRITERIA

MECSIP CATEGORY / FMECA	MISSION COMPLETION	FLYING QUALITIES	HAZARD
• SAFETY CRITICAL	N/A	S*	IV, III*
• MISSION CRITICAL	C, B**	3	III
• DURABILITY CRITICAL	B	2	N/A
• DURABILITY NON CRITICAL	A	1	N /A
• OTHER	A	1	N/A

* POSSIBLE SAFETY IMPLICATION, REQUIRES REVIEW

** GROUND ALERT ASSIGNED B LEVEL IN FMECA

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MECSIP - CRITICAL ITEMS

SAFETY CRITICAL: FAILURE WOULD CAUSE LOSS OF THE AIR VEHICLE OR INJURY TO PERSONNEL OR EXTENSIVE DAMAGE TO CRITICAL EQUIPMENT OR STRUCTURE

MSSION CRITICAL: FAILURE WOULD (a) PROHIBIT THE EXECUTION OF A CRITICAL MISSION, (b) SIGNIFICANTLY REDUCE THE MISSION CAPABILITY, AND/OR (c) SIGNIFICANTLY INCREASE THE SYSTEM VULNERABILITY DURING A CRITICAL MISSION

DURABILITY CRITICAL: FAILURE MAY RESULT IN A MAJOR ECONOMIC IMPACT ON SYSTEM OR SUBSYSTEM PERFORMANCE BY REQUIRING COSTLY MAINTENANCE AND/OR REPAIR OR REPLACEMENT, WHICH IF NOT PERFORMED WOULD SIGNIFICANTLY DEGRADE PERFORMANCE AND OPERATIONAL READINESS

DURABILITY NONCRITICAL: FAILURE WOULD RESULT IN MINOR ECONOMIC IMPACT ON SYSTEM OR SUBSYSTEM PERFORMANCE BUT WOULD REQUIRE MAINTENANCE AND/OR REPAIR OR REPLACEMENT TO ASSURE CONTINUED PERFORMANCE

OTHER/EXPENDABLE: ALL OTHER COMPONENTS NOT CLASSIFIED IN ABOVE CATEGORIES

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EXAMPLE-COMPONENT CLASSIFICATION

FUEL FEED SYSTEM

FUNCTION FMECA (PART II)SCENARIO	MISSION	FLYING QUALITIES	HAZARD	COMPONENTS	CLASSIFICATION
• RESTRICTED/STOPPED FLOW TO ENGINE AND OR APU	B/C	-	-	• VALVES • LINES • COUPLINGS • CONTROLS	**MISSION** (REDUCED THRUST)
• FUEL LEAKAGE IN FLIGHT - SPILLED FUEL ON GROUND	C	3	$\boxed{III}$	• LINES • COUPLINGS • HOSES • VALVES	**SAFETY** (POTENTIAL FOR FIRE)

I

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FMECA - COMPONENT CLASSIFICATION

HYDRAULIC POWER SYSTEM -4 INDEPENDENT SYSTEMS*

	MISSION	FLYING QUALITIES	HAZARD
• ONE SYSTEM FAILURE	A	1	I
• TWO SYSTEM FAILURES	C	3	II
• ANY THREE FAILURES	C	S	III

• HYDRAULIC ACCUMULATORS [GENERAL]

- POTENTIAL HAZARD TO PERSONNEL
- POTENTIAL DAMAGE TO ADJACENT STRUCTURE/SYSTEMS

* NO SINGLE COMPONENT FAILURE WOULD RESULT IN SAFETY OR MISSION CRITICAL CLASSIFICATION

PARTS CLASSIFICATION

COMPONENT	SAFETY CRITICAL	MISSION CRITICAL	DURABILITY CRITICAL	DURABILITY NONCRITICAL
• PNEUMATIC STARTER[ATS]	(2)	X (ALERT)		
• PTO SHAFT/POWER TRANS. SHAFT	(2)	X (ALERT)		
• GEAR BOX[ADG]		X (ALERT)		
• ECS LINES			X (4)	
• BABY PANTS			X (4)	
• BLOWER			X (4)	
• INLET LIP ACTUATOR		X		
• FUEL LINES	X (LEAK)	X		
• FUEL FLOW METER	X (LEAK)	X		
• STARTER ACCUMULATOR	(2)	X		
• HYDRAULIC TUBING	(1)		X	X
• HYDRAULIC PUMP	(1)		X	X
• CLUTCH/SOLENOID[APU/ADG]	(3)	X		

[1] LOCATION DEPENDENT (FIRE RISK)

[2] HAZARD RISK TO ADJACENT SYSTEM

[3 HAZARD IF FAILS TO DISENGAGE

[4] FMECA NOT AVAILABLE

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FINDINGS - COMPONENT CLASSIFICATION PROCESS

- FMECA - USEFUL FOR ESTABLISHING COMPONENT CRITICALITY
- ENGINEERING JUDGEMENT REQUIRED FOR FINAL DETERMINITION
- DEFINITIONS REQUIRED FOR:
  - -FUNCTION CRITICALITY
  - -COMPONENT CRITICALITY RELATIVE TO FUNCTION
  - -MISSION PHASE(S) AFFECTED
  - -HAZARD POTENTIAL
- MECSIP DEFINITIONS - NO CHANGES RECOMMENDED
- MANY COMPONENTS - MISSION CRITICAL

FUEL LINE EXAMPLE

- FUEL LINE COMPONENTS - CLASSIFED AS SAFETY CRITICAL
- UNDER MECSIP COMPONENTS WOULD BE CANDIDATES FOR:
  - EXPANDED DESIGN CRITERIA, DAMAGE TOLERANCE
  - SERVICE LIFE/ DURABILITY CRITERIA
  - PRODUCT INTEGRITY CONTROL
    - EXPANDED M&P REQUIREMENTS AND CONTROLS
    - ENHANCED INSPECTIONS
    - TRACEABILITY
  - RESTRICTED REPROCUREMENT
  - ACCEPTANCE TESTING
  - IN SERVICE TRACKING
  - SCHEDULED PREVENTIVE MAINTENANCE

THE APPLICABILITY AND POTENTIAL IMPACT OF THESE REQUIREMENTS TO THE FUEL LINES WAS EXAMINED BY REVIEWING CURRENT PRACTICE

TYPICAL PROCESS - FUEL LINE ELEMENT

- TUBE
  - FORMING
  - HEAT TREAT/WELD
  - CHECK HARDNESS
  - DYE PENETRANT INSPECTION
  - SWAGE FERRULE TO TUBE *
  - X-RAY FERRULE SWAGE JOINT*
  - CHECK DIMENSIONS OF FERRULE INSTALLATION*
  - PROOF PRESSURE CHECK OF ASSEMBLY*
- COUPLING INSTALLATION
  - INSTALL INITIAL CLAMPS
  - INSTALL O-RING *
  - CHECK CLEARANCE BETWEEN MATING FERRULES *
  - ENGAGE NUT THREADS AND TIGHTEN TO PRESCRIBED THREAD EXPOSURE VALUES *
  - COMPLETE CLAMP INSTALLATION
- SYSTEM PROOF PRESSURE

* CRITICAL PROCESS STEPS FOR SEAL

22 12/1/89 5:57

FUEL LINES - M&P - SUMMARY

- ALUMINUM FUEL LINE PRACTICES TYPICAL OF OTHER METALLICS
- ALL TUBING COMPONENTS - 100% DYE PENETRANT INSPECTED
- FUEL SYSTEM IS PROOF PRESSURE CHECKED ON EACH AIRCRAFT
- INSTALLATION PRACTICES DEVELOPED TO PROVIDE CLEARANCE & MINIMIZE INSTALLATION STRESSES
- TUBE QUALITY RELATED TO
  - SURFACE IMPERFECTIONS
  - OVALITY
  - WALL THINNING ASSOCIATED WITH BENDING
  - DIMENSIONAL TOLERANCES
  - HEAT TREATING/WELDING/STRESS RELIEVING, ETC.
- OVALITY - DOMINANT FACTOR - 10% ALLOWED IN LOW PRESSURE LINES
- PROOF PRESSURE CHECKS FOR LEAKAGE - NOT EFFECTIVE IN SCREENING OUT METALLIC DEFECTS

FUEL LINE STRUCTURAL INTEGRITY

- AREAS INVESTIGATED
  - STRENGTH - PRESSURE LOADING, BENDING/TORSION
  - FLAW TOLERANCE/DURABILITY
    - CRACK PROPAGATION OF INITIAL SURFACE FLAWS
    - CRACK INITIATION
  - VIBRATION
    - SUPPORT SPACING
    - BENDING STRESSES
  - SENSITIVITIES
    - OVALITY
    - THINNING DUE TO BENDING
    - TOROIDAL EFFECTS

CURRENT PRACTICE

- **PRESSURES**
  - **NORMAL OPERATING PRESSURE (p) SPECIFIED**
  - **PROOF - 2 x p (NO MALFUNCTION, FAILURE, PERMANENT DAMAGE)**
  - **BURST - 3 x p (NO RUPTURE OR LEAKAGE)***
  - **SURGE - MAX. NOT TO EXCEED PROOF, OCCURRENCES OF SURGE PRESSURE TO BE MINIMIZED TO PREVENT FATIGUE PROBLEMS**

* BURST - 4 x p FOR COMPONENTS IN REFUEL, TRANSFER AND FEED SYSTEMS

FUEL LINE STRUCTURAL INTEGRITY

- FINDINGS
  - LOW NOMINAL OPERATING STRESSES
  - TUBE OVALITY - MOST PROMINENT " OFF NOMINAL FACTOR"
  - STRENGTH CHECKS FOR "WORST CASE"/OFF NOMINAL CONDITIONS SHOW POSITIVE MARGINS
  - INITIAL 0.010" DEEP SURFACE FLAW BELOW GROWTH THRESHOLD USING NOMINAL STRESSES AND MIN. WALL THICKNESS
  - CONVENTIONAL FATIGUE (S/N) ANALYSIS SHOWS ADEQUATE MARGIN USING NOMINAL STRESSES AND NOTCHED DATA, AND WORST CASE/"OFF NOMINAL" STRESSES AND UNNOTCHED DATA
  - TWO LIFETIME TOLERANCE FOR WORST CASE/OFF NOMINAL STRESSES ≈ 0.020 " DEEP SURFACE FLAW
  - LINES SHOULD BE LEAK BEFORE BREAK

APPLICABILITY OF "SAFETY CRITICAL" CRITERIA TO FUEL LINE EXAMPLE

(BASELINE - CURENT PRACTICE)

CANDIDATE REQMTS.	CURRENT PRACTICE ADEQUATE	ENHANCED PRACTICE REQUIRED	CRITERIA
• DAMAGE TOLERANCE CRITERIA		X	• INITIAL AND ACCIDENTAL FLAW TOLERANCE - LEAK BEFORE RUPTURE
• DURABILITY CRITERIA		X	• INITIAL QUALITY/ LIFE REQUIREMENT
• PRODUCT INTEGRITY CONTROLS			
-SPECIAL M&P REQUIREMENTS	X		• LOW STRESSES, ADEQUATE INTEGRITY WITHIN M&P BOUNDARIES
- SPECIAL INSPECTIONS	X		• CURRENT SCREENING ADEQUATE
- TRACEABILITY		X	• MAY BE DESIRABLE - ECONOMICS
• RESTRICTED REPROCUREMENT		X	• TO INITIAL SPECS AND PROCESSES
• IN SERVICE TRACKING	X		• VARIATIONS IN USAGE NOT SIGNIFICANT
• SCHEDULED PREVENTIVE MAINTENANCE	X		• NONE IN FORSEEABLE FUTURE
• ACCEPTANCE TESTS (ADDITIONAL)	X		• CURRENT ATPS ARE ADEQUATE

SUMMARY

- EXTENSIVE DATA BASE HAS BEEN ASSEMBLED THAT REPRESENTS "CURRENT" PRACTICE FOR MECHANICAL SYSTEMS
- HIGH DEGREE OF APPLICABILITY TO MECSIP
- MECSIP DOES NOT REQUIRE NEW " CULTURE" WITHIN THE INDUSTRY
- VENDORS NEED TO " COME UP TO SPEED "
- CURRENT ASSESSMENT RESULTS WILL HELP FOCUS INTEGRATED APPROACH

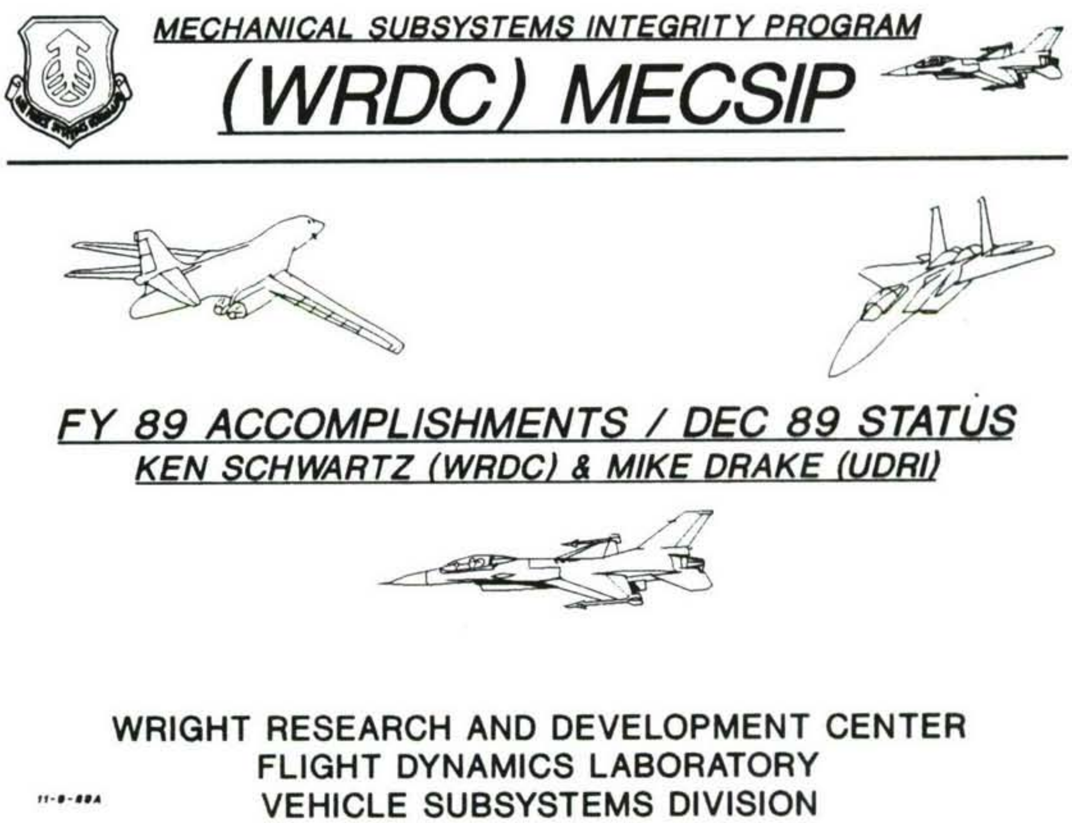
FUTURE EFFORT

- COMPLETE REVIEW AND ASSESSMENT OF DATA BASE
- CONTINUE IN -HOUSE ANALYSIS OF COMPONENTS
- USE FINDINGS IN TRAINING
- SPIN OFF R&D IN PROMISING AREAS

SLIDE #	NAME	DESCRIPTION
1	11-9-89A.CHT	SAN ANTONIO TITLE CHART

The purpose of this paper is to provide a brief summary of the more significant accomplishments and planning changes occuring since the last WRDC/MECSIP presentation 1 year ago. The paper will be presented in two parts by representatives from WRDC and the University of Dayton Research Institute (UDRI).

DISPLAY TIME:



SLIDE #	NAME	DESCRIPTION
2	7-20-89E.CHT	WRDC MECSIP ORGANIZATIONAL RELATIONSHIP

This chart depicts how the WRDC MECSIP effort integrates with the ASD MECSIP process and the Air Force R&D Aquisition/ Logistics structure. This integration will provide an assessment of Aquisition/Logistics needs followed by a direction of R&D programs.

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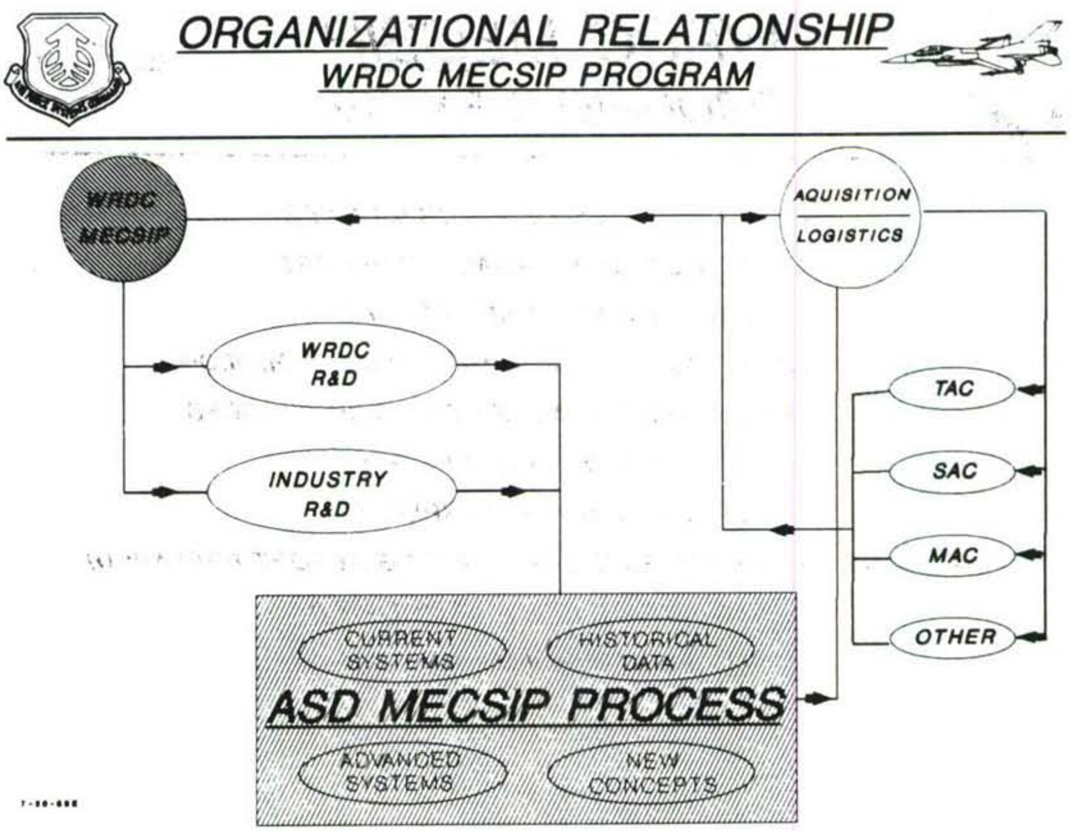

SLIDE #	NAME	DESCRIPTION
3	11-9-89B.CHT	SANANTONIO PITCH FY-89 PROGRESS

This slide provides a summary of WRDC MECSIP accomplishments made over the last 12-15 month time period.

DISPLAY TIME:

WRDC MECSIP
PROGRESS FY-89

CLEAR DEFINITION OF 6.2 OBJECTIVES

FINALIZATION OF 6.2 ANALYSIS EFFORT

PLANNING OF 6.3 ADP EFFORT

BASELINE DEFINITION OF WRDC MECSIP METHODOLOGY

UNDERSTANDING/ACCEPTANCE OF MECSIP PROCESS

F-15 ANALYSIS COMPLETED

F-16 ANALYSIS 75% COMPLETED

COMPONENT TECH AREA SUPPORT (BRIEFING REPORT PREPARED)

CONTRACTUAL EFFORTS INITIATED

AFIT ALS APPROACH ANALYSIS

11-9-89B

SLIDE #	NAME	DESCRIPTION
4	9-15-89A.CHT	6.2 MECSIP OBJECTIVES

One of the primary concerns of the WRDC MECSIP effort is the wide range of potential program approaches and/or objectives facing the effort. Over the past year however three principal objectives were identified which scopes the WRDC MECSIP effort for the next several years. The first and most important is establishment of an R&D process or methodology to be utilized. This will then be followed by the identification of critical components for process application and the identification of technology areas or gaps for future technology advancements.

DISPLAY TIME:

WRDC MECSIP
6.2 PROGRAM OBJECTIVES

ANALYZE/ESTABLISH R&D MECSIP METHODOLOGY

FORMULATE
REFINE
SPECIFY

ANALYZE/ESTABLISH R&D MECSIP COMPONENTS

DEFINE
CATAGORIZE
MERIT/RATE

IDENTIFY BASIC R&D TECHNOLOGY AREAS

ESTABLISH
MERIT/RATE

9-15-89A

SLIDE #	NAME	DESCRIPTION
5	7-24-89H.CHT	CURRENT AIRCRAFT STUDIES (F-15/F-16)

Two aircraft studies were also completed over the past year the details of which are noted on this chart.

DISPLAY TIME:

CURRENT AIRCRAFT STUDIES

F-15 MECSIP STUDY

CONTRACT

DATA SOURCES
- AFR 66-1 DATA BASE
  - MacAir SOFTWARE
  - MacAir FIELD REPORTING

SUBSYSTEMS
- LANDING GEAR
- FUEL SYSTEMS
- HYDRAULICS

OBJECTIVES
- IDENTIFY FAILURES
- IDENTIFY TECH GAPS
- DEVELOP MECSIP PLAN

F-16 MECSIP STUDY

IN-HOUSE

DATA SOURCES
- AFR 66-1DATA BASE
  - MODAS
  - CDS
  - GENERAL DYNAMICS
  - FIELD SURVEYS

SUBSYSTEMS
- ALL

OBJECTIVES
- IDENTIFY FAILURES
- IDENTIFY TECH GAPS
- COMPARE (F-15/F-16)

7-24-89H

SLIDE #	NAME	DESCRIPTION
6	9-20-89A.CHT	F-15 FINDINGS 1

With reguard to the F-15 analysis some of the more significant findings are noted here and on the following chart.

DISPLAY TIME:

WRDC MECSIP

F-15 FINDINGS (WRDC RELEVANT)

- *D056 DATA NON APPLICABLE TO COMPONENT PARTS*
- *D056 DATA POTENTIALLY MISLEADING IN SEVERAL AREAS*
  - *TIRE DATA BIGGEST OFFENDER*
- *D056 MAINTENANCE DATA REPORTING*
  - *OPEN TO QUESTION*
  - *IMPROVEMENTS NEEDED*
  - *ALS (A/C LOGISTICS SYSTEM) PROPOSED*
    - *PC TOUCH SCREEN*
    - *DOWN STEP MENU DRIVEN*
    - *SPECIALIZED PARTS BAR CODED*
- *METHOD TO ASSESS INTEGRITY DEVELOPED*
  - *MIM (MECSIP INTEGRITY MODEL)*
    - *DOLLAR BASED*
    - *WHAT IF TYPE ANALYSIS*

9-20-89A

SLIDE #	NAME	DESCRIPTION
7	9-20-89B.CHT	F-15 FINDINGS 2

F-15 findings continued.

DISPLAY TIME:

WRDC MECSIP

F-15 FINDINGS (WRDC RELEVANT)

- *CLEAR (CLOSED LOOP FAILURE REPORTING)*
  EXPAND BEYOND FLIGHT TEST
- *AFISC SAFETY DATA CONCLUSIONS*
  LIMITED ACCESSIBILITY
  DATA IS BRIEF & NON WUC RELATED
  DATA OFTEN NOT RELEVANT TO SPECIFIC PART
  ONLY SEVEREST CASES INCLUDED
  NO HANDS ON QUERY
  1-2 WEEKS DELAY
- *FUTURE METHODOLOGY (FAILURE ANALYSIS)*
  AUTOMATE MIM INPUT SECTION
- *FUTURE METHODOLOGY (MECSIP/DESIGN)*
  RESEARCH GAPS GENERAL IN NATURE
  MORE TECHNOLOGY THAN METHODOLOGY RELATED
  PRODUCT IMPROVEMENT SYNDROME

9-20-89B

SLIDE #	NAME	DESCRIPTION
8	9-20-89C.CHT	F-15/F-16 IMPROVEMENT AREA CONCLUSIONS

One of the objectives of both the F-15 & F-16 studies was to try and identify technology or improvement areas where deficiencies were uncovered during the analysis effort. This chart summarizes the results of that identification process.

DISPLAY TIME:

F-15/F-16 STUDIES

IMPROVEMENT AREA CONCLUSIONS

F-15 IMPROVEMENT AREAS

LANDING GEAR
- L/R MAIN LDG GEAR
- TIRES/WHEELS/BRAKES

FUEL SYSTEM
- WING FUEL TANKS
- GAGING
- FUEL FLOW INDICATING
- ENGINE FUEL SUPPLY

HYDRAULICS
- UTILITY SYS
- STABALATOR CONTROL
- UTILITY MANIFOLD
- CANOPY ACTUATOR

F-16 IMPROVEMENT AREA

LANDING GEAR
- TIRES/WHEELS/BRAKES
- ACTUATORS (CORROSION)

FUEL SYSTEM
- WING FUEL TANKS
- GAGING
- FUEL FLOW INDICATING
- ENGINE FUEL SUPPLY

CREW STATIONS
- CANOPY ASSEMBLY

WEAPONS DELIVERY
- WINGTIP LAUNCHER

FLIGHT CONTROLS
- FLIGHT CONTROL COMPUTER
- ELEC/MECH ACTUATOR
- POWER DRIVE UNIT
- ELECTRONIC COMPONENT ASSY.

9-20-89C

SLIDE #	NAME	DESCRIPTION
9	8-03-89B.CHT	MECSIP PLANNING STATUS 6.2/6.3 ONLY

The next two charts are included to illustrate the progress made over the past year within the WRDC MECSIP effort. This first chart shows the status of the WRDC MECSIP program as of last september 1988.

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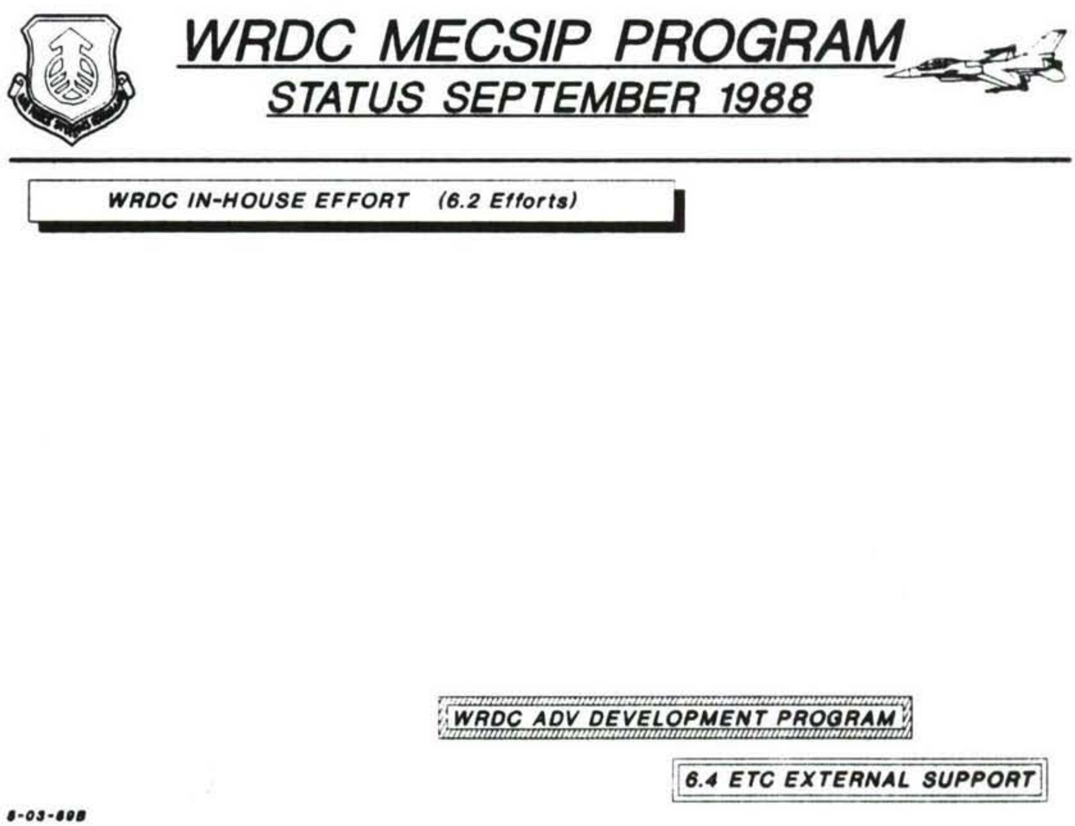

SLIDE #	NAME	DESCRIPTION
10	11-9-89C.CHT	SAN ANTONIO MECSIP PLANNING TODAYS STATU

This chart illustrates the status of the WRDC MECSIP effort one year later. More specifically over 14 separate in-house or contractual efforts have been added, approved, or are underway in this area. Time does not permit a discussion on each and every one of these programs but I have covered the critical aspects on the more significant findings to date and Mr. Michael Drake of UDRI will cover some interesting planned expansions into new data bases and some advanced methods planned for WRDC MECSIP technology assessment and component definition.

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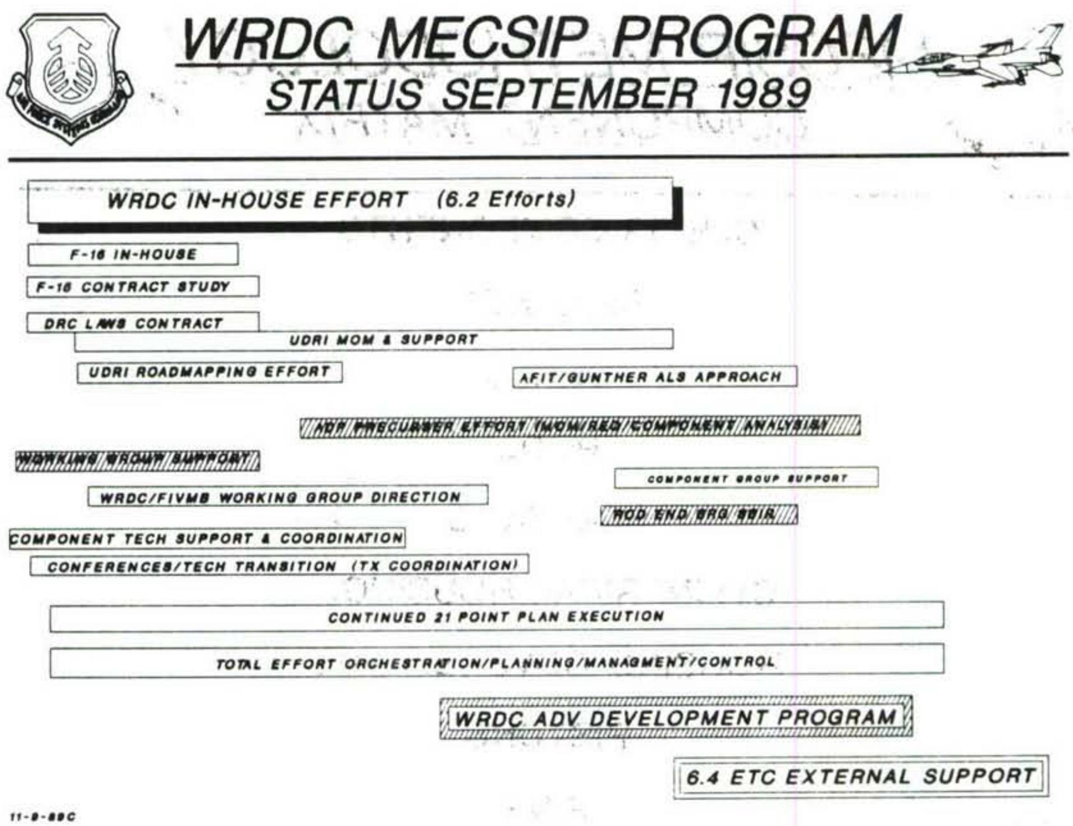

SLIDE #	NAME	DESCRIPTION
11	5-12-89B.CHT	MECSIP METHODOLOGY (component matrix)

Over the past year WRDC/FIVMB concluded that attacking subsystem integrity at the basic component level was the most cost effective approach to the problem. Through this approach we can achieve the broadest application of any resulting WRDC MECSIP methodology. This slide represents a partial listing of typical components for future consideration. It is components such as these that will be referred to as components or critical components in the remainder of this discussion.

DISPLAY TIME:

MECSIP METHODOLOGY

COMPONENT MATRIX

CLEVIS COMPONENTS

BEARINGS

SEALS

RODS

ROD ENDS

CYLINDRICAL HOUSINGS

HOUSING COVERS/CAPS

PISTONS

PINS

5-12-89B

SLIDE #	NAME	DESCRIPTION
12	9-22-89C.CHT	ACTUATOR COMPONENTS

Taking typical components noted in previous component lists we see that an aircraft subsystem methodology model could be assembled if a family of component methodology models were available.

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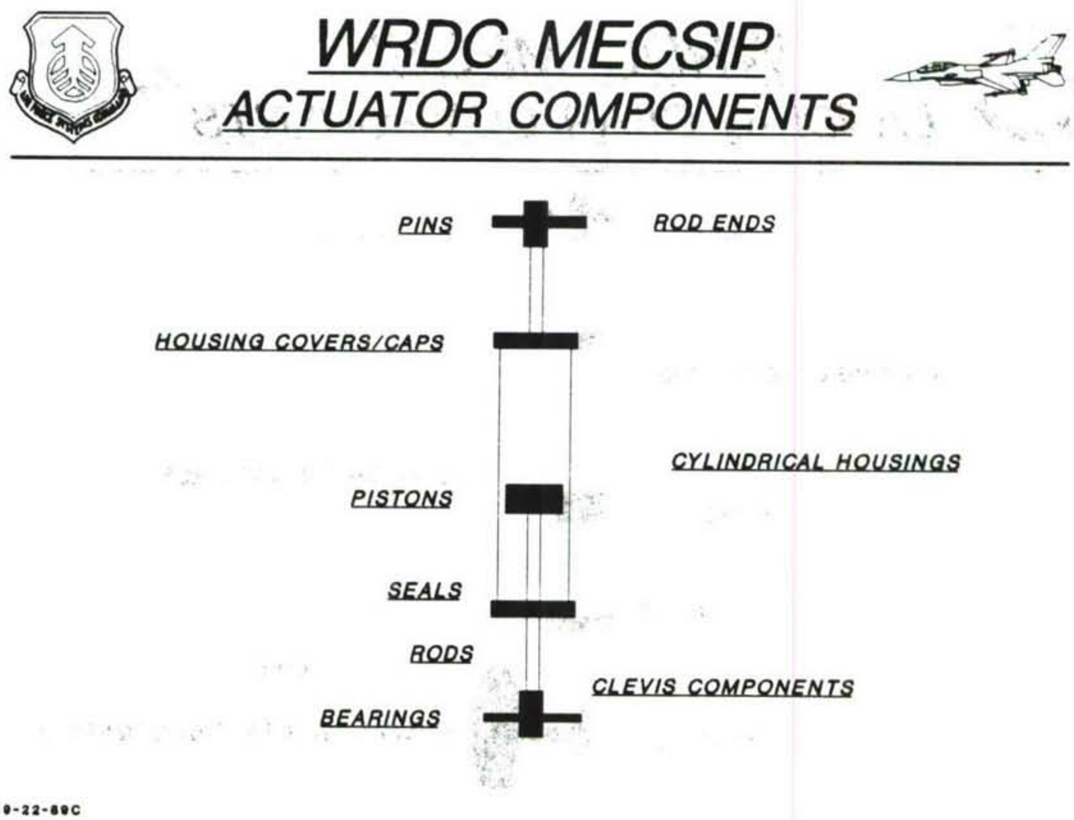
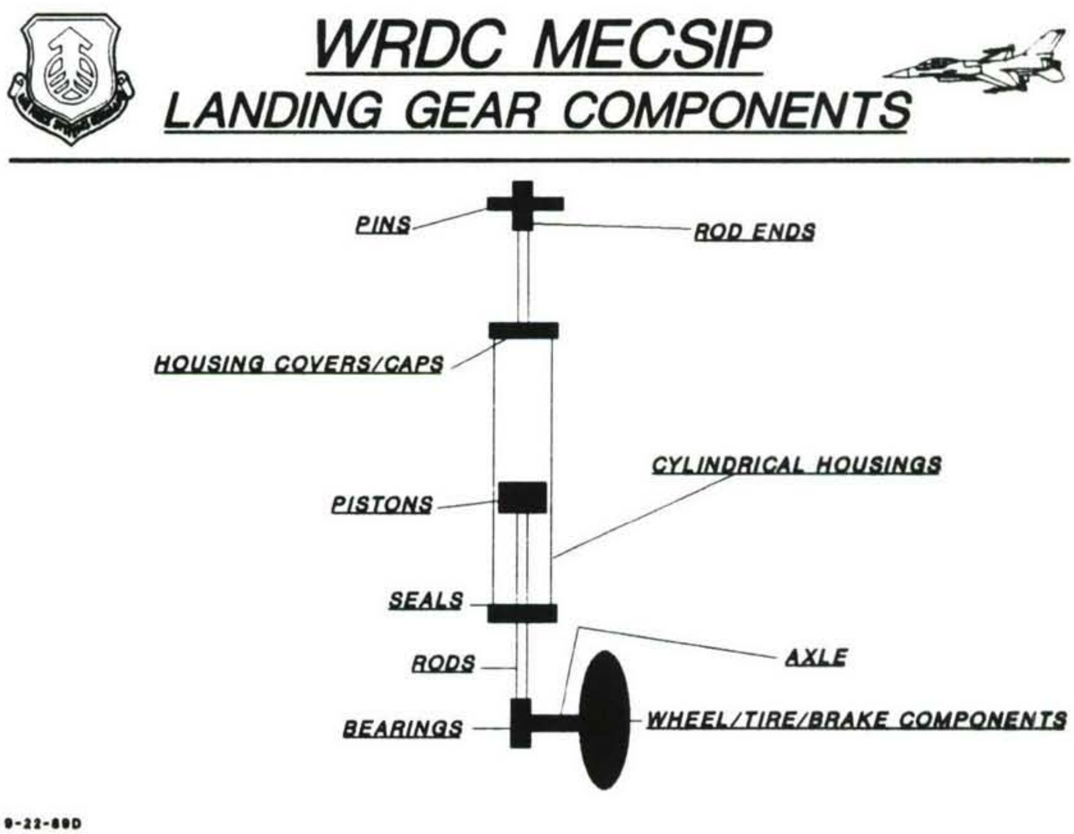


SLIDE #	NAME	DESCRIPTION
13	9-22-89D.CHT	LANDING GEAR COMPONENTS

Additionally if we replace only a select few of the components within the subsystem model we can easily create entirely new equally applicable subsystem methodology models. The example shown here replaces the lower set of clevis components with wheel tire and brake component methodology models to create a landing gear methodology model.

DISPLAY TIME:



SLIDE #	NAME	DESCRIPTION
14	MEC_1 .CHT	

The approach utilized in the MECSIP effort is to analyze R&M data from a component perspective. From this analysis critical technology gaps will be defined along with potential solutions. The end result will be an increase in supportability of mechanical systems.

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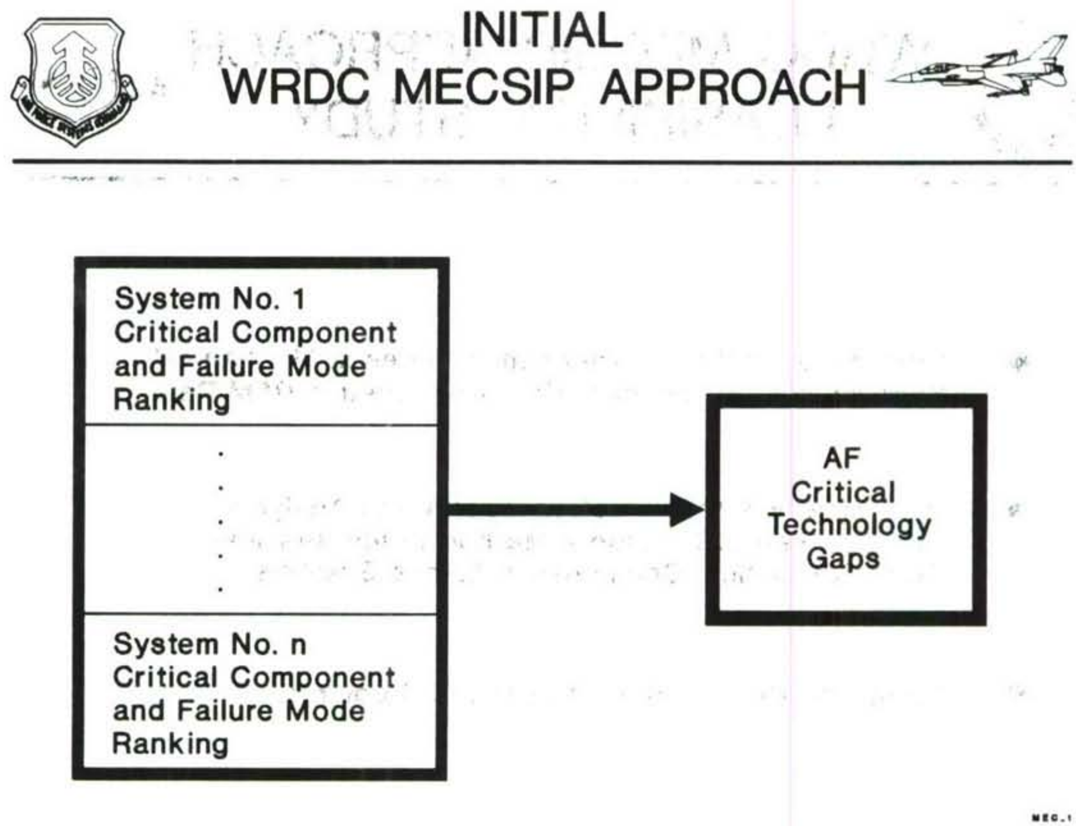

SLIDE #	NAME	DESCRIPTION
15	MEC_5 .CHT	

A contractual effort was undertaken by the team of DRC & UDRI to assess the feasibility of the WRDC MECSIP approach. The objectives of this effort are noted in this chart.

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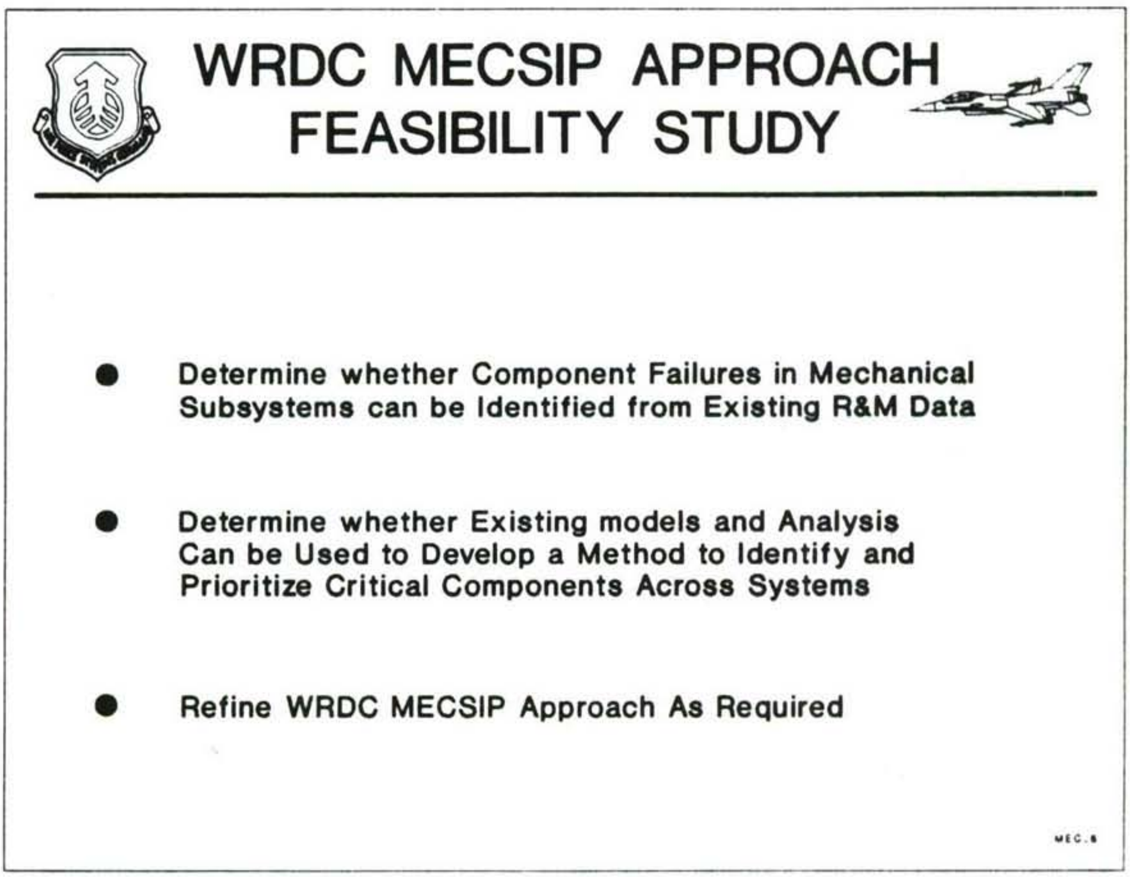

SLIDE #	NAME	DESCRIPTION
16	MEC_6 .CHT	

The fundamental part of the WRDC MECSIP approach is the determination of those components most critical to the goals of the MECSIP process. The basic approach to identifying these components (CCI process) will be to first identify primary systems & critical subsystems. These will then be further broken down to a sub-subsystem level followed by a component breakout and analysis.

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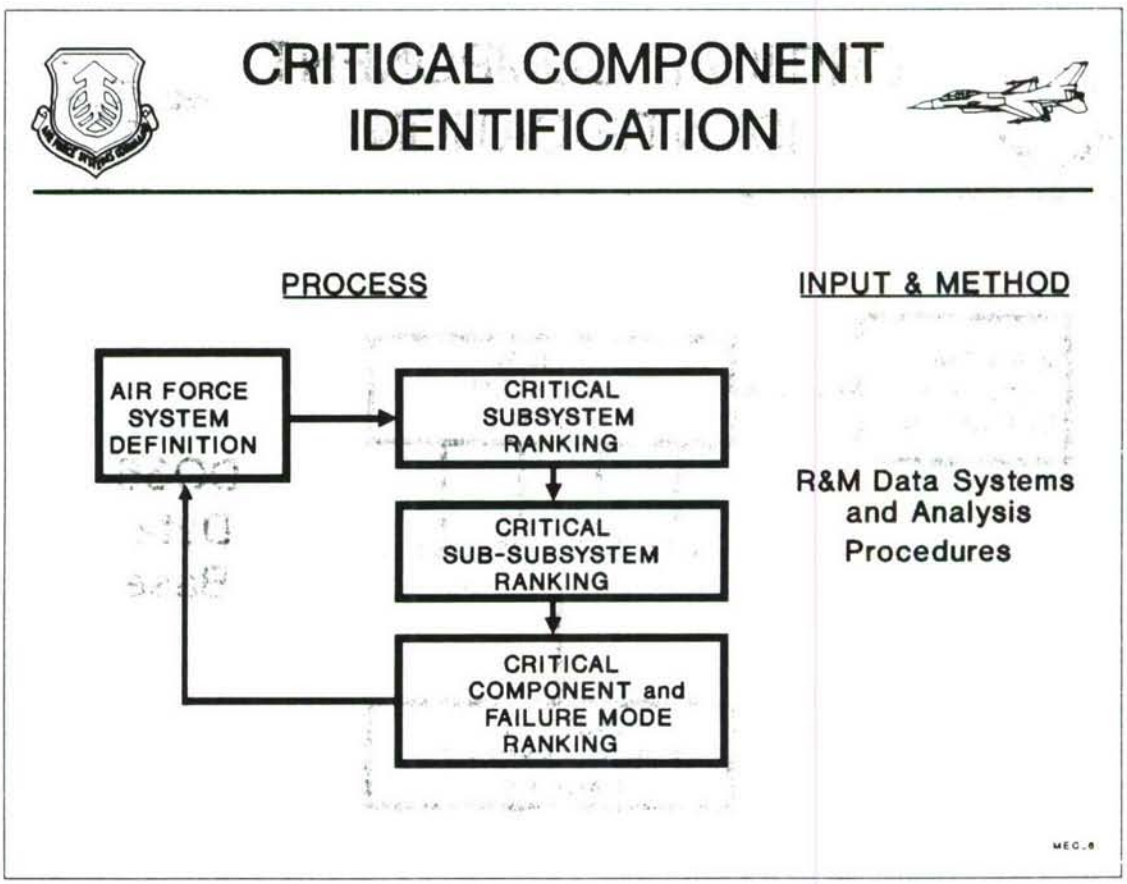
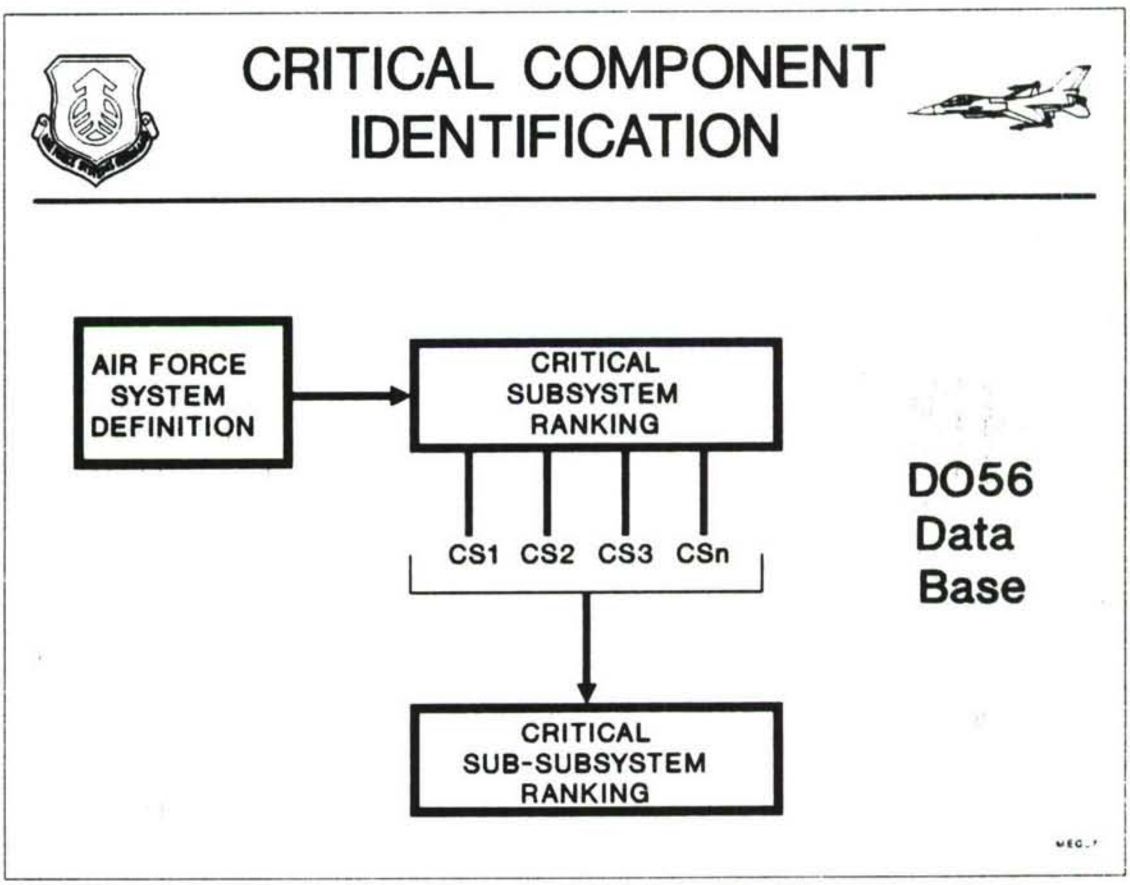


SLIDE #	NAME	DESCRIPTION
17	MEC_7 .CHT	

A breakdown and detailed evaluation of this process is shown here. Current R&M data (DO56) and analysis systems allow the ranking of critical subsystems such as landing gear and flight control subsystems. Current data (DO56) also allows the further division of each subsystem into critical sub-subsystems.

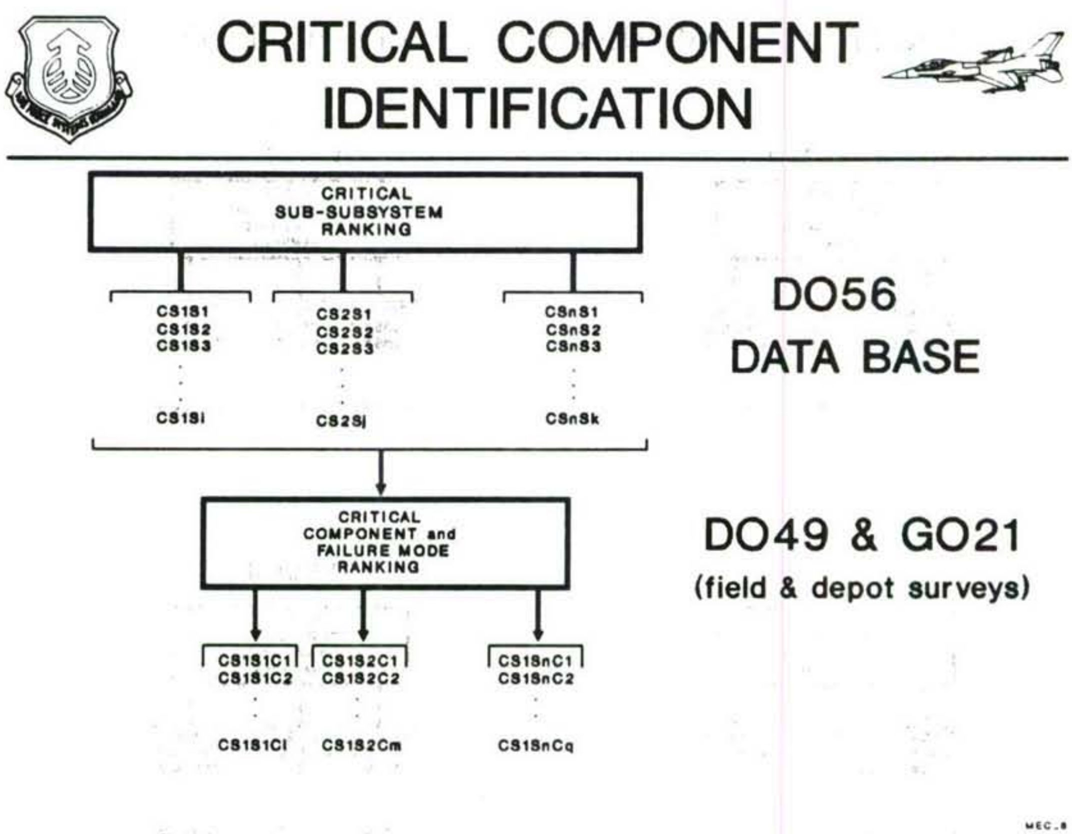
DISPLAY TIME:



SLIDE #	NAME	DESCRIPTION
18	MEC_8 .CHT	

This division to the critical sub-subsystem level takes us from the subsystem level (i.e. landing gear, flight control) to a more specific sub-subsystem item such as an oleo strut or control actuator. From these specific sub-sub areas then we will further divide into the basic component level. The (DO49) and the (GO21) data complimented with field and depot surveys should provide the required data for this division.

DISPLAY TIME:

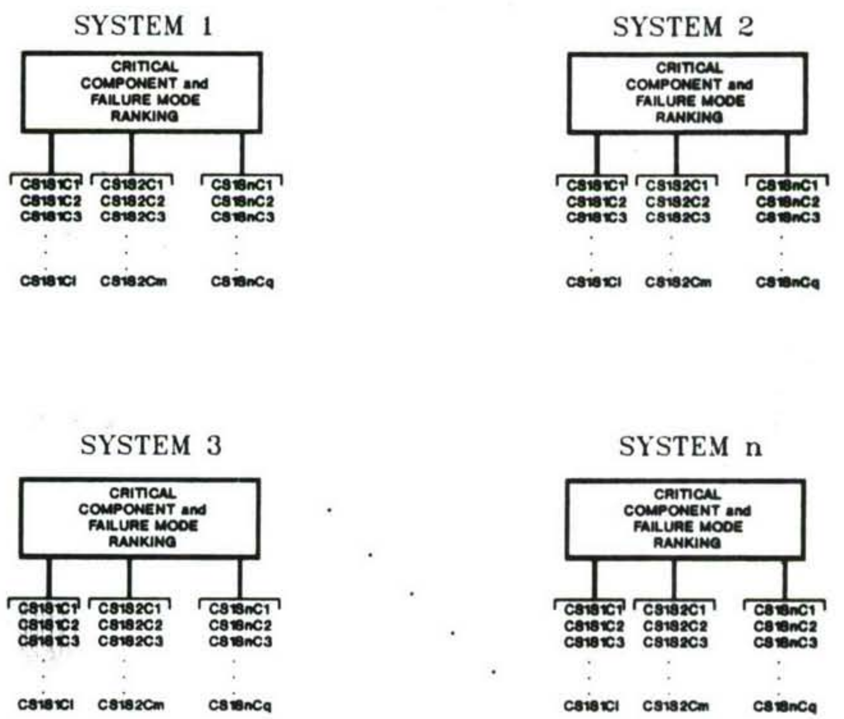


SLIDE #	NAME	DESCRIPTION
19	MEC_11 .CHT	

Once we can analyze down to the component level, we will have a large collection of critical component data for each A.F. system. Traditional engineering approaches would be to initiate programs to solve the problems on the most critical components for a specific system. The WRDC MECSIP approach however will be to use this mass of data to define critical component technology gaps which, if filled, would have the most impact on all A.F. systems. To do this, a method to integrate all this data must be determined. The key here is to define a generic components and failure mode definitions.

DISPLAY TIME:

CRITICAL COMPONENT IDENTIFICATION



SLIDE #	NAME	DESCRIPTION
20	MEC_9 .CHT	

The missing link in being able to generate the critical technology gaps is the integration using generic components and failure definitions. Using this data-collapser shown in this chart we can now look across all the A.F. systems and define critical technology gaps.

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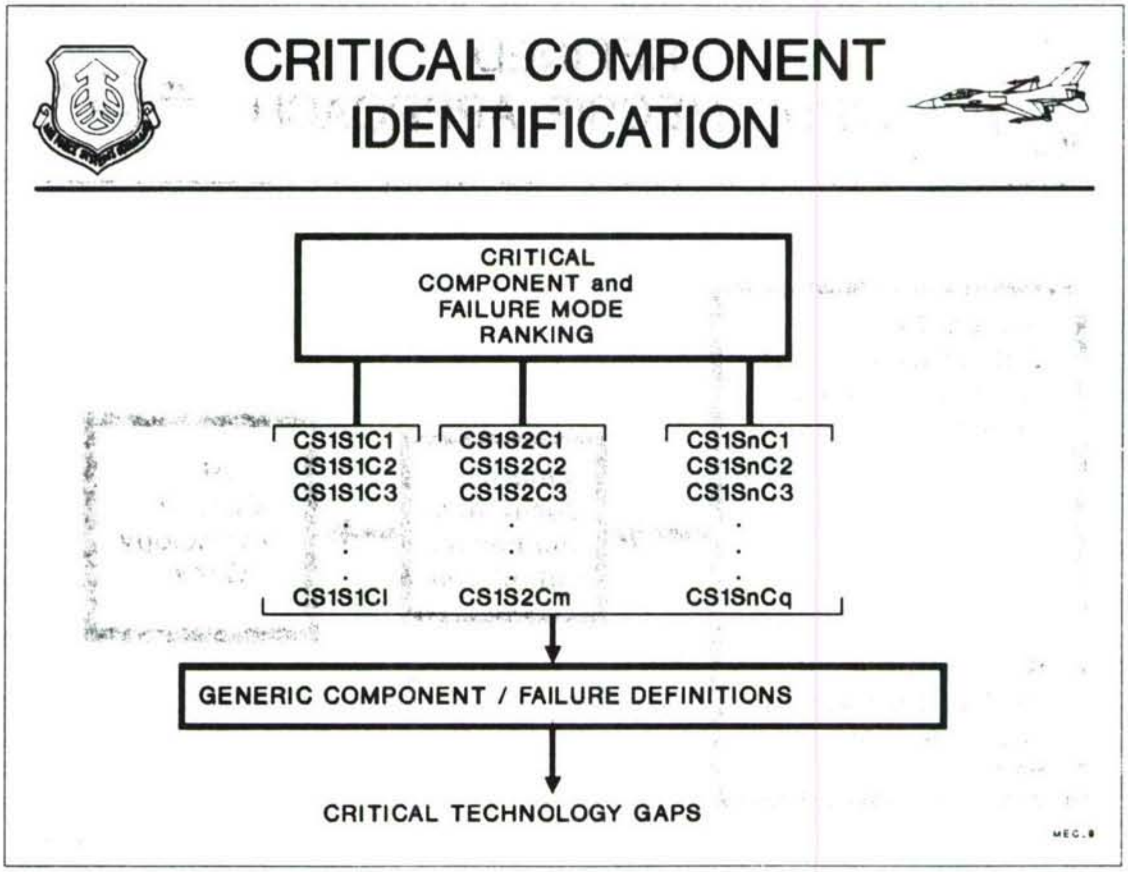

SLIDE #	NAME	DESCRIPTION
21	MEC_4 .CHT	

A refined WRDC MECSIP approach shown here now has the integrating function using the generic component & failure definitions which allow the definition of the technology gaps which have the major impact on all A.F. systems.

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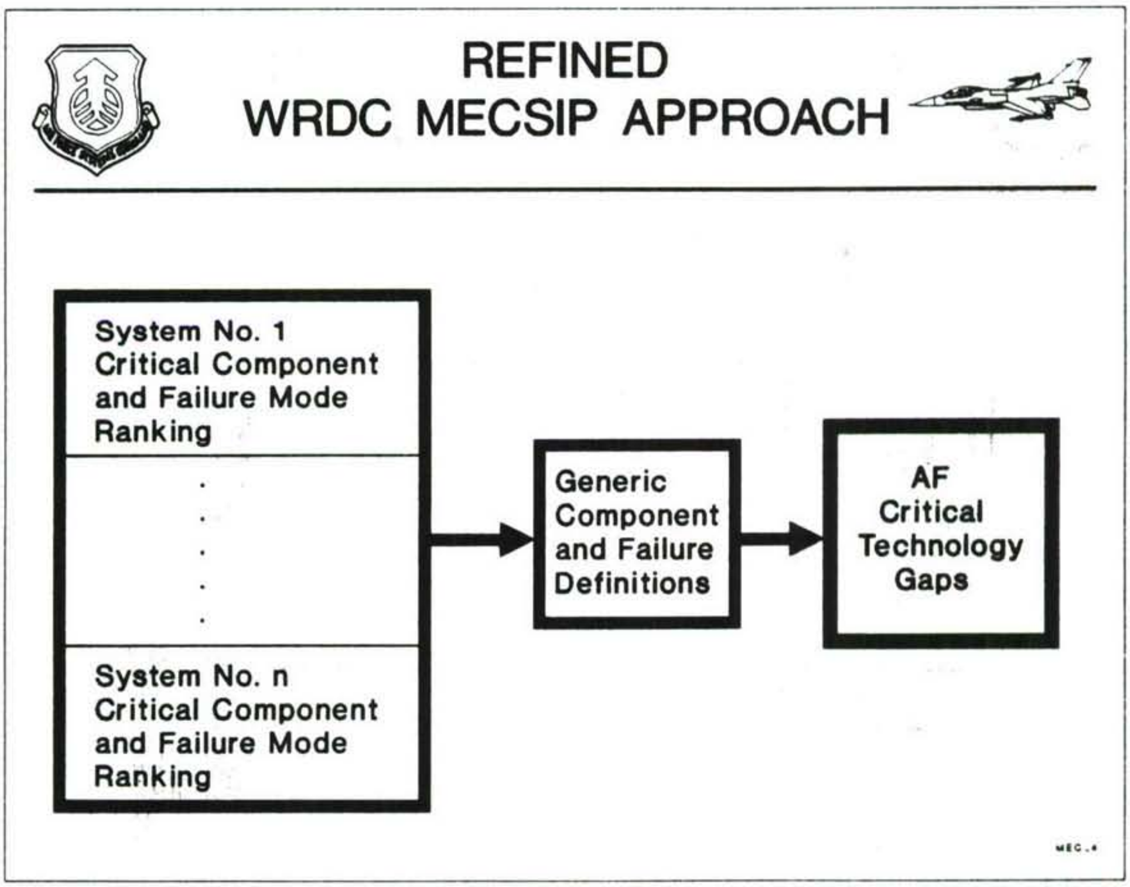

SLIDE #	NAME	DESCRIPTION
22	MEC_16 .CHT	

To summarize-

The WRDC MECSIP approach would start with an analysis using the DO56 & GO33 data systems and WSMIS & MODAS type analysis tools. Critical subsystems & sub-subsystems will be established. Using this information new translators will be generated to proceed from WUC based information to component information utilizing DO49 & GO21 information supplimented with technical teardown reports and other surveys. All results will then be used as input for a WRDC MECSIP operator. The output of this operator will be the critical component rankings and critical technology gaps.

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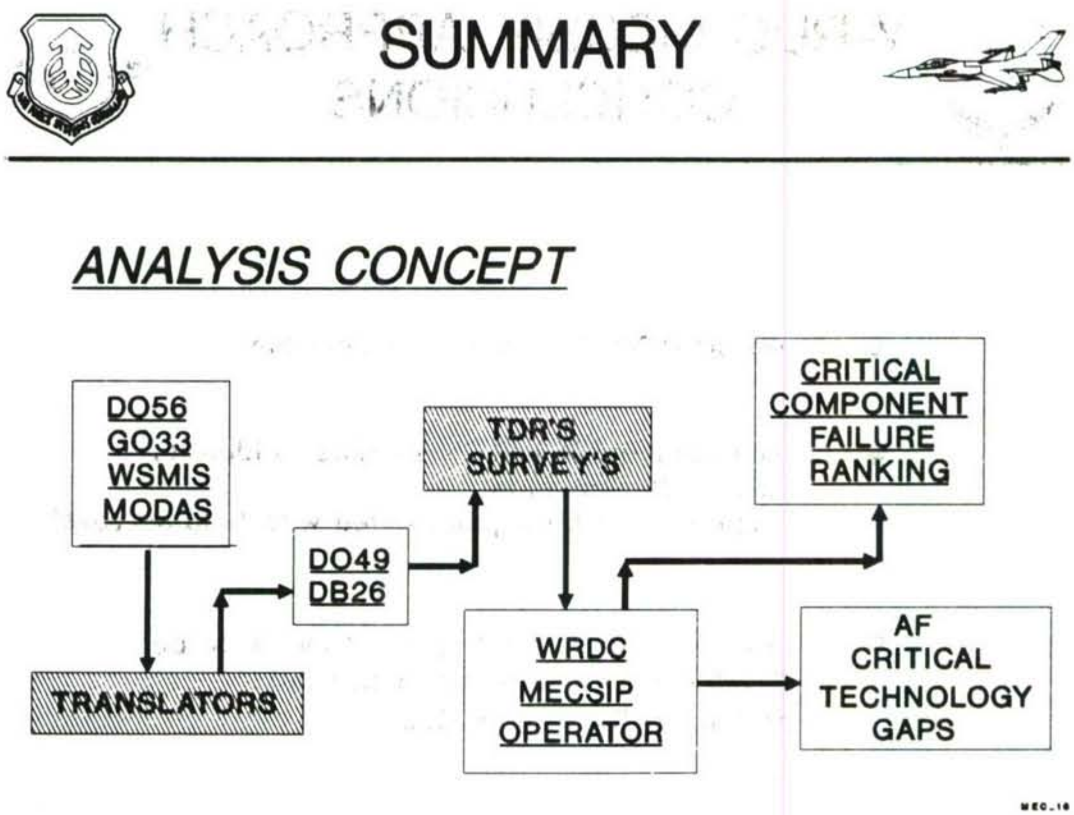

SLIDE #	NAME	DESCRIPTION
23	MEC_19 .CHT	

The conclusions of the feasibility study are as shown.

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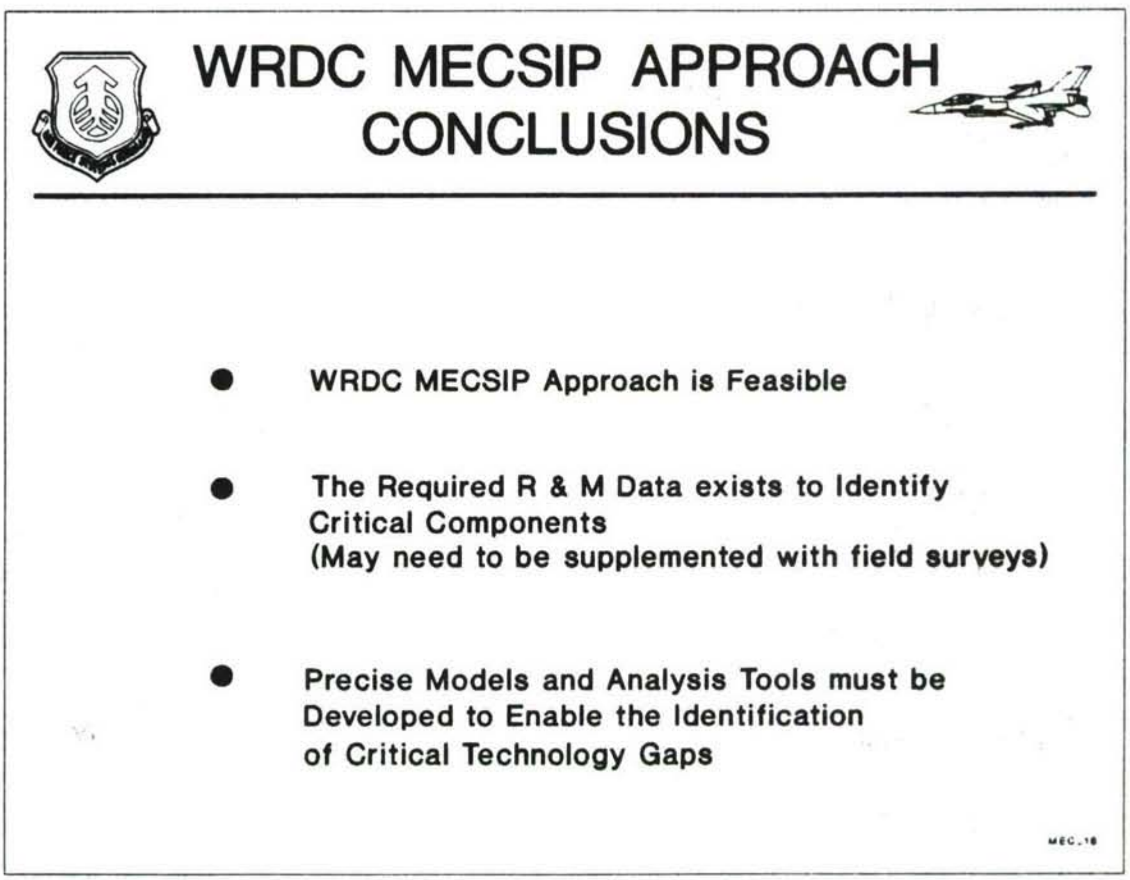


Force Management Technology Application To Aging Aircraft

USAF Structural Integrity Program Conference

San Antonio, Texas
5-7 December 1989

Dr. T. F. Christian, WR-ALC
D. O. Hammond, WR-ALC

J. B. Cochran, LASC-G
D. S. Morcock, LASC-G

GA-8845-01
11-16-89

Force Management technology applications are alive and well at WR-ALC for C-130 and C-141 cargo aircraft. Dr. Tom Christian is the C-130 Aircraft Structural Integrity Program (ASIP) Manager at WR-ALC, and Dev Hammond is the C-141 ASIP Manager. Joe Cochran and Doug Morcock are C-141 Force Management Program engineers at LASC-Georgia Division. This presentation is based on the C-141 but is a generic discussion regarding Force Management of aging aircraft, particularly cargo/tanker aircraft.

Force Management Technology Application To Aging Aircraft

Background

- Recent Commercial Aircraft Incidents Highlight "Aging Aircraft" Problems
- Military Cargo Aircraft Entering "Aging Aircraft" Era of Operations
  - Produced in 1960's
  - Approaching/Exceeding One Design Lifetime of Usage
- Extended Usage Likely
  - Budget Constraints
  - Delayed Production of Replacement Aircraft
  - Increasing Military Airlift Requirements

GA-8845-2
11-16-89

BACKGROUND

"Aging Aircraft" problems have dominated the aircraft news scene in recent months with commercial aircraft incidents. Military transport aircraft are subject to similar problems. Many of the present military cargo/tanker aircraft were produced in the 1960's and have accrued service usage approaching or exceeding the original design lifetime. It is likely that extended usage of these aircraft will be necessary due to budget constraints and a resulting delay in the production of replacement aircraft such as the C-17. It is also likely that military airlift requirements will increase in the 1990's due to changing world conditions. This will necessitate still further extension of present aircraft usage beyond IOC of the new aircraft.

Force Management Technology Application To Aging Aircraft

Background (Cont'd)

- Basic Aircraft Designs/Development Support Extended Usage
  - "Fail-Safe" Structure
  - Static Strength, Durability, Residual Strength Design and Tests
- Force Management "Tools" Available To Support Extended Usage
- All Aspects of Force Management Should Be Reviewed
  - Structure
  - Systems
  - Corrosion
  - Past, Current, Projected Usage
  - Inspection/Mod/Maintenance Actions
  - Reliability & Maintainability
  - Resources and Support Requirements

GA-8845-3
11-16-89

BACKGROUND (Cont'd.)

Fortunately, the capacity for extended operational usage has been "built in" to the present military cargo/tanker aircraft. "Fail-safe" design of the structure should ensure safe extended usage of the properly maintained and operated aircraft. Force Management "tools" are available to aid in this task. However, we need to address not only the structural capability for extended operations but also the other considerations which affect optimum usage and safety of the "aging aircraft." These will be discussed further in this presentation.

Force Management Technology Application To Aging Aircraft

Outline

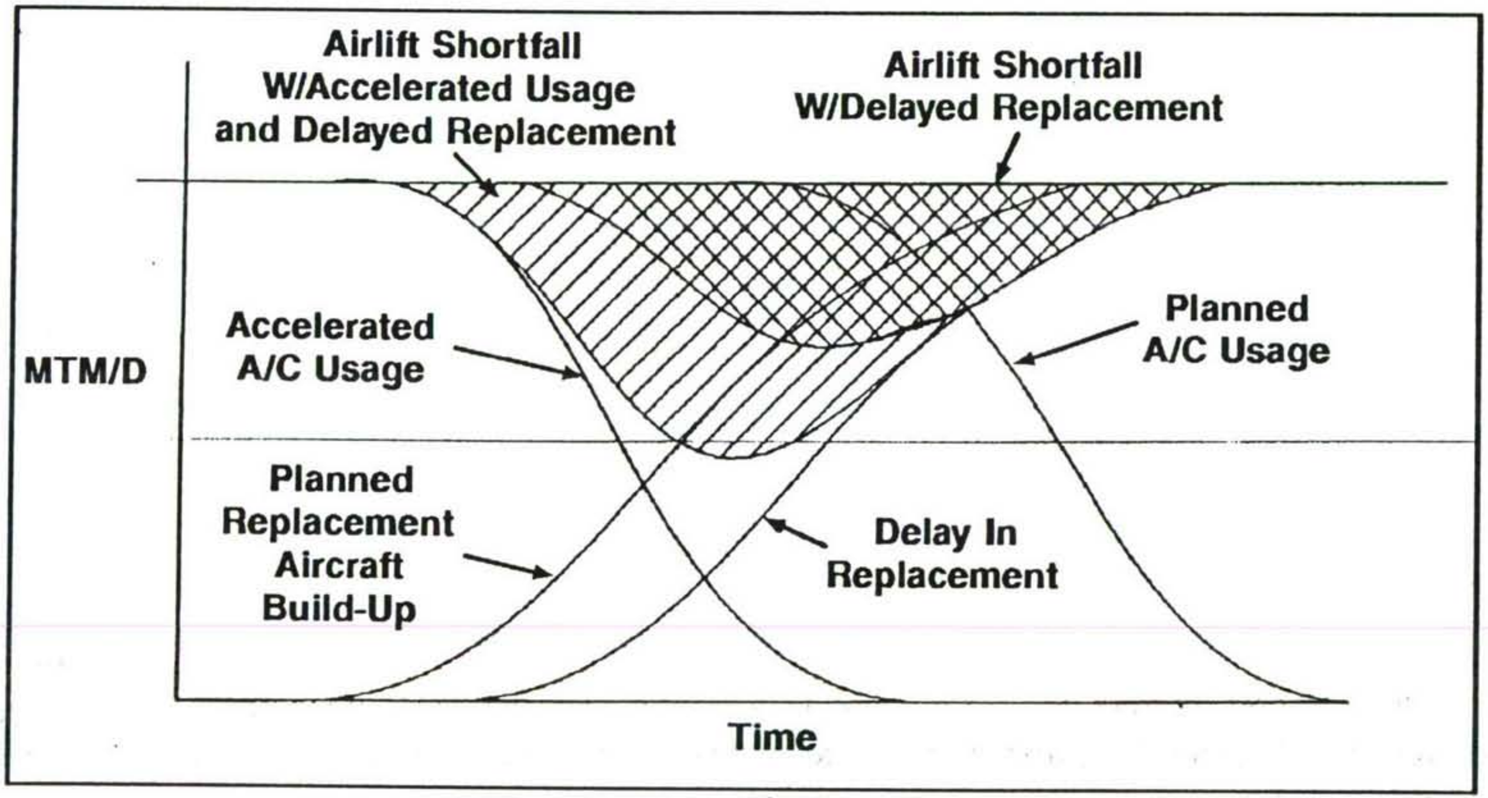
- Need
- When "Aging"?
- Tools
- Applications
- Phaseout Indicators

GA-8845-4

OUTLINE

Today's presentation will briefly discuss the need for extended usage, when an aircraft is considered "old", what are the available "tools", applications for Force Management of "aging" aircraft, and indicators of the need for phase-out of the weapon system.

Force Management Technology Application To Aging Aircraft



GA-8845-06
10-12-89

THE NEED (Cont'd.)

Why not just fly the aircraft as originally planned and then retire it?

This graph shows possible effects on airlift capability of increased usage severity for present aircraft and delay in production of a replacement aircraft. Airlift shortfall below national security requirements can occur if phaseout of the present aircraft is too soon. With timely planning and action, the present aircraft operation can be extended until the replacement is on line.

Force Management Technology Application To Aging Aircraft

When Is An Aircraft "Aging"?

- Approaching/Exceeding Design Lifetime
- Wear and Tear
- Increasing Down Time, Maintenance Costs
- Systems Degrading (e.g., Wiring)
- Supportability Increasingly Difficult

GA-8845-7
10-17-89

WHEN "AGING"?

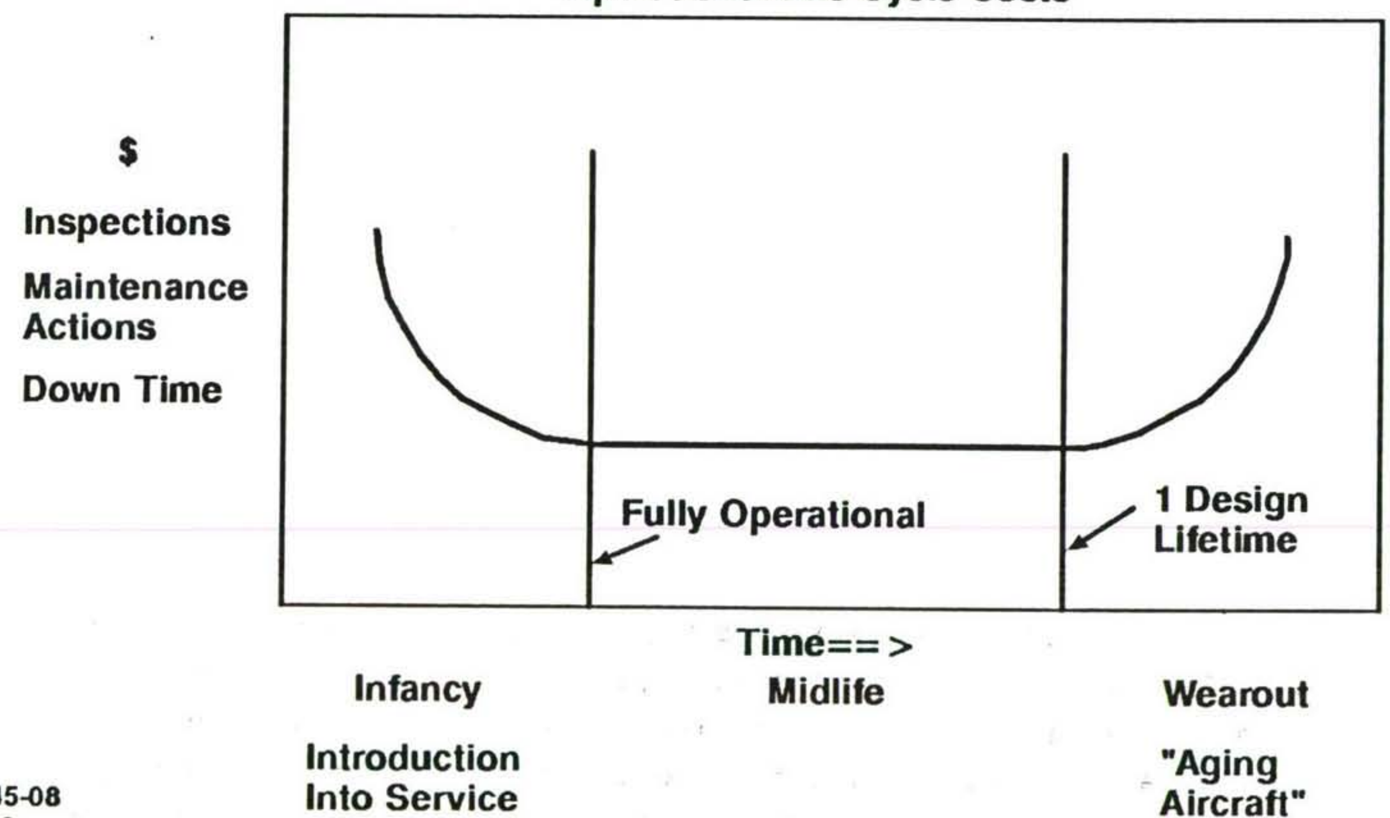
When is an aircraft "aging" or "getting old"?

Simply stated, an aircraft is "getting old" when keeping it flying becomes a pain. This can be expressed in the several ways shown here, and is generally a combination of most of them. Safe operation of present aircraft can be extended "almost indefinitely," but at the expense of more and more down time and maintenance cost.

Force Management Technology Application To Aging Aircraft

When Is An Aircraft "Aging" ? (Cont'd)

Operational Life Cycle Costs

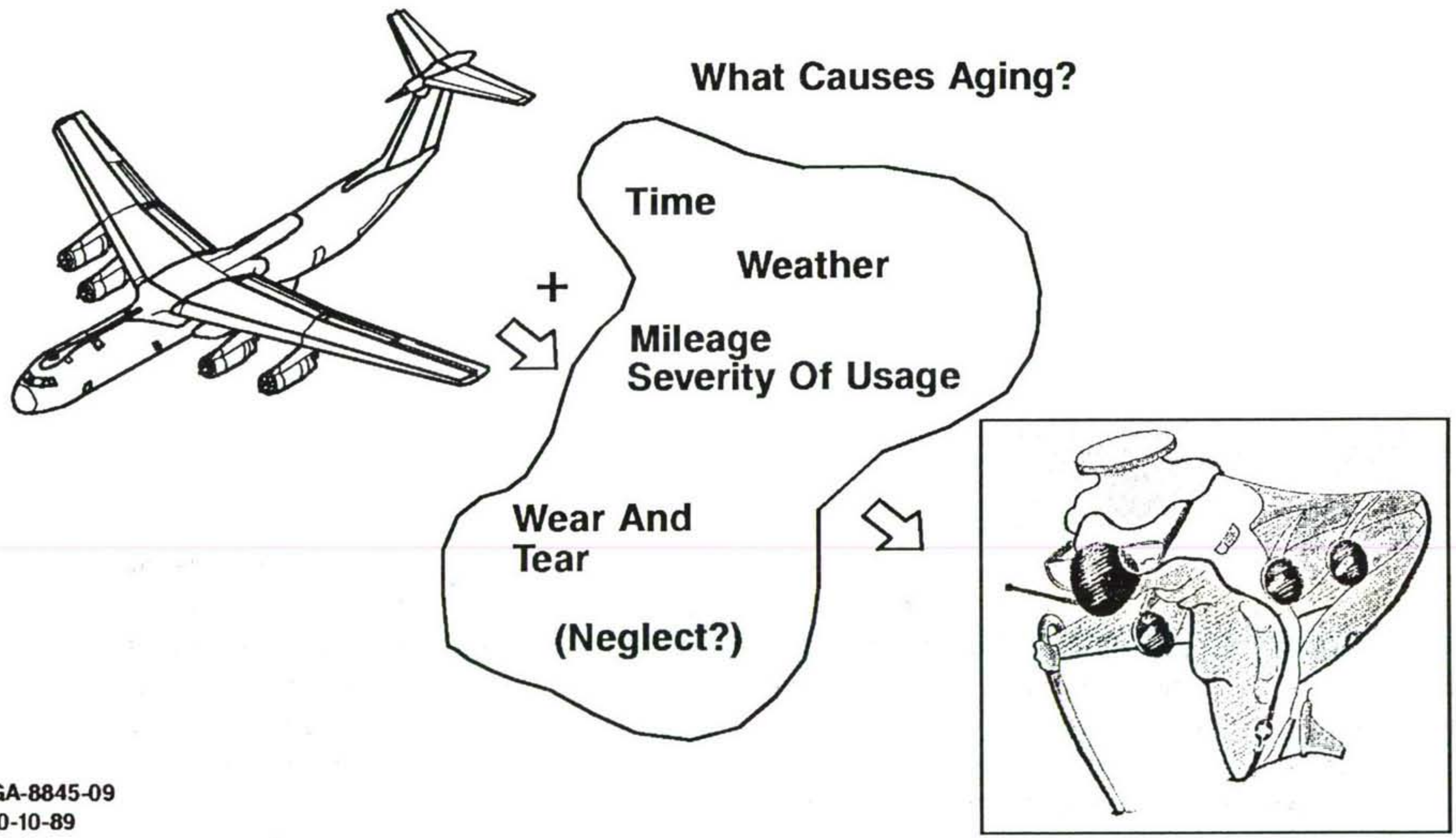


GA-8845-08
10-10-89

WHEN "AGING" (Cont'd.)

This concept of "aging" can be illustrated by this "bathtub" curve. When aircraft first enter service, more time is spent on inspection and maintenance until the characteristics of the system can be established. During its operational life, planned maintenance intervals tend to be gradually extended, while unscheduled repair/replacement actions gradually increase. Beginning at about one lifetime of equivalent severity of usage, these required actions begin to escalate, with increasing down time and costs.

Force Management Technology Application To Aging Aircraft



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WHAT CAUSES AGING?

The causes of aging are "intuitively obvious." Less obvious is how very significant these aging factors are on aircraft. The exposure to the elements, corrosive environment, wear and tear, and maintenance and protective measures can make the difference between extended usage capability and premature phase-out. A good analogy would be to assume you drive an automobile with all-aluminum structure. It can be protected and operated properly and last "indefinitely," or it can be neglected and abused and early replacement will be necessary.

Force Management Technology Application To Aging Aircraft

Force Management Tools

- Built-In Capabilities
  - Static Strength
  - Durability
  - Residual Strength
  - Damage Tolerance
  - Redundant Systems
  - Qualification Tests
  - MTBF
  - Corrosion Protection
  - Composite Repairs
- Service Usage Monitoring and Analyses
  - Loads/Environment Spectra Survey
  - Individual Airplane Tracking Program
  - Structural Maintenance Records
  - Analyses, Updates
  - $\boxed{\text{Data Feedback}}$

GA-8845-10
11-16-89

FORCE MANAGEMENT TOOLS

Military standards and aircraft design/production practices have been developed over the years based on experience and advancements in technology. The present aircraft, generally developed in the 1960's, have built-in capabilities for extended usage. In addition, service usage monitoring programs are in place which can aid in Force Management. Also, new Force Management tools continue to be developed, primarily in the areas of corrosion protection, composite repairs, and management techniques.

"Data Feedback" is the gasoline that makes the engine run. We cannot over emphasize this aspect of Force Management.

Force Management Technology Application To Aging Aircraft

Force Management Tools (Cont'd)

- Developing Technologies
  - Corrosion Prevention Integrity Program (CPIP)
  - Functional Systems Integrity Program (FSIP)
  - Resources & Support (R&S) Evaluations
  - Weapon System Master Plan (WSMP)
  - Automation Tools

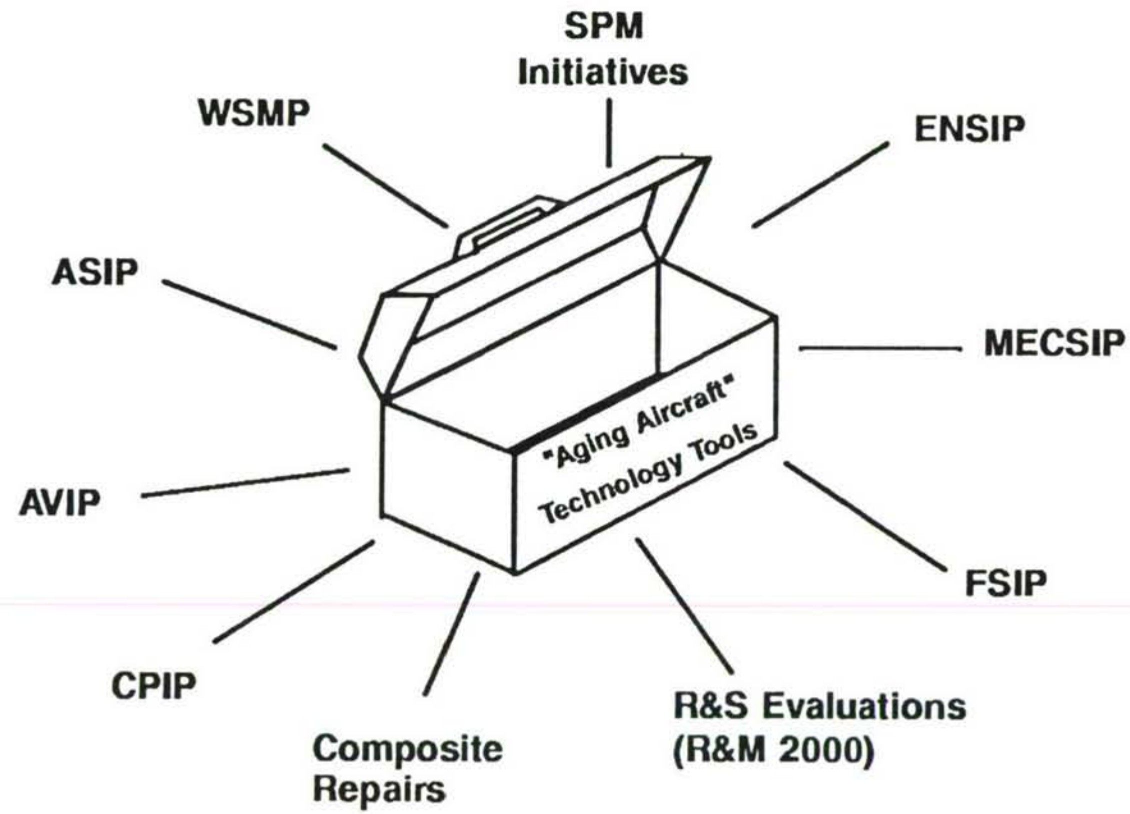
GA-8845-11
10-17-89

FORCE MANAGEMENT TOOLS (Cont'd.)

Developing technologies particularly useful for aging aircraft are listed here. I will discuss these individually in more detail. "Automation Tools" refers to computerized collection, using, and displaying of Force Management data.

Force Management Technology Application To Aging Aircraft

Force Management Tools (Cont'd)



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10-10-89

FORCE MANAGEMENT TOOL BOX

The Systems Program Manager (SPM) has a box of technology tools available for his use in managing an aging weapon system. Many of these tools (programs) are still being developed, but the concepts have been discussed in annual Aircraft Structural Integrity Program (ASIP) conferences over the last ten years. The present need is to continue development, integrate the tools, and apply them in a concerted manner to meet aging aircraft usage goals.

We have included "SPM Initiatives" as a technology tool. The effectiveness of anything in the tool box is directly dependent on the skill and initiative of the System Program Manager.

Force Management Technology Application To Aging Aircraft

Aircraft Structural Integrity Program (ASIP) MIL-STD-1530A

Task I	Task II	Task III	Task IV	Task V
Design Information	Design Analyses and Development Tests	Full Scale Testing	Force Management Data Package	Force Management
ASIP Master Plan Structural Design Criteria Damage Tolerance & Durability Control Plans Selection of Mat'ls, Processes, & Joining Methods	Materials and Joint Allowables Load Analysis Design Service Loads Spectra Design Chemical/ Thermal Environment Spectra	Static Tests Durability Tests Damage Tolerance Tests Flight & Ground Operations Tests Sonic Tests Flight Vibration Tests	Final Analyses Strength Summary Force Structural Maintenance Plan Loads/Environment Spectra Survey Individual Airplane Tracking Program	Loads/Environment Spectra Survey Individual Airplane Tracking Data Individual Airplane Maintenance Times Structural Maintenance Records

GA-8845-13
11-16-89

AIRCRAFT STRUCTURAL INTEGRITY PROGRAM (ASIP)

MIL-STD-1530A and its supporting Military Specifications provide a roadmap for design, development, testing, fielding, and Force Management of a weapon system. The Aircraft Structural Integrity Program (ASIP) defined by these documents is the most advanced of all the Force Management tools available to the Systems Program Manager (SPM). Its contents are well known to ASIP personnel. However, this tool is continually changing as the life cycle for a weapon system goes on and as new technologies such as composite repairs and automation become available.

Force Management Technology Application To Aging Aircraft

Why a Corrosion Prevention Integrity Program

- Corrosion Is the Life Limiter
- Corrosion Is Costly
  - Repairs
  - Downtime
  - Safety
  - Readiness
- A Program Is Required To Enhance Service Life
- Avoid the Same Fate of Civil Aviation
  - Spending $800M for Corrosion Repairs

GA-8845-14
10-17-89

WHY A CORROSION PREVENTION INTEGRITY PROGRAM

A vital, new, developing Force Management tool is the Corrosion Prevention Integrity Program (CPIP). Aircraft have been designed and built for corrosion prevention for years; also, they have corroded and received corrosion control maintenance for years. But the primary keys to extended operation of aging aircraft are structural cracking and corrosion damage, and corrosion can precipitate and accelerate structural cracking. Corrosion efforts, including repairs of corrosion damage actually caused by improper corrosion control measures, has cost many manhours and down days. An aggressive, organized Corrosion Prevention and Control Program is required to enhance service life. New FAA requirements for aging civil aircraft are present evidence of the need and potential cost of such a program - but the alternatives are safety hazards with still more costly repairs or early phaseout.

Force Management Technology Application To Aging Aircraft

Corrosion Tracking and Prediction Program

- For Tracking Structural Repairs/Corrosion
- For Corrosion Prediction
- Can Be Used To Track Systems Problems
- Will Provide a "Handle" on Corrosion
  - Manning
  - Costs
  - Procurement
  - Repair Projection

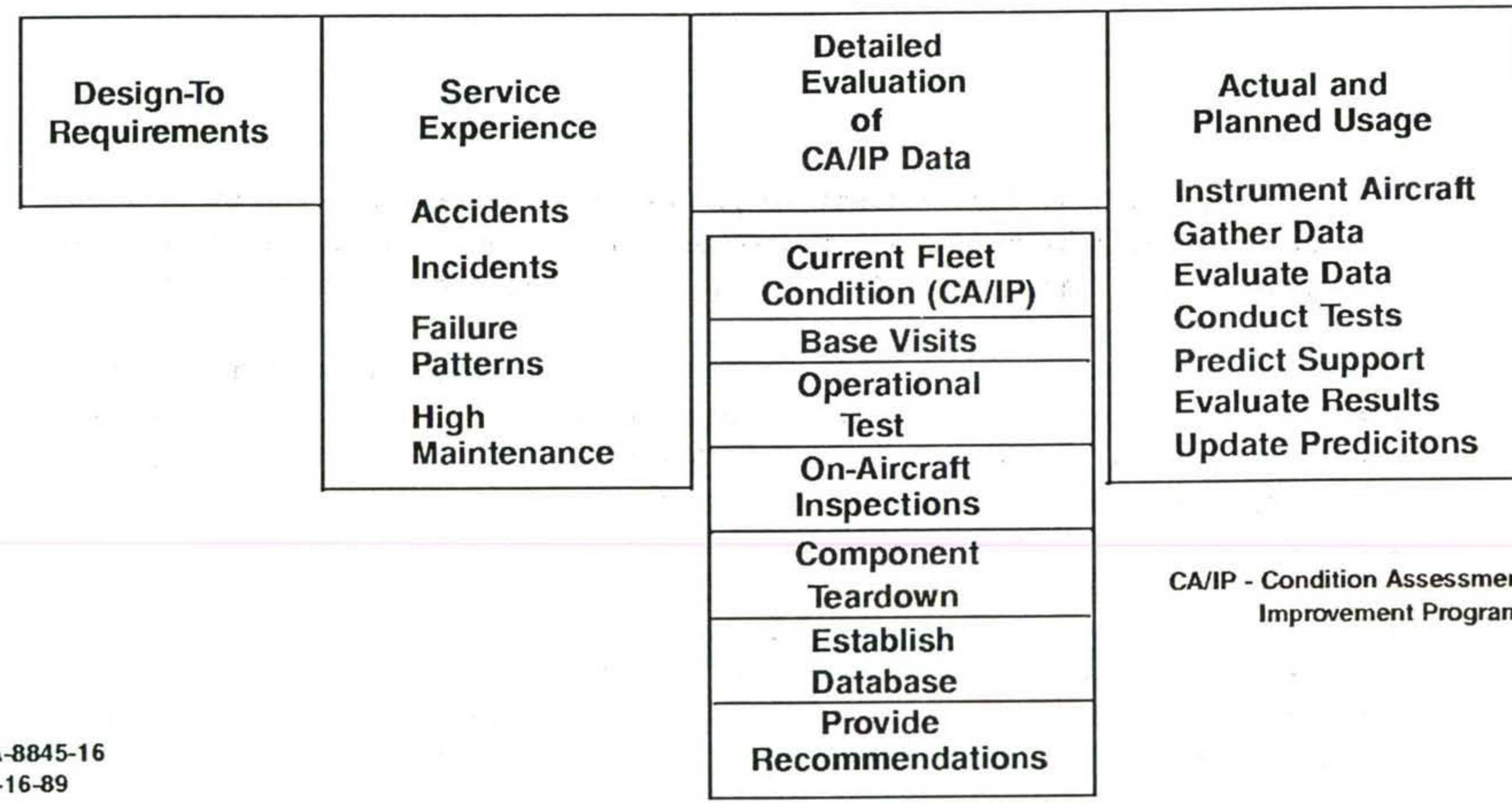
GA-8845-15
10-17-89

CORROSION TRACKING AND PREDICTION PROGRAM

A computerized Corrosion Tracking and Prediction Program has been developed for the C-141 and is being implemented at WR-ALC. The program is essentially generic. It is patterned after the Individual Aircraft Tracking (IAT) structural tracking program. It includes a corrosion repairs data base and monitoring of corrosion "drivers" for individual aircraft. When the data are analyzed, the prediction module will be adjusted to identify likely regions of corrosion and predict when preventive or control action should be taken. Costs, manhours, and parts replacement or repair projections are also anticipated.

Force Management Technology Application To Aging Aircraft

Functional Systems Integrity Program (FSIP)



CA/IP - Condition Assessment/ Improvement Program

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11-16-89

FUNCTIONAL SYSTEMS/INTEGRITY PROGRAM (FSIP)

Another new Force Management tool is the Functional Systems Integrity Program (FSIP). Functional systems have a development cycle similar to the ASIP for structure. However, the Force Management monitoring for operational aircraft appears to not be organized. Recent actions have been initiated by Air Force to obtain condition data through the Condition Assessment/Improvement Program (CA/IP). This effort is presently underway for C-130 and C-141 aircraft and is yielding significant data which will aid in spares procurement and systems improvement planning. Further action for an effective FSIP includes defining and monitoring the parameters contributing to the functional system problems. The CA/IP, MECSIP, ENSIP, and AVIP initiatives from AFLC are elements of an overal FSIP.

This Force Management "tool" is vital for aging aircraft. Structural failures of functional components used beyond their planned lifetimes have resulted in loss of several civilian and military transport aircraft.

Force Management Technology Application To Aging Aircraft

Composite Repairs - Benefits

- No Fastener Removals From Damaged Structure
- No Fasteners To Attach Patch
- Retard Damage Progression Significantly
- Reliable, Practical Fab/Installation Method
- Minimum Development Testing
  - Use/Extend ARL Methods Wherever Possible
  - Detailed Finite Element Modeling and Analyses
- Benefit Force Management
  - Cost/Time Effective Over Metal Repair
  - Reduced Inspection Burden

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11-16-89

COMPOSITE REPAIRS

Composite repairs for metal structure are an emerging technology demonstrating significant cost and down time benefits compared to conventional metal repairs. This "tool" was initially developed by Australian Research Labs (ARL) and used on C-130, F-111, and Mirage aircraft. With WR-ALC and RAMTIP funding, LASC-Georgia Division has extended applications to the C-141 and defined a composite repairs kit for Air Force use. LASC-G also is developing repairs for the C-130. C-141 composite repairs are on five aircraft to date, with additional installations planned. The boron/epoxy or graphite/epoxy repair doublers are bonded onto the structure without addition or removal of fasteners, and effectively retard crack growth. For known problem areas, preventive doubler installations can be applied to aging aircraft at a scheduled down time. The potential R&M benefits of this "tool" are impressive, and composite repairs will contribute significantly to aging aircraft airlift capability.

Force Management Technology Application To Aging Aircraft

Resources & Support (R&S) Program Objectives

- Develop the Capability To Link WSMP and MAC R&M Action Plan
- Develop a System That Prioritizes Actions Required To Support User Requirements Within the Limits of Available Funding
- Provide Coordination and Timely Support to Other Force Management Activities

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11-16-89

RESOURCES & SUPPORT (R&S) PROGRAM

In the present fiscal environment, we all see more needs for action than there are resources to apply to the needs. This fact of life has increased the need for the Reliability and Maintainability (R&M) or Resources and Support (R&S) Force Management tool to provide guidance regarding best use of the available resources. The R&S Program is to link the Weapon System Master Plan (WSMP) requirements and the MAC R&M Action Plan requirements for an overall Weapon System Force Management Program. It also is to aid in obtaining timely and appropriate action by the many support elements of the overall SPM fuction.

Force Management Technology Application To Aging Aircraft

R&S Program Development Process

- Establish Measures of Merit
- Evaluate Current Capability
- Evaluate 10 Year Projection
- Prioritize Against Available Funding

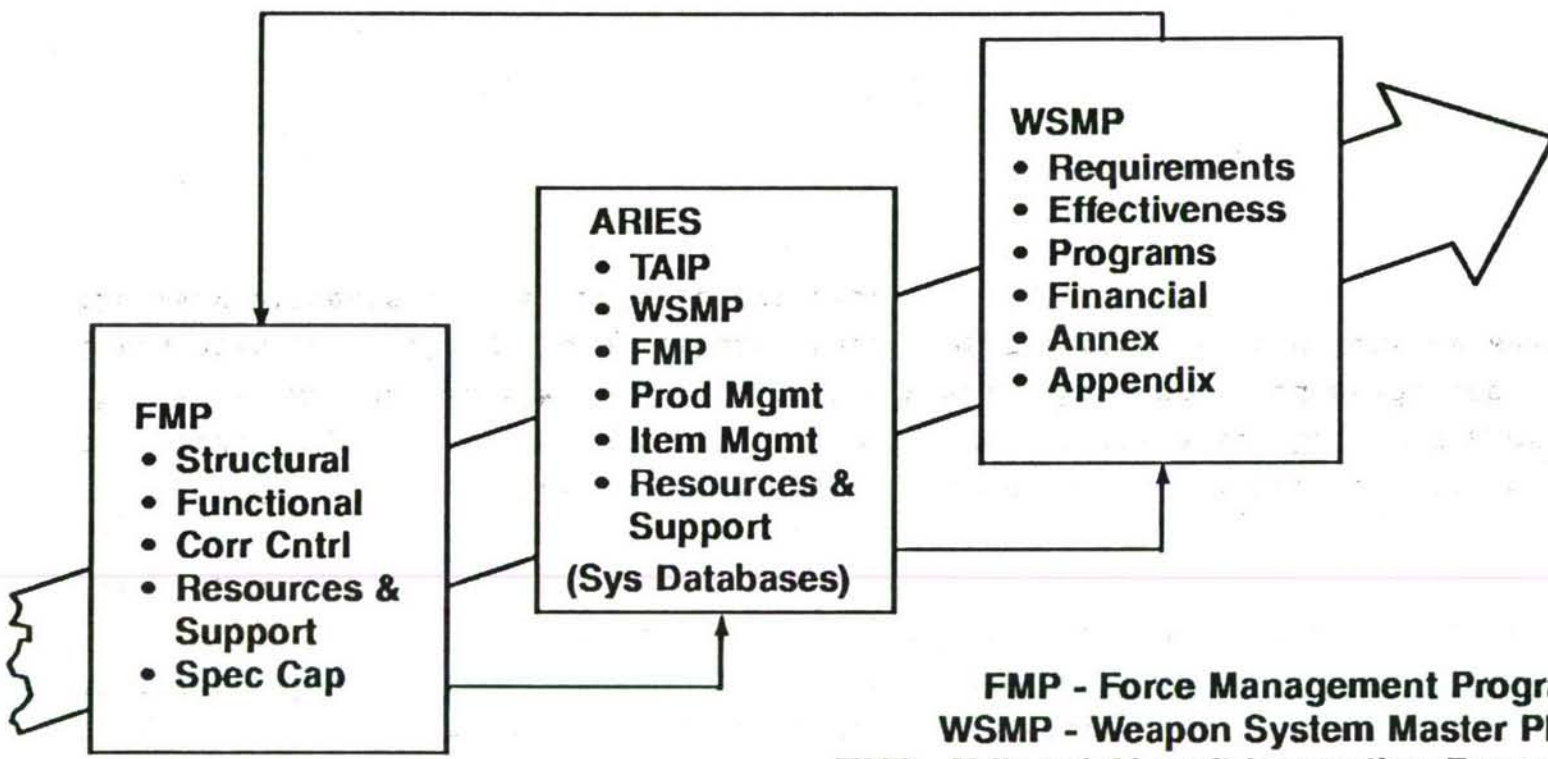
GA-8845-19
11-16-89

R&S PROGRAM DEVELOPMENT PROCESS

How do we develop a Resources & Support program for aging aircraft? First, we should establish the means (Measures of Merit) of evaluating the potential Force Management candidate activities, followed by an assessment of the current capability and needs of each support activity. Next, a projection of future needs is necessary (ten years ahead if possible). With this knowledge, the candidate needs must be prioritized against available funding to establish an R&M Action Plan for that weapon system and to identify shortfalls and their impact for future adjustments. This is a challenging task and must utilize statistical tools and a structured approach to the needs across many disciplines and organizational boundaries.

Force Management Technology Application To Aging Aircraft

FMP/WSMP,Integration



FMP - Force Management Program
WSMP - Weapon System Master Plan
TAIP - Tailored Aircraft Inspection Program
ARIES - Automated Readiness Integrated Engineering System

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10-10-89

FMP/WSMP, INTEGRATION

As you know, managing a weapon system is a real challenge! The Air Force Weapon System Master Plan (WSMP) is an aid to covering all the bases. AFLC is working to automate the generic WSMP for improved efficiency, commonality, and access. The individual weapon system SPM must respond to the WSMP and also benefits from it. He also has locally generated tools such as the Individual Aircraft Tracking (IAT) program. The overall combination of activities for which the SPM is responsible is termed the Force Management Program (FMP) for his weapon system.

This diagram shows the relationship of the FMP and the WSMP. The block in the middle represents the computerized system data bases available to support the SPM's FMP and the SPM's portion of the WSMP. This Automated Readiness Integrated Engineering System (ARIES) furnishes the computerized link between the many elements of the SPM's activities.

Force Management Technology Application To Aging Aircraft

System Program Manager (SPM) Initiatives

- SPM Responsible for Overall System
- SPM Primary Feedback To Weapon System Master Plan (WSMP)
- SPM Knows "Health" of His System and Its Support Functions
- SPM Can Evaluate Relative Merits of Life Extension Options
- Tasking for Life Extension Comes From User; SPM Implements
- SPM Obtains, Controls Resources for His Weapon System
  - Initiatives and Emphases Reflect SPM's Interests
- SPM Is $\boxed{\text{Key}}$ To Optimum Force Management

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11-16-89

SPM INITIATIVES

The System Program Manager (SPM) is the <u>key</u> to optimum Force Management. His initiative determines the effectiveness and future potential of his weapon system. This is particularly true with regard to the aging aircraft. The SPM is the one office knowledgeable of and responsible for the "health" of his aircraft and all its support functions. His initiative is a Force Management "tool" in itself, producing and affecting the quality of all the other "tools" at his disposal.

Force Management can be accomplished pro-actively, minimizing surprises and delays and cost-effectively maximizing safety and longevity. Or, at the other extreme, Force Management can be accomplished through "crisis management" just by managing reactively. Optimum management of aging aircraft is a constant uphill struggle for the SPM - but isn't the view from the top worth it?

Force Management Technology Application To Aging Aircraft

Applications

- Existing and Developing Tools Can Be Used To
  - Optimize Inspection/Mod/Maintenance Actions
  - Maximize Airlift Capability
  - Preserve Assets for as Long as Required
  - Predict Phaseout
- Air Force Must Establish Requirements for Extended Operation
- Comprehensive Aging Aircraft Program Necessary for Best Use of Assets
  - Structure
  - Avionics
  - Corrosion Prevention
  - Functional Systems
  - Capabilities

GA-8845-22
11-16-89

APPLICATIONS

We've considered the available Force Management tools briefly at length. Now let's talk about applications to aging aircraft. As we see it,

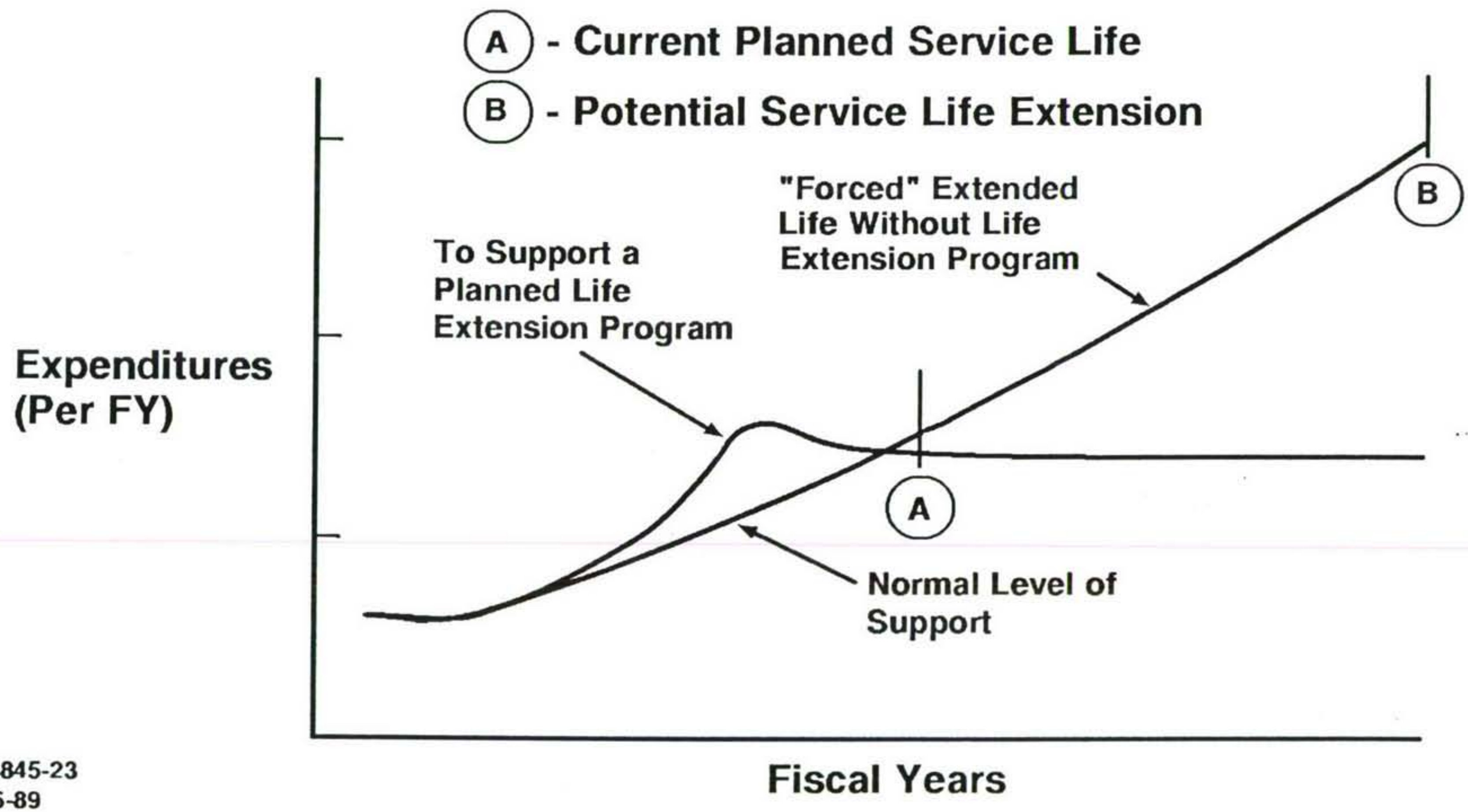
- Tools are available and under development which can be used to achieve the Air Force goals for that weapon system, <u>provided they are used in time.</u>

- Air Force must specify that extended usage is necessary; otherwise, the SPM has no impetus to make it happen.

- For optimum Force Management, a comprehensive aging aircraft program is necessary. The "health" of all contributors to overall longetivy must be maintained throughout the period of usage.

Force Management Technology Application To Aging Aircraft

Funding Impact



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FUNDING IMPACT

This graph illustrates the benefits of planning ahead for extended usage. If the aircraft is used for an extended time on a "business as usual" basis, down time and repair costs will escalate as shown in Point B. However, if the requirement for extended usage is established and a life extension program is initiated in a timely manner, the costs and down time prior to achievement of the current planned service life (Point A) may be higher, but the overall costs and down time will be significantly less.

This graph was developed to show the benefit of timely engineering task funding for the life extension, but the principle holds for maintenance costs and aircraft availability as well.

Force Management Technology Application To Aging Aircraft

Safety and Utilization

- Tools: Force Management Tools
- Actions:
  - Maintain
  - Feedback Results Very Important
  - Analyze
  - Inspect/Mod/Repair/Replace Components for Optimum Utilization With Safety and Cost-Effectiveness

GA-8845-24
10-18-89

SAFETY AND UTILIZATION

For effective Force Management, tools are available or in development. However, they must be used for optimum safety and aircraft availability. The needed actions are obvious but require constant emphasis in Air Force circles:

- The aircraft must be maintained, particularly regarding proper corrosion protection and lubrication.

- Data on aircraft usage and inspection/maintenance/repair action must be fed back and analyzed to determine "health" and project next actions.

- Planning for extended usage should consider the several options of inspection, modification, repair, or replacement of troublesome components. Safety and cost-effectiveness with minimum down time will result from a mix of these options for different components.

Force Management Technology Application To Aging Aircraft

Note: Timeliness Is Essential!

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10-10-89

TIMELINESS

Again, let me emphasize that timeliness is essential! "It's too late to close the gate after the horse is out!"

Force Management Technology Application To Aging Aircraft

Concerns of Extended Operational Usage

- Delays in Inspection and/or Repairs Can Cause Loss of Aircraft Availability, Greatly Increased Repair Cost, and Safety of Flight Concerns
- Inadequate Inspection Feedback Increases These Problems
- Increased Usage or Severity of Usage May Deplete Remaining Structural Capability Below MAC Airlift Needs

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11-16-89

CONCERNS OF EXTENDED OPERATIONAL USAGE

With aging aircraft there is an increasing need for "Tender, Loving Care." The likelihood of structural cracking or functional systems breakdown is increased. There should be a corresponding increase in the attention given to aircraft condition. Delays in inspection or repairs and inadequate feedback of condition data can result in increased overall costs, reduced safety of flight, and aircraft availability. In addition, changes in usage or severity of usage must be monitored and appropriate structural/functional actions initiated to ensure that MAC airlift needs can be supported. Aggressive Force management is necessary to offset the tendency to neglect the "old shoe."

Force Management Technology Application To Aging Aircraft

Phaseout Indicators and Drivers

- Increased Operational Cost - Uneconomical to Operate
- Escalating Repairs/Down Time Beyond Acceptable Levels
- Repairs/Replacements for Safety Beyond Economic Limits
- Corrosion Requiring Excessive Repair/Replacement Time and $
- Obsolescence
- Systems Not Supportable
- Replacement Aircraft
- New Threats - Survivability Requirements Cannot Be Met Economically
- Capabilities No Longer Required

GA-8845-27
11-16-89

PHASEOUT INDICATORS AND DRIVERS

How do we know when phaseout is approaching? - A rule of thumb would say that if the design operational life is being approached, then a life extension program should be implemented or phaseout should be planned. Without the life extension program, continued operation is possible but at increased risk and the escalating costs and down time of the "forced life extension" mode. A primary consideration, corrosion, can result in premature phaseout.

Other "drivers" are listed here which relate to the aircraft no longer being capable of accomplishing its task due to changing threat environments beyond economic feasibility to overcome, or no longer being needed because a better way to perform its mission is available, or the mission itself has changed or is no longer necessary. These considerations are external forces outside the matter of aging aircraft but are often the real reason for phaseout.

Force Management Technology Application To Aging Aircraft

Summary

- Aircraft Now Being Utilized Beyond Original Design Lifetimes
- Existing and Developing Force Management Tools Can Optimize This Usage
  - Maximize Airlift Capability Through Reduced Down Time
  - Minimize Inspection/Mod/Maintenance Costs
  - Predict Phaseout
- Service Data Feedback and Analysis Essential To Force Management
- Air Force Must Establish Requirements for Extended Usage
- Comprehensive Aging Aircraft Program Must Be Established
  - Plan Ahead
  - Work the Plan
- Timeliness Is Essential

GA-8845-28
11-16-89

SUMMARY

To summarize:

- Airlift capability is now being required of weapon systems beyond their original design goals. The airframe is normally capable of this extended usage.

- There are Force Management tools available and in development which can aid in optimum Force Management for extended usage.

- Proper use of the tools, and particularly the feedback and analysis of data regarding inspection/maintenance/mod/repair actions, is essential to this optimum Force Management.

- For effective management of aging aircraft, it is necessary that the Air Force establish requirements for any desired extended usage. Otherwise, resources will not be made available for required actions to support the extension.

- A comprehensive operational usage extension program must be established and implemented to minimize overall life cycle costs and down time.

- Timeliness is essential. Lead times for funding contracts, parts, program development, and implementation are long. "It's too late to close the barn door after the horse is out."

- Finally, with proper Force Management, optimum usage of the aging aircraft should be possible until phaseout is caused by external "drivers" rather than by wear and tear.

Force Management Technology Application To Aging Aircraft

Summary (Cont'd)

What We Know About the Aircraft	+	How We Use And Determine Aircraft Health	⇨	Determines How We Manage And Optimize Aircraft Service Life
Original Design Tested/Validated		Projected Usage		Extended Service Life
DADTA		Data Gathering (Feedback)		Maximize Mission Capability
Usage To Date		KEY: - Forms - Oscar - Less		Minimize Un-Scheduled Downtime

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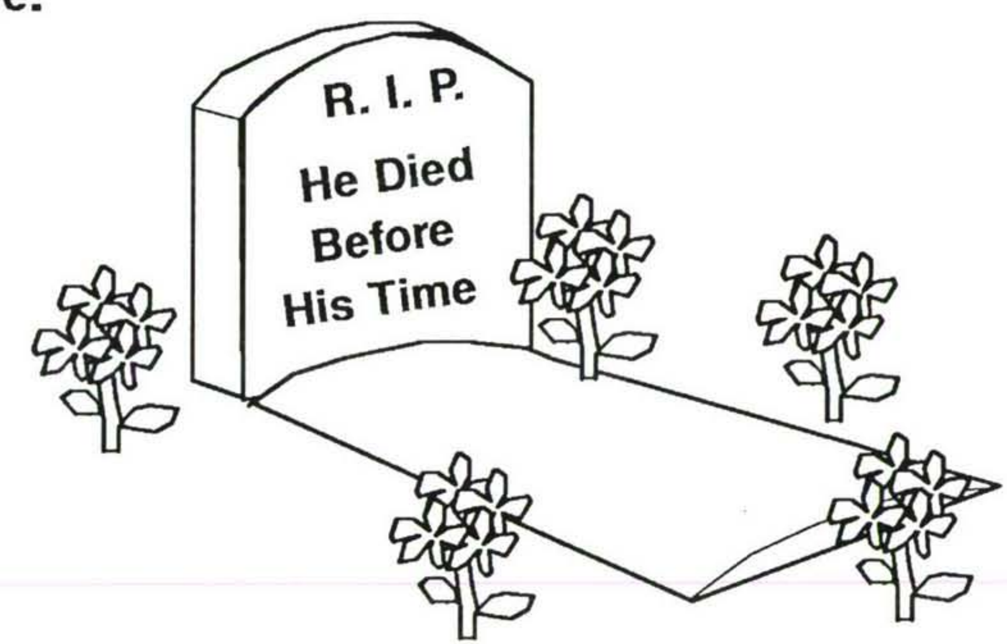
11-16-89

SUMMARY (Cont'd.)

What we know about the aircraft plus how we use it and monitor its health determines how we can best manage it and optimize extended usage. It has already been said that the SPM is the key to optimum force management, but the key to determining aircraft health is data <u>feedback</u>.

Force Management Technology Application To Aging Aircraft

Alternative:



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11-16-89

ALTERNATIVE

The alternative is to permit the aircraft to be forced into phaseout because of corrosion or lack of pro-active Force Management.

A-7D WING ASIP RE-EVALUATION

by

J. M. SMITH

PRESENTED TO:

1989 USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE

SAN ANTONIO, TEXAS

5-7 DECEMBER 1989

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A-7D WING ASIP RE-EVALUATION

- BACKGROUND
  - A-7 CORSAIR II - ORIGINAL DESIGN
  - AIR FORCE A-7D DEVELOPMENT
  - AIR FORCE ASIP
- SERVICE HISTORY
  - FLIGHT RECORD
  - INSPECTION HISTORY
- CURRENT EVENTS
  - 2ND INTERMEDIATE SPAR INCIDENT
  - REAR SPAR INCIDENT
  - FLEET INSPECTIONS
- ASIP RE-EVALUATION
  - AIRCRAFT USAGE ASSESSMENT
  - FLIGHT STRAIN SURVEY
  - INCIDENT AIRCRAFT FRACTOGRAHIC RESULTS
  - WING TEARDOWN INSPECTION/CORRELATION
  - STRESS SPECTRA UPDATE
  - TEST CORRELATION/CRACK GROWTH ANALYSIS RESULTS
- CONCLUSIONS

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A-7D WING ASIP RE-EVALUATION

BACKGROUND

- A-7 CORSAIR II
  - DESIGNED AS CARRIER BASED LIGHT ATTACK AIRPLANE FOR NAVY
  - FIRST FLIGHT - SEPTEMBER 27, 1965
  - TOTAL PRODUCED (NAVY, AIR FORCE, FOREIGN) - 1545
  - MODELS:
    - NAVY - A-7A, B, C, E - 991 PRODUCED
    - AIR FORCE - A-7D - 489 PRODUCED
    - GREEK - A-7H - 65 PRODUCED
    - PORTUGUESE - A-7P, TA-7P - 50 CONVERTED FROM A-7A
  - WING DESIGNED AND TESTED TO MIL-A-8866 A BLOCK SPECTRUM - 8900 FLT HRS

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A-7D WING ASIP RE-EVALUATION

BACKGROUND

- AIR FORCE A-7D
    - DERIVATIVE OF A-7A/B
    - ROLLS ROYCE TF-41 ENGINE
    - BASICALLY SAME STRUCTURE
    - NAVY LOADS & SPECTRA
    - SERVICE LIFE OF 4000 FLT HRS

- AIR FORCE ASIP PROGRAM - 1975-76
    - DEVELOPED A-7D FLIGHT-BY-FLIGHT STRESS SPECTRA
        - BASED ON FLIGHT SURVEY DATA
            - 1256 FLT HRS
            - 9 AIRCRAFT
            - 2 BASES
    - COMPLETE ANALYTICAL REVIEW OF AIRCRAFT
        - DAMAGE TOLERANCE CRITERIA DEFINED AS .05 INCH CORNER CRACK AT CRITICAL LOCATION PER MIL-A-83444
        - ANALYZED TO MIL-STD-1530 REQMTS

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A-7D WING ASIP RE-EVALUATION

BACKGROUND

- ESTABLISHED A-7D ECONOMIC LIFE OF 8000 FLT HRS WITH MINOR ENHANCEMENTS AND ROUTINE MAINTENANCE
- ESTABLISHED A-7D INSPECTION INTERVALS AT 2000 FLT HRS AT WING STATION 24.6
- TRACKING PROGRAM ESTABLISHED

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A-7D WING ASIP RE-EVALUATION

SERVICE HISTORY

- TOTAL FLIGHT HOURS THRU 1988 (ALL MODELS) -

  > 5,000,000 FLT HRS

  - A-7D 1,452,843+ FLT HRS
  - A-7A,B,C,E 3,452,590+ FLT HRS
  - OTHER 100,000 FLT HRS

- INSPECTION HISTORY
  - GENERALLY CRACK FREE PRIOR TO DEC 1988
  - NO MAJOR WING STRUCTURAL FAILURES

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A-7D WING ASIP RE-EVALUATION

CURRENT EVENTS

- A-7D 71-363 WING FAILURE
  - 28 DEC 88 - MELROSE BOMBING RANGE
  - R/H WING DEPARTED AIRCRAFT
  - SUBSEQUENT INVESTIGATION REVEALED FATIGUE CRACKING AT SECOND INTERMEDIATE SPAR
  - NOT PREVIOUSLY IDENTIFIED AS CRITICAL LOCATION

- A-7D 70-990 13 INCH CRACK FROM YW 53.7 REAR SPAR FWD TO 5TH INTERMEDIATE SPAR
  - LAB ANALYSIS CONFIRMED FATIGUE FAILURE
  - KNOWN CRITICAL AREA - PART OF ASIP INSPECTION - LOCATION OF FATIGUE TEST FAILURE

- FLEET INSPECTIONS
  - AIR FORCE AND NAVY INSPECTIONS INITIATED
  - SURFACE EDDY CURRENT - TOTAL OF 126 HOLES/ AIRPLANE
  - IN HOLE INSPECTION OF 44 HOLES/AIRPLANE

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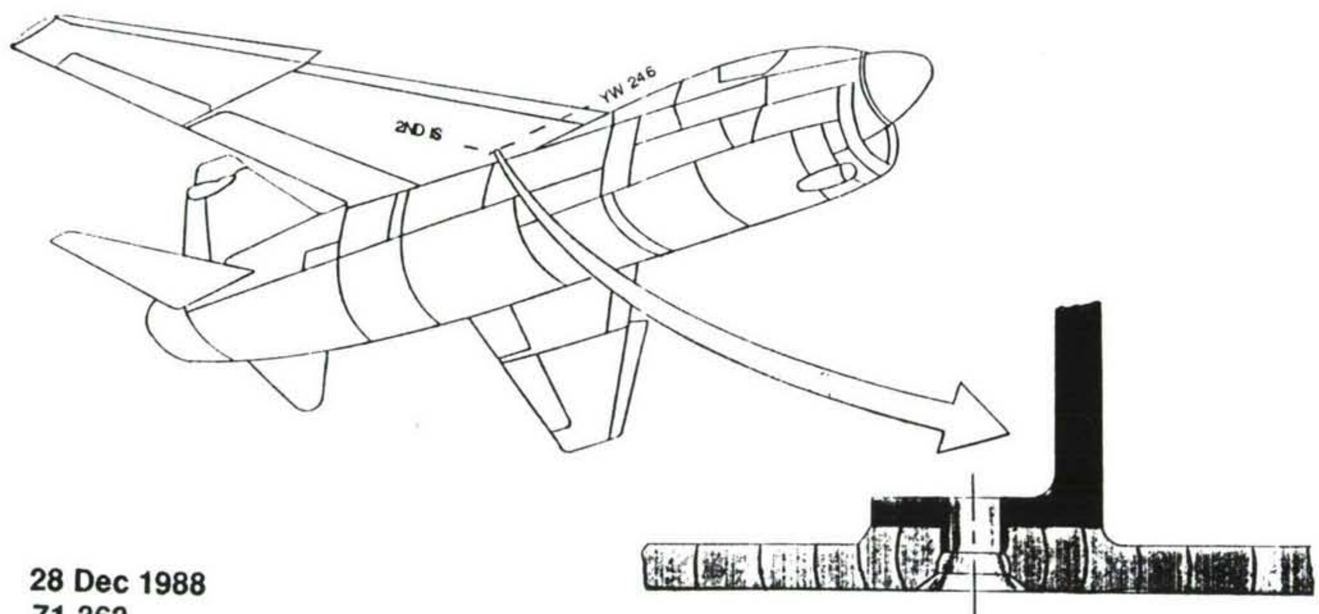
A-7D WING ASIP RE-EVALUATION

AIRCRAFT INCIDENT BACKGROUND

ALBUQUERQUE INCIDENT

FAILURE OF RIGHT HAND WING



DATE:	28 Dec 1988
A/C:	71-363
WING:	RH
LOCATION:	2nd Inter. Spar at YW 24.6
OBSERVATION:	Failure at ~ 4100 FH Previous Non-Critical Location, Evidence of Prior Fuel Leakage

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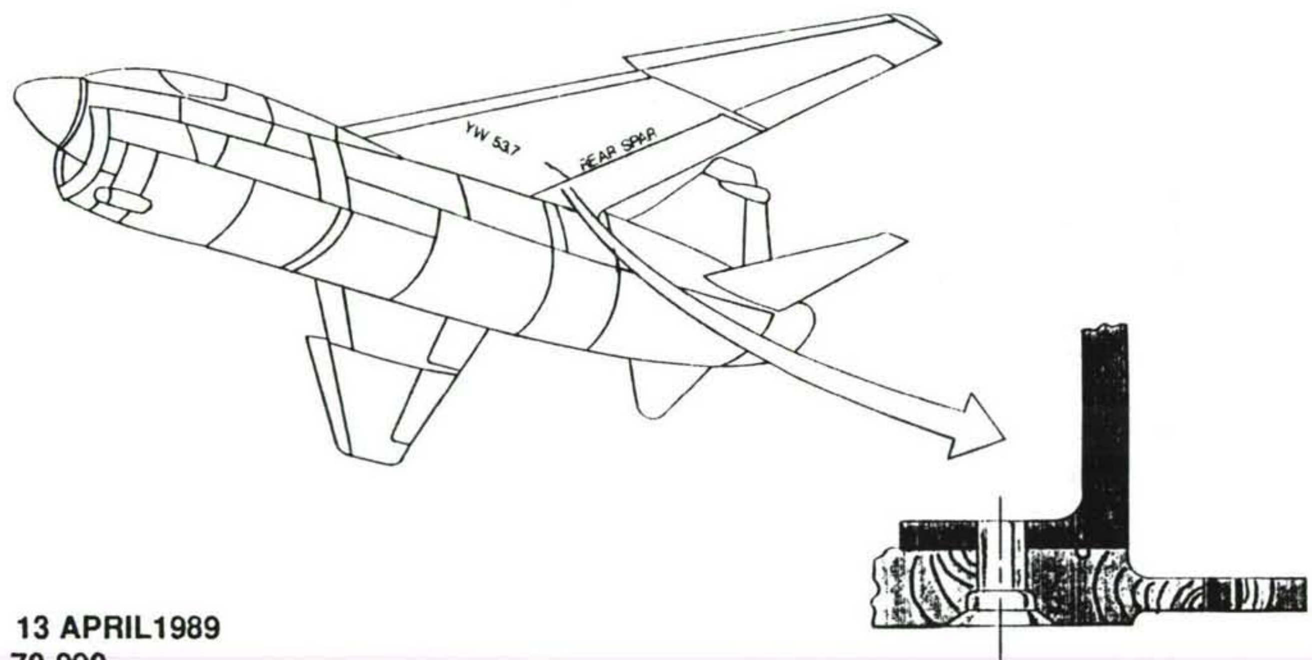
A-7D WING ASIP RE-EVALUATION

AIRCRAFT INCIDENT BACKGROUND (Cont.)

TUCSON INCIDENT

13 INCH CRACK



DATE:	13 APRIL1989
A/C:	70-990
WING:	LH
LOCATION:	Rear Spar at YW 53.7
OBSERVATION:	13 Inch Crack at ~ 4400 FH From Known Critical Location

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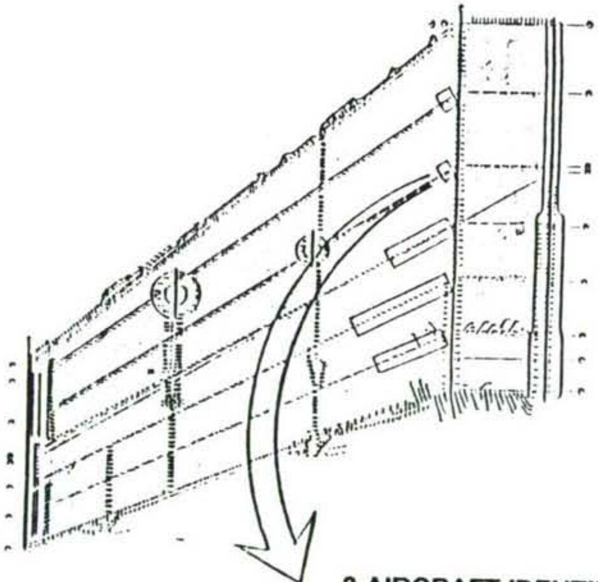
A-7D WING ASIP RE-EVALUATION

FLEET INSPECTIONS

INSPECTIONS DEFINED FOR ALL END FASTENERS AND ASIP LOCATIONS

SURFACE EDDY CURRENT

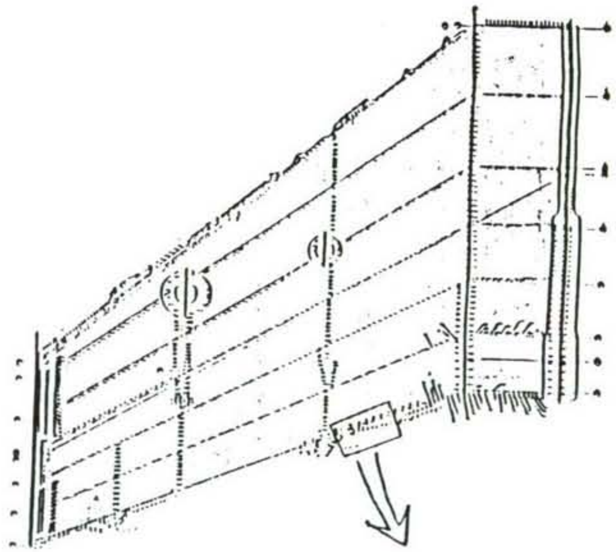
FOR LOCATIONS WITH CRITICAL FLAW SIZE BEYOND FASTENER HEAD (126 HOLES/AIRPLANE)



3 AIRCRAFT IDENTIFIED AS REQUIRING SKIN REPLACEMENT

IN-HOLE EDDY CURRENT

FOR LOCATIONS WITH CRITICAL FLAW SIZE UNDER FASTENER HEAD (44 HOLES/AIRPLANE)



FIVE AIRCRAFT IDENTIFIED AS REQUIRING SKIN REPLACEMENT

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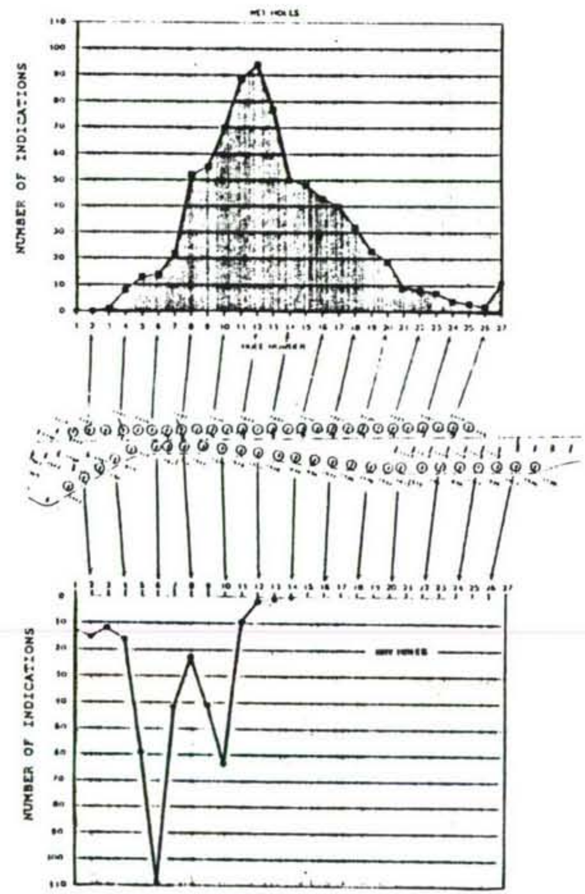
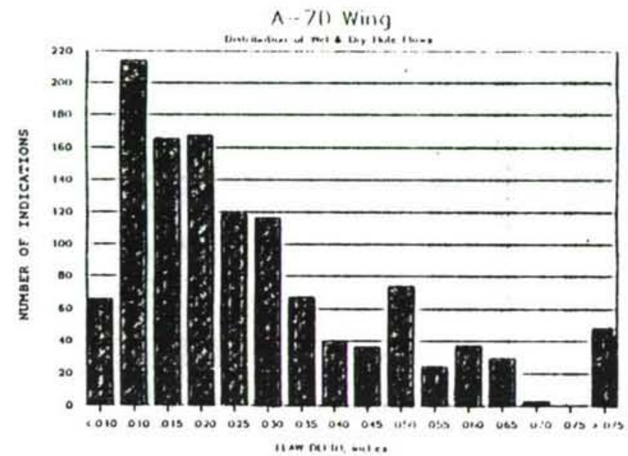
A-7D WING ASIP RE-EVALUATION

FLEET INSPECTIONS (CONT'D)

RESULTS OF IN-HOLE EDDY CURRENT INSPECTION

A-7D AIRCRAFT





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A-7D WING ASIP RE-EVALUATION

CURRENT RESULTS

- AIR FORCE (A-7D/K'S PER TCTOS 608/609/610)
    - ALL (364) AIRPLANES INSPECTED
        - 352 RETURNED TO FULL OPERATIONAL STATUS
        - 4 AWAITING REPAIR
        - 8 AWAITING SKIN REPLACEMENT

- NAVY (A-7E/TA-7C'S PER AFB 255)
    - ALL AIRPLANES INSPECTED
        - NO FLAWS DETECTED

- PORTUGAL (A-7P/TA-7P'S PER AFB 255)
    - ALL AIRPLANES INSPECTED
        - ONE POSSIBLE FLAW INDICATED

- GREECE
    - NONE TO DATE

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A-7D WING ASIP RE-EVALUATION

PROGRAM ELEMENTS

- AIRCRAFT USAGE ASSESSMENT
- FLIGHT STRAIN SURVEY
- INCIDENT AIRCRAFT FRACTOGRAPHY RESULTS
- WING TEARDOWN
- STRESS SPECTRA UPDATE
- COUPON TEST CORRELATION/CRACK GROWTH ANALYSIS

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A-7D WING ASIP RE-EVALUATION

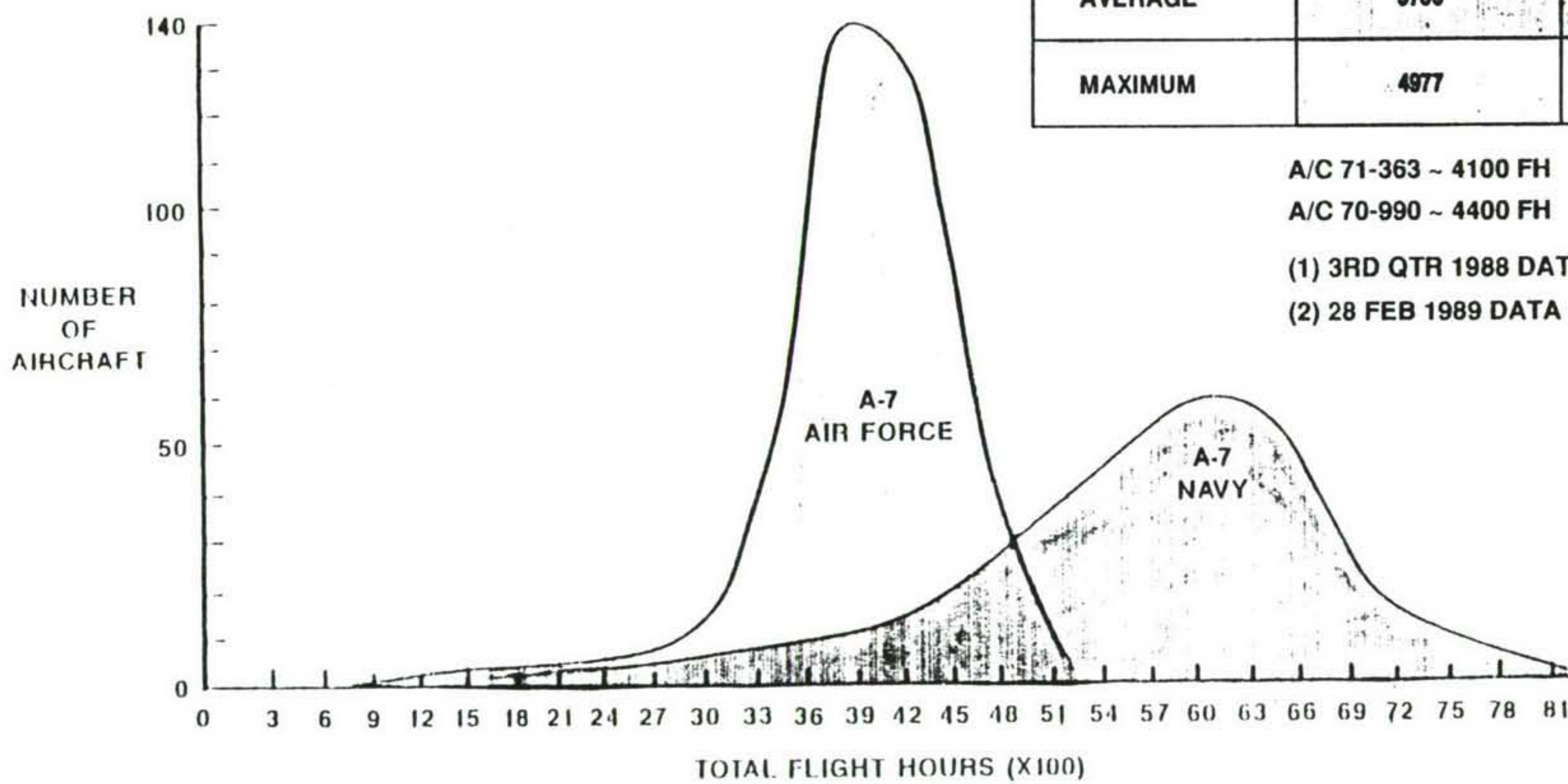
AIRCRAFT USAGE DATA

A-7 FLIGHT HOUR DISTRIBUTION

	AIR FORCE (1)	NAVY (2)
AVERAGE	3783	5499
MAXIMUM	4977	7782

A/C 71-363 ~ 4100 FH
A/C 70-990 ~ 4400 FH

(1) 3RD QTR 1988 DATA
(2) 28 FEB 1989 DATA



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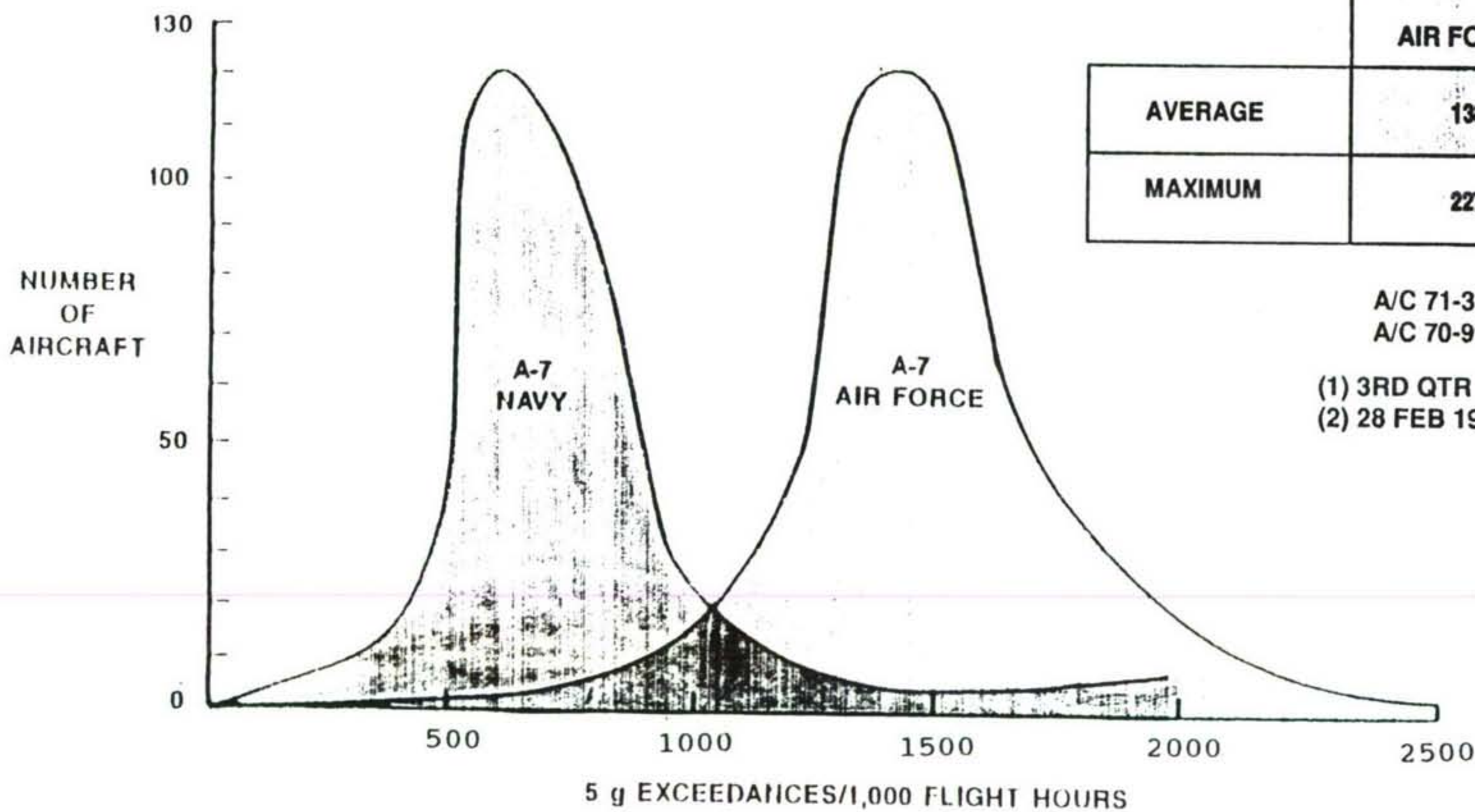
A-7D WING ASIP RE-EVALUATION

AIRCRAFT USAGE DATA (CONT.)

A-7 COUNTING ACCELEROMETER DATA

5g EXCEEDANCES PER 1,000 FH



	AIR FORCE (1)	NAVY (2)
AVERAGE	1380	639
MAXIMUM	2275	1299

A/C 71-363: 1826
A/C 70-990: 1706

(1) 3RD QTR 1988 DATA
(2) 28 FEB 1989 DATA

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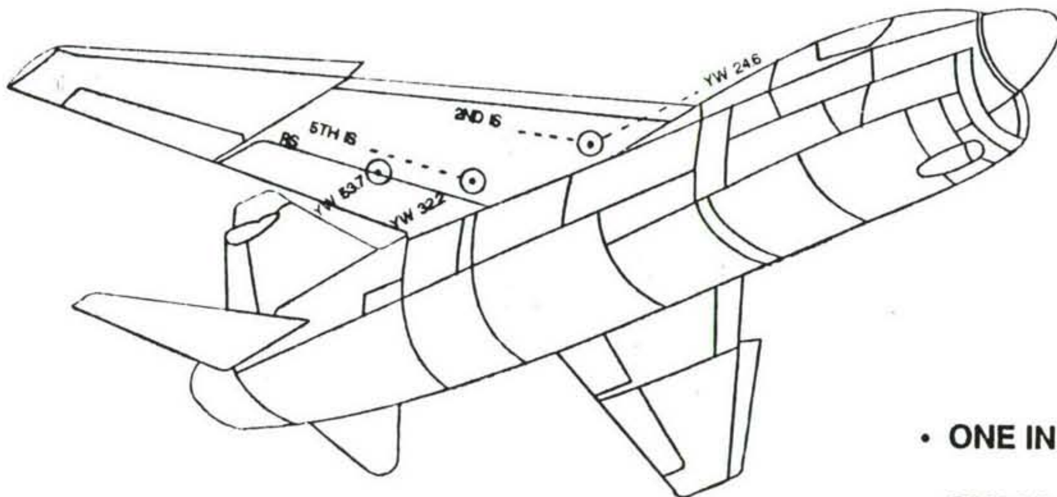
A-7D WING ASIP RE-EVALUATION

FLIGHT STRAIN SURVEY



- ONE INSTRUMENTED A-7E AIRCRAFT
- STRAIN GAGES AT BOTH INCIDENT LOCATIONS AND A-7D TRACKING LOCATIONS.
- GAGES MOUNTED ON BOTH WINGS.
- MANEUVERS AT VARYING MACH - ALTITUDE - STORE CONFIGURATIONS
  - SYMMETRIC PUSHOVER/PULL-UPS
  - UNSYMMETRIC (RPO'S, LFR)
  - DIVE TOSS, LATERAL TOSS
  - JINKING
  - ACCELERATED DEPARTURES

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A-7D WING ASIP RE-EVALUATION

A-7E STRAIN SURVEY

- SURVEY RESULTS IMPACTING STRESS SPECTRUM
    - STRAIN DATA AIDED CORRELATION OF UPDATED WING FEM
        - FEM PROVIDES STRESS TRANSFER FUNCTIONS
        - 17% HIGHER TRANSFER FUNCTION AT 2ND INTERMEDIATE SPAR
        - 13% HIGHER TRANSFER FUNCTION AT REAR SPAR
    - IDENTIFIED BUFFET INDUCED CYCLES

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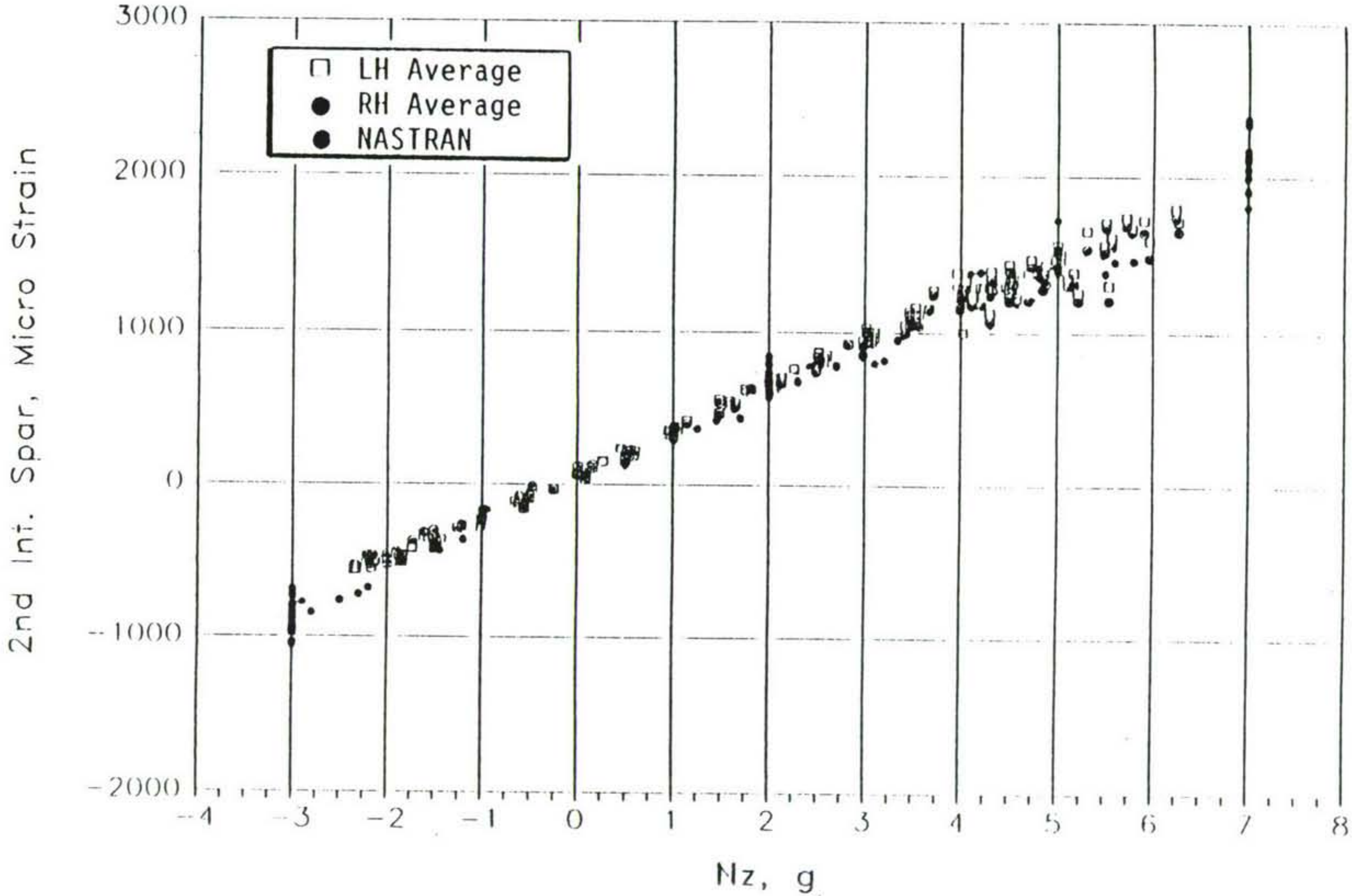
A-7D WING ASIP RE-EVALUATION

WING SKIN STRAIN - SYMM PUSH/PULL



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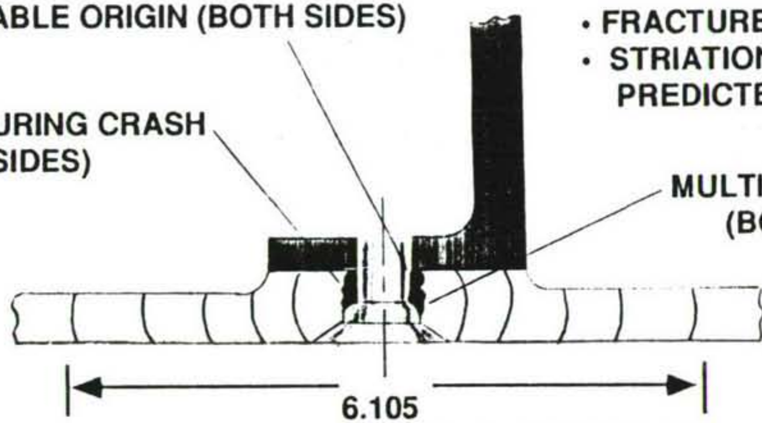
A-7D WING ASIP RE-EVALUATION

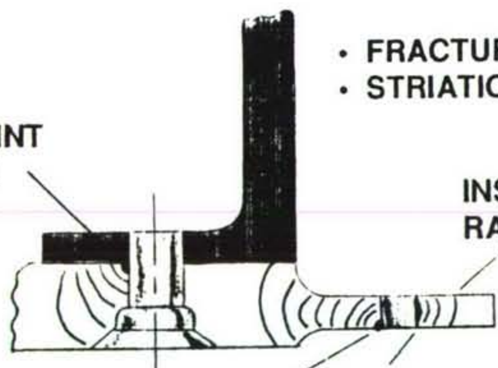
INCIDENT AIRCRAFT FRACTOGRAPHY RESULTS

ALBUQUERQUE INCIDENT



- FRACTURE SURFACE WELL DEFINED
- STRIATION COUNT AGREES WITH PREDICTED LOAD SPECTRUM

TUCSON INCIDENT



- FRACTURE SURFACE IN POOR CONDITION
- STRIATION COUNT INCONCLUSIVE

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A-7D WING ASIP RE-EVALUATION

WING TEARDOWN

- SIX WING SKINS TORN DOWN
  - TWO A-7D SKINS @ LTV = 4000 FH EACH
  - TWO A-7E SKINS @ LTV = 5930 FH EACH
  - TWO A-7E SKINS @ NRL = 7243 FH EACH

- MATERIAL PROPERTIES VERIFIED (AIR FORCE TASK ONLY)
  - da/dn VS ΔK
  - KIC
  - FTY, FTU, %e
  - CONDUCTIVITY

- IN-HOLE EDDY CURRENT INSPECTIONS
  - ALL HOLES IN SKIN
  - ALL HOLES IN SPAR CAP (AIR FORCE TASK ONLY)

- 200 - 1600 HOLES BROKEN OPEN ON EACH SKIN
  - NUMBER OF HOLES WITH FLAWS DEFINED
  - FLAW DEPTH AND LENGTH DEFINED
  - HOLE QUALITY DEFINED

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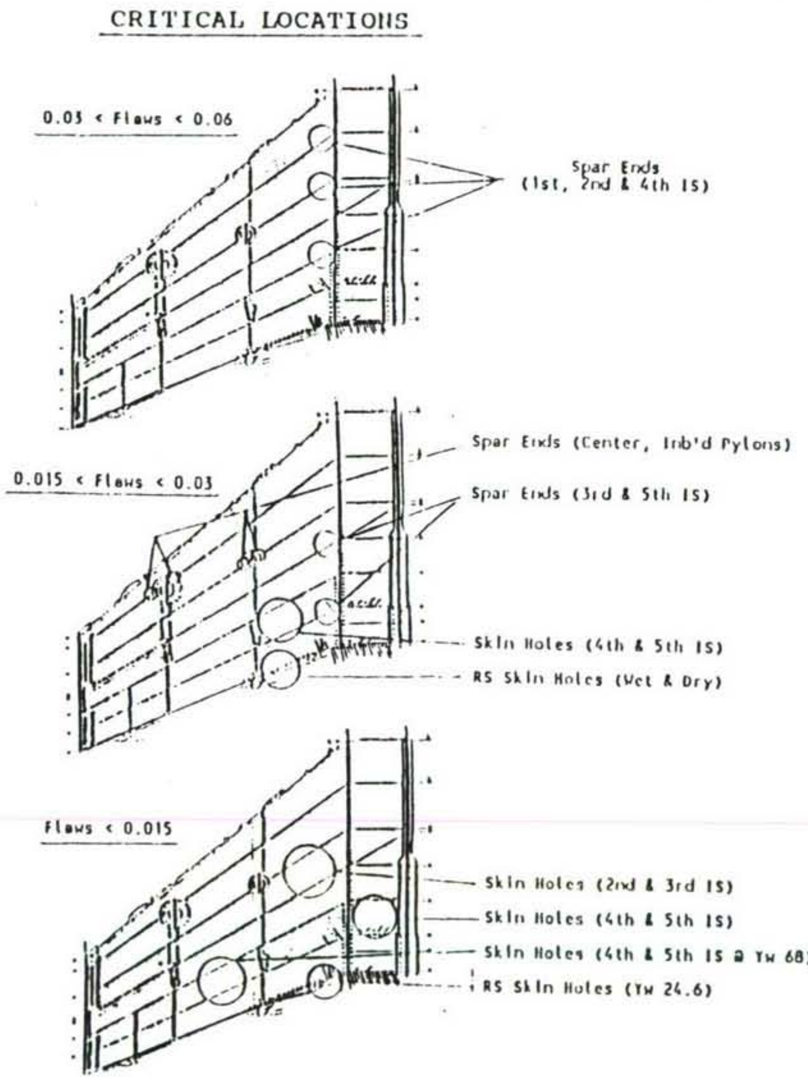
A-7D WING ASIP RE-EVALUATION

WING TEARDOWN



	LTV TEARDOWN	NRL TEARDOWN
HOLES BROKEN OPEN	1458	3200
HOLES W/FLAWS	277	461
FLAWS > .030	6/2.2%	8/1.7%
.015 < FLAWS < .030	7/2.6%	16/3.5%
.010 < FLAWS < .015	17/6.2%	15/3.3%
.005 < FLAWS < .010	38/13.9%	46/10.0%
FLAWS < .005	206/75.2%	380/83.0%
# OF HOLES W/FLAWS		
• @ 2 SIDES	183	250
• @ 1 SIDE	94	211
% HOLES W/FLAWS		
• FROM MECH DEFECTS	60%	---
• FROM ANODIZE/CORR PIT	40%	---

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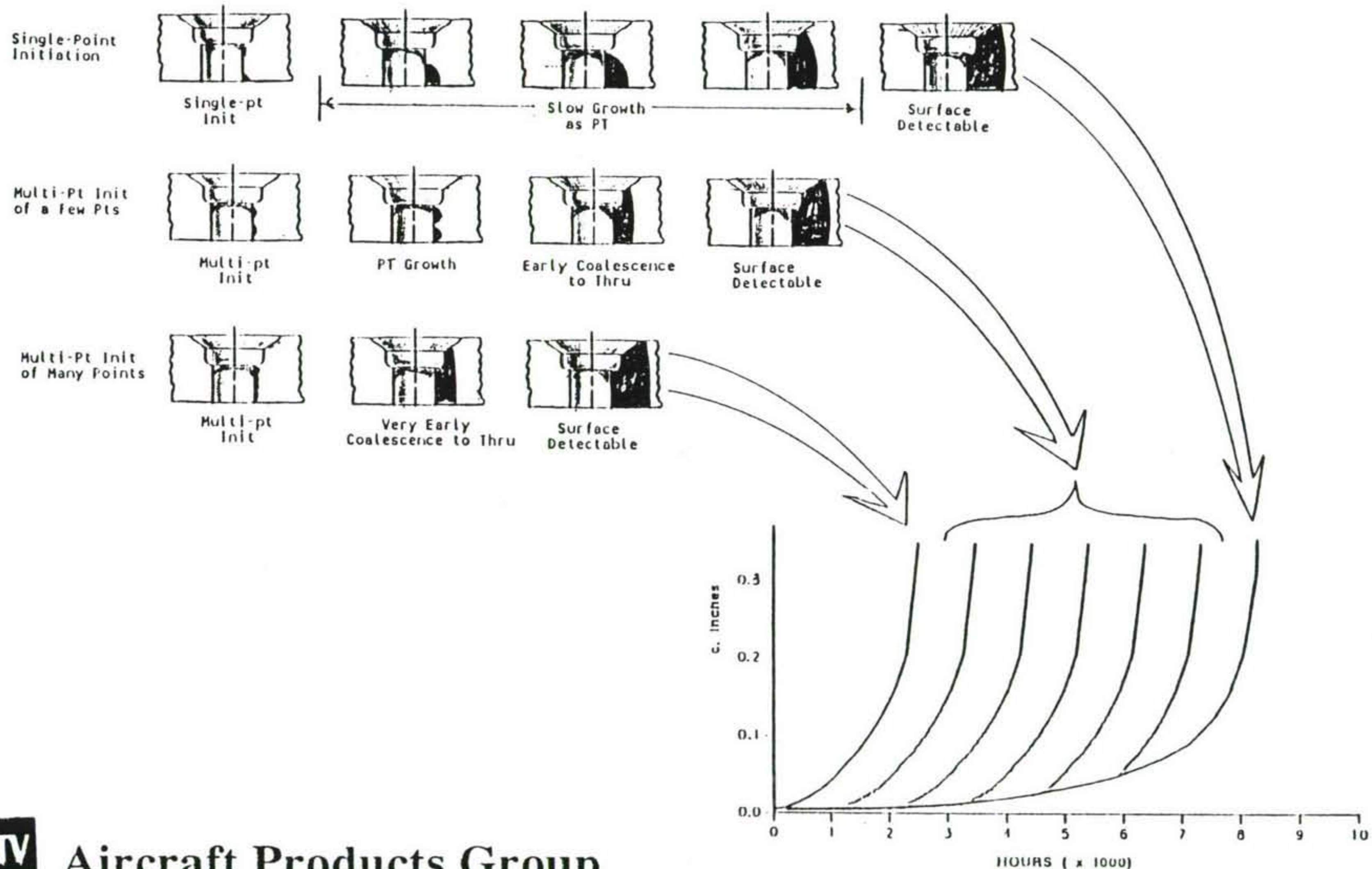
A-7D WING ASIP RE-EVALUATION

WING TEARDOWN (CONT.)

IDENTIFICATION OF CRACK TYPES



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A-7D WING ASIP RE-EVALUATION

A-7D WING INVESTIGATION

- STRESS SPECTRA DEVELOPMENT
    - A-7E FLIGHT STRAIN SURVEY
    - BUFFET
    - STRESS TRANSFER FUNCTIONS
    - UPDATED SPECTRA/SEQUENCES

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A-7D WING ASIP RE-EVALUATION

CHARACTERIZE AND INCORPORATE BUFFET INDUCED CYCLES

- A-7E FLIGHT STRAIN SURVEY
  - BUFFET INDUCED STRESS OSCILLATIONS RECORDED
    - DURING SOME PULL-UP MANEUVERS (HIGH AOA)
    - OBSERVED OSCILLATIONS AT 2ND, 5TH AND REAR SPAR
      - 7-8 HZ, WING 1ST BENDING FREQUENCY

- BUFFET
  - TURBULENT AIRFLOW RESULTING IN REPEATED SEPARATION AND REATTACHMENT OF AIRFLOW
    - CYCLIC LIFT ON WING
  - BUFFET ONSET IS FUNCTION OF AOA AND MACH NUMBER
  - AMF EXTENSION DELAYS BUFFET ONSET AT HIGH AOA

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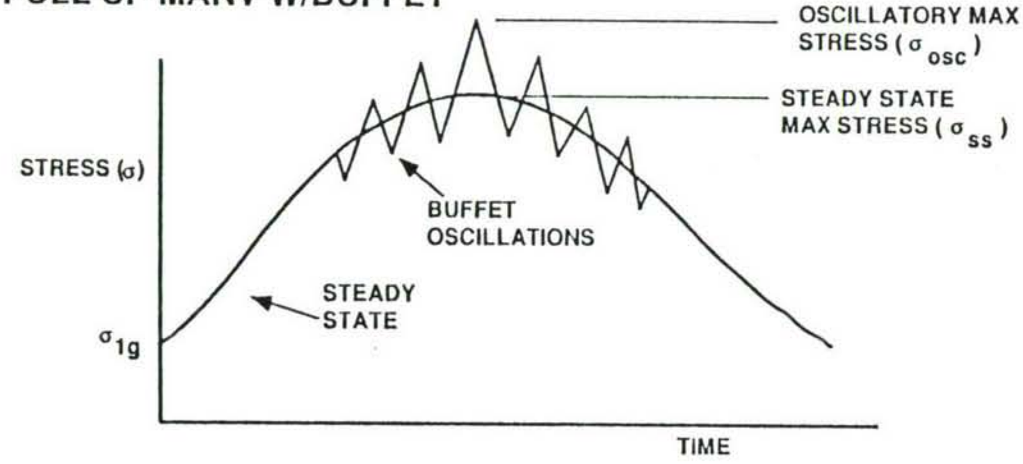
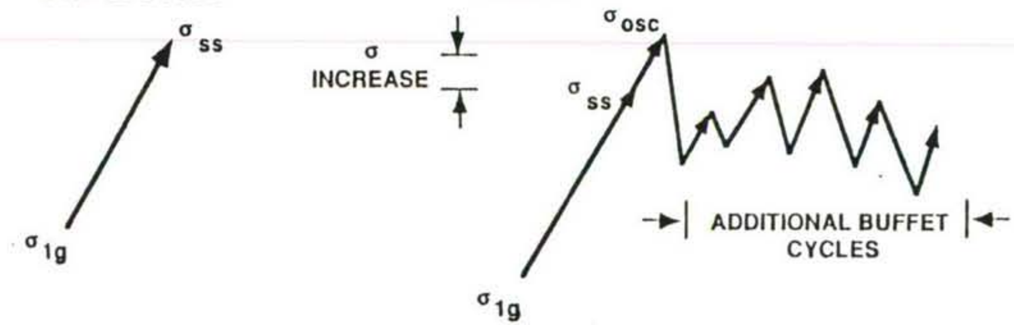
A-7D WING ASIP RE-EVALUATION

SCHEMATIC OF BUFFET INDUCED CYCLES

- PULL UP MANV W/BUFFET



- MANV REPRESENTATION



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A-7D WING ASIP RE-EVALUATION

A-7D STRESS TRANSFER FUNCTIONS

- FEM SOLUTION FOR 102 FLIGHT CONDITIONS
  - VARIATION IN STORE CONFIGURATION AND FUEL STATE
  - VARIATION IN MACH, ALTITUDE, WEIGHT

- CRITICAL LOCATION STRESSES FROM UPDATED/ CORRELATED FEM

- CRITICAL LOCATION STRESS RELATED TO DAMAGE TRACKING LOCATION (32SG) STRESS
  - LEAST SQUARES FIT OF FEM DATA
  - STRESS TRANSFER FUNCTIONS ARE OF FORM

$$\sigma_{LOCI} = TI * \sigma_{32SG}$$

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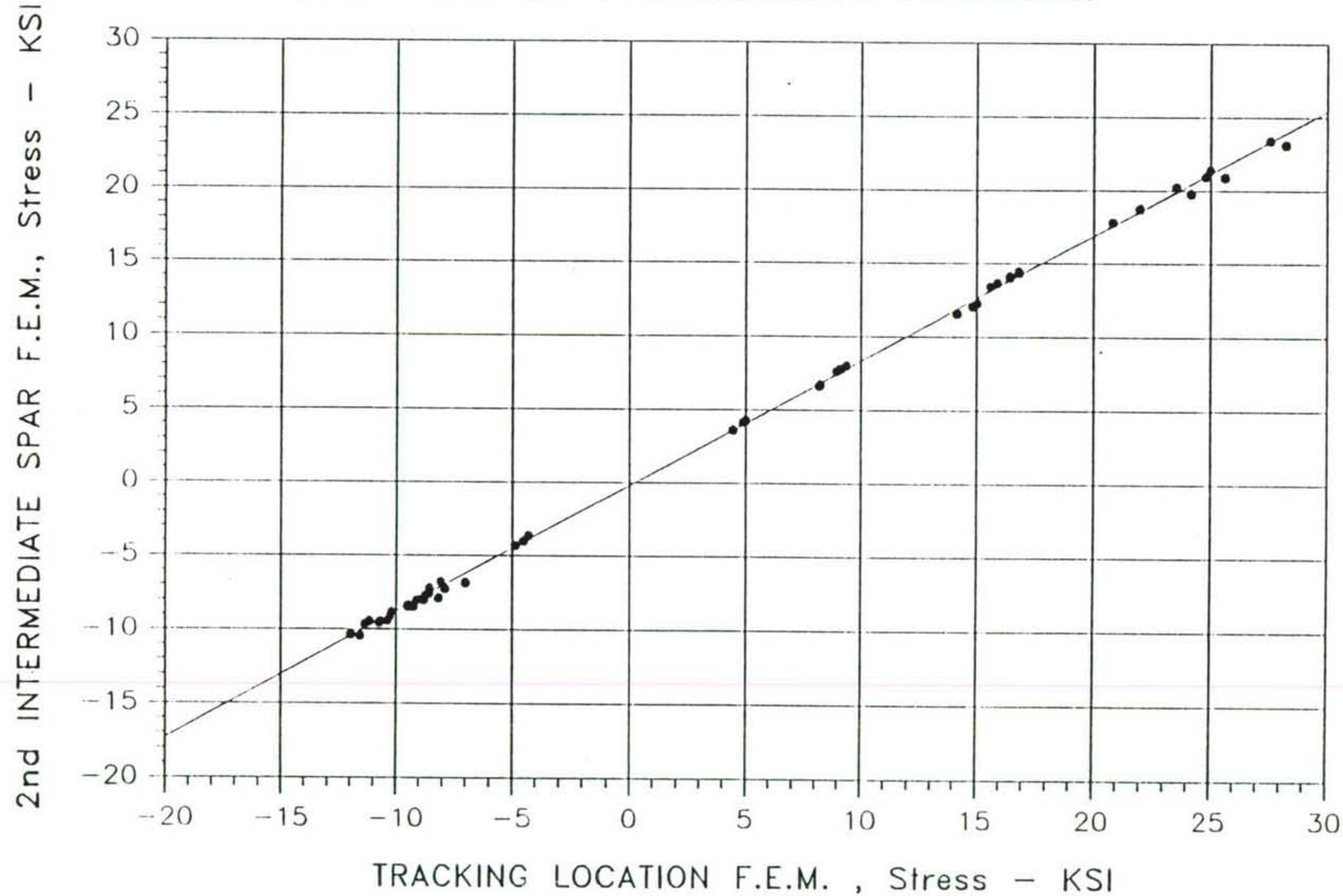
A-7D WING ASIP RE-EVALUATION

A-7D FINITE ELEMENT MODEL



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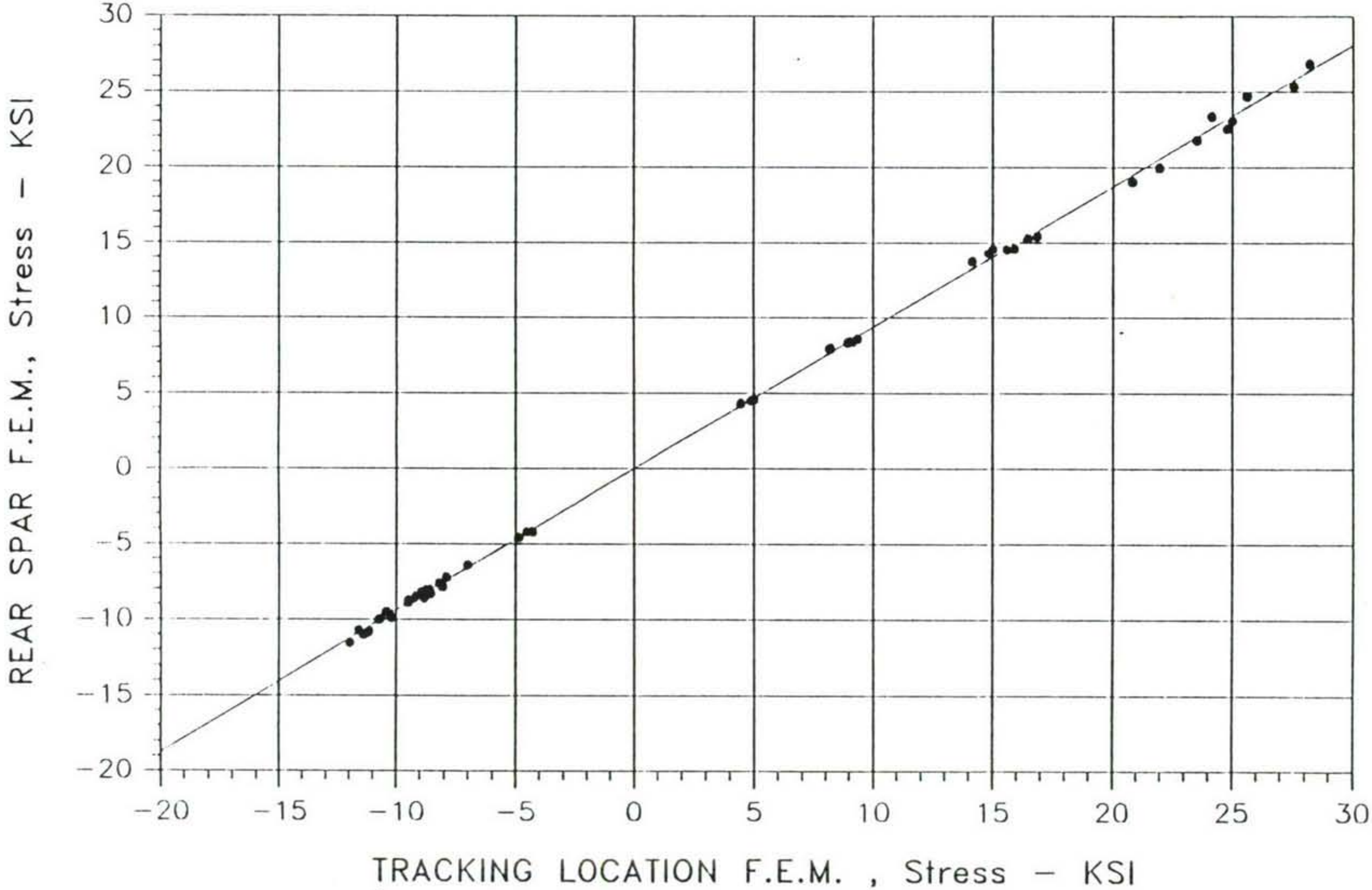
A-7D WING ASIP RE-EVALUATION

A-7D FINITE ELEMENT MODEL



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A-7D WING ASIP RE-EVALUATION

UPDATED A-7D STRESS SPECTRA/SEQUENCES

- TWO INCIDENT AIRCRAFT - TAILORED USAGE
  - BASELINE RECORDED Nz/STRESS SPECTRUM AT TRACKING LOCATIONS
    - TAILORED Nz SPECTRUM TO MATCH INCIDENT A/C CA Nz COUNTS
    - RESULTS IN TAILORED MANEUVER STRESS SPECTRUM
  - ADD/MODIFY STRESS CYCLES FOR BUFFET
  - ADD GUST STRESS CYCLES
    - MATCH A/C 363 FRACTOGRAPHIC STRIATION COUNT
  - STRESS TRANSFER FUNCTIONS FOR OTHER LOCATION SPECTRA/SEQUENCES

- FLEET AIRCRAFT - BASELINE USAGE
  - BASELINE RECORDED Nz/STRESS SPECTRUM AT TRACKING LOCATION
  - BUFFET AND GUST STRESS CYCLES
  - STRESS TRANSFER FUNCTIONS

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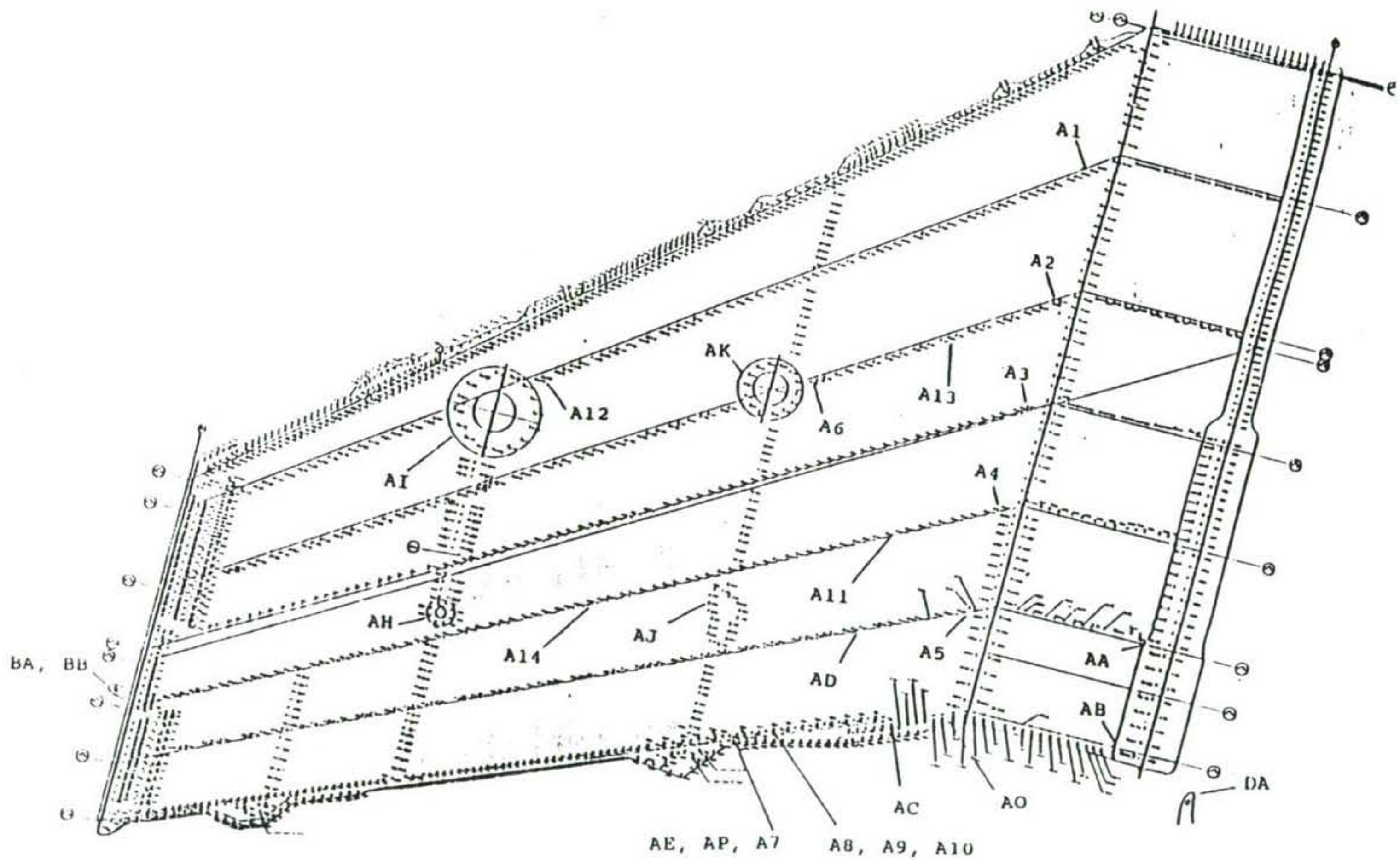
A-7D WING ASIP RE-EVALUATION

A-7D CRITICAL LOCATION SUMMARY



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A-7D WING ASIP RE-EVALUATION

LIST OF 14 CURRENT CRITICAL LOCATIONS

CODE	DESCRIPTION
AA	LWS, BL-0 @ 5TH INTERMEDIATE SPAR
AB	LWS, BL-0 @ REAR SPAR
AC	LWS, WS-24.6 @ REAR SPAR
AD	LWS, WS-32.2 @ 5TH INTERMEDIATE SPAR
AE	LWS, WS-53.7 @ REAR SPAR
AH	LWS, CENTER PYLON AFT STUB HOLE
AI	LWS, CENTER PYLON FWD POST HOLE
AJ	LWS, INB'D PYLON AFT STUB HOLE
AK	LWS, INB'D PYLON FWD POST HOLE
AO	BOOMERANG STRAP, WS-24.6 @ REAR SPAR
AP	LOWER CAP, WS-53.7 @ REAR SPAR
BA	LUG BASE, WINGFOLD RIB (OWP)
BB	LUG HOLE, WINGFOLD RIB (OWP)
DA	FS480 BULKHEAD WING AFT ATTACH LUG

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A-7D WING ASIP RE-EVALUATION

LIST OF 14 NEW CRITICAL LOCATIONS

CODE	DESCRIPTION
A1	LWS, WS-24.6 @ 1ST INTERMEDIATE SPAR
A2	LWS, WS-24.6 @ 2ND INTERMEDIATE SPAR
A3	LWS, WS-24.6 @ 3RD INTERMEDIATE SPAR
A4	LWS, WS-24.6 @ 4TH INTERMEDIATE SPAR
A5	LWS, WS-24.6 @ 5TH INTERMEDIATE SPAR
A6	LWS, PYLON @ 2ND INTERMEDIATE SPAR
A7	LWS, TAB AREA @ WS-53.7
A8	LWS, WET AREA @ WS-50
A9	LWS, TAB AREA @ WS-50
A10	REAR SPAR @ WS-50
A11	LWS, WS-32.2 @ 4TH INTERMEDIATE SPAR
A12	LWS, PYLON @ 1ST INTERMEDIATE SPAR
A13	LWS, WS-32 @ 2ND INTERMEDIATE SPAR
A14	LWS, WS-68 @ 4TH INTERMEDIATE SPAR

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A-7D WING ASIP RE-EVALUATION

SUMMARY OF 2ND INTERMEDIATE SPAR ELEMENT TEST

CONFIGURATION	SPECTRUM	STRESS LEVEL	LIFE (FH)	COMMENTS
NOMINAL	BASELINE	85%	44000	ORIGINAL STRESS LEVEL & USAGE; VERY LARGE LIFE
NOMINAL	TAILORED	100%	14500 13500	STRESS INCREASED TO MATCH UPDATED NASTRAN AND FLT TEST LIFE TOO LARGE, BUT SIGNIFICANT REDUCTION
NOMINAL	TAILORED	110%	12000	STRESS INCREASED BY 10% BUT NO SIGNIFICANT REDUCTION IN LIFE
NOMINAL WITH 120# TORQUE	TAILORED	100%	25000 32000	SHORTER FASTENERS USED WHICH PROVIDE CLAMP-UP SIGNIFICANT INCREASE IN LIFE (FACTOR OF 2)
NOMINAL SS	TAILORED	100%	13500	SIMULATED SCRATCH ADDED; SINGLE POINT PART THRU GROWTH; NO CHANGE IN LIFE
NOMINAL MI SS MI SS	TAIL. W/BUFF	100%	4100 4600	BUFFET & SIMULATED SCRATCH ADDED; MULTI-PT INITIATION AND RAPID TRANS. TO THRU FLAW; SIGNIF. REDUCTION IN LIFE; INCIDENT A/P MATCHED
OVERSIZE HOLE OVERSIZE HOLE WITH NO FAST.	TAILORED TAIL. W/BUFF TAILORED TAIL. W/BUFF	100%	18250 14350 20250 12550	OVERSIZED HOLES TESTED WITH AND WITHOUT BUFFET; NO SIMULATED SCRATCH; APPROX. 23% REDUCTION IN LIFE DUR TO BUFFET OVERSIZED HOLES WITH NO FAST. TESTED WITH AND WITHOUT BUFFET; NO SIMULATED SCRATCH; APPROX. 38% REDUCTION IN LIFE DUE TO BUFFET
REPAIR	TAIL. W/BUFF	100%	NF NF	HALF INCH BUSHING REPAIR TESTED. NO FLAWS IN 1600 FH; 0.02 IN. PREFLAW ADDED; FAILURE IN 3RD HOLE (NON-BUSHED) AFTER 12-14000 ADD. FH

NOTES: NOMINAL CONFIGURATION CONTAINED FASTENERS TOO LONG TO OBTAIN CLAMP UP
SS - SIMULATED SCRATCH ADDED
MI - MULTI-INITIATION AND EARLY THRU FLAW
NF - NO FAILURE

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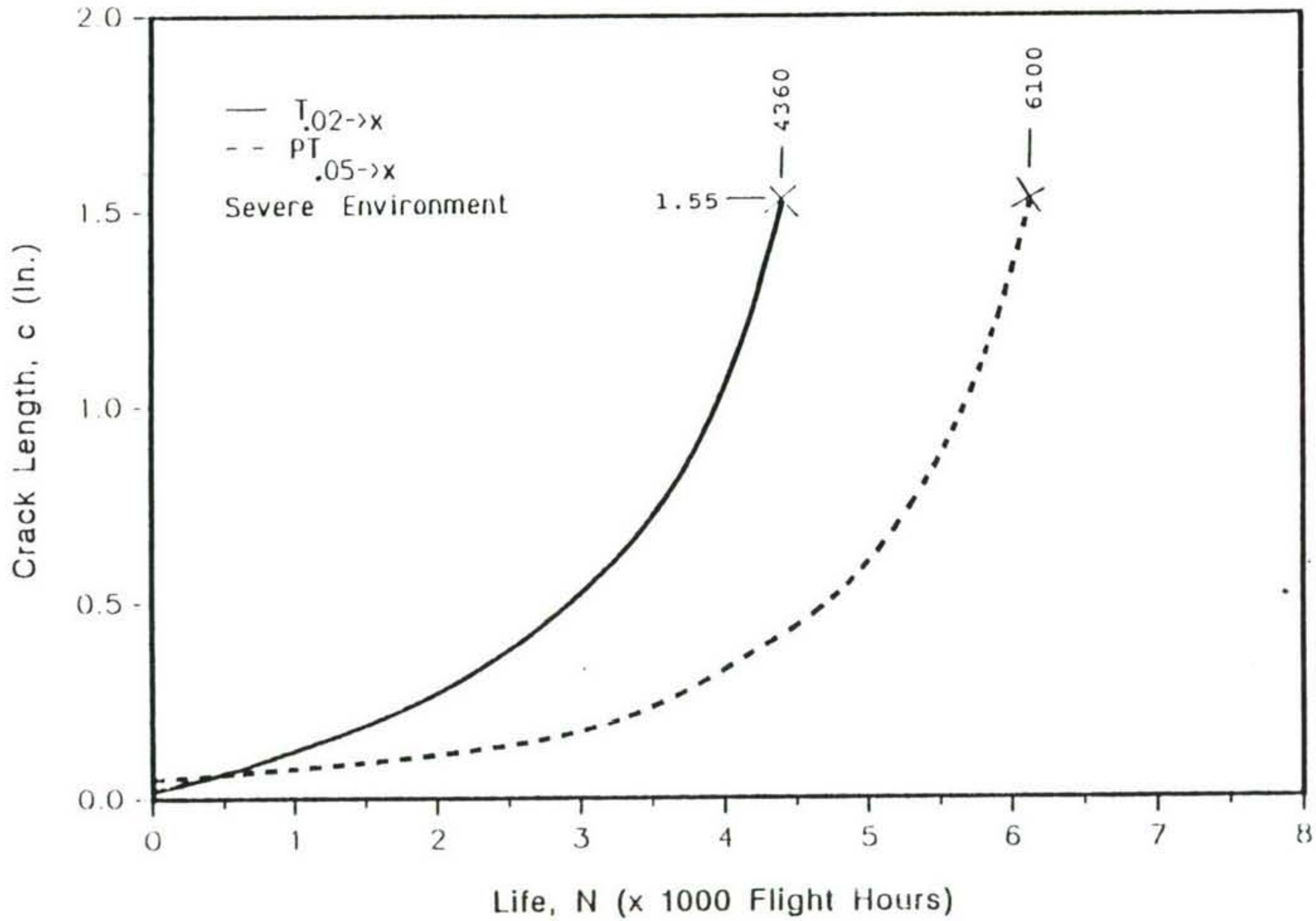
A-7D WING ASIP RE-EVALUATION

A-7D LOCATION A2 - DT

LWS @ WS24.6 2ND INTERM SPAR - SEV ENV



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A-7D WING ASIP RE-EVALUATION

SUMMARY OF FINDINGS

- ASIP CAN PROTECT AIRCRAFT SAFETY
- HIGH PROBABILITY OF FLAW GROWTH FROM BOTH SIDES OF FASTENER HOLE
- BUFFET CAN REDUCE CRACK GROWTH LIFE APPROXIMATELY 30% (ADDITIONAL FLIGHT TEST DATA REQUIRED TO COMPLETELY CHARACTERIZE BUFFET)
- A-7 INCIDENTS RESULTED FROM MULTI-POINT INITIATION PRODUCING THRU THICKNESS CRACK GROWTH
- FAILURE MECHANISM IDENTIFIED AND CORRELATED ANALYTICALLY/VERIFIED BY TEST
- CONTINUED A-7D OPERATION UNDER UPDATED ASIP RESULTS

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STRUCTURAL FATIGUE

RISK ASSESSMENTS

ELEMENTS AND EXAMPLE

1989 USAF Structural Integrity
Program Conference
5-7 Dec 1989, San Antonio, Texas

Ken Walker
Consultant
674 County Square Dr
Suite [illegible]
Ventura CA [illegible]
[illegible]

STRUCTURAL FATIGUE RISK ASSESSMENT:

A QUALITATIVE TOOL CAPABLE OF PROVIDING INSIGHT FOR STRUCTURAL FATIGUE RELATED FLEET MANAGEMENT

OBJECTIVE OF PRESENTATION:

ENCOURAGE USE OF STRUCTURAL FATIGUE RISK ASSESSMENTS BY:

- DESCRIBING PRINCIPAL ELEMENTS
- DISCUSSING CAPABILITIES AND LIMITATIONS
- RECOMMENDING APPLICATION STRATEGY
- DEMONSTRATING VALUE

(ALL IN 30 MINUTES)

THERE ARE THREE PRINCIPAL ELEMENTS:

> 1. ESTIMATES OF CRACK POPULATIONS

2. ESTIMATES OF INSPECTION PROGRAM EFFECTIVENESS

3. CRITERION FOR ACTION: SAFETY, ECONOMICS, FLEET READINESS

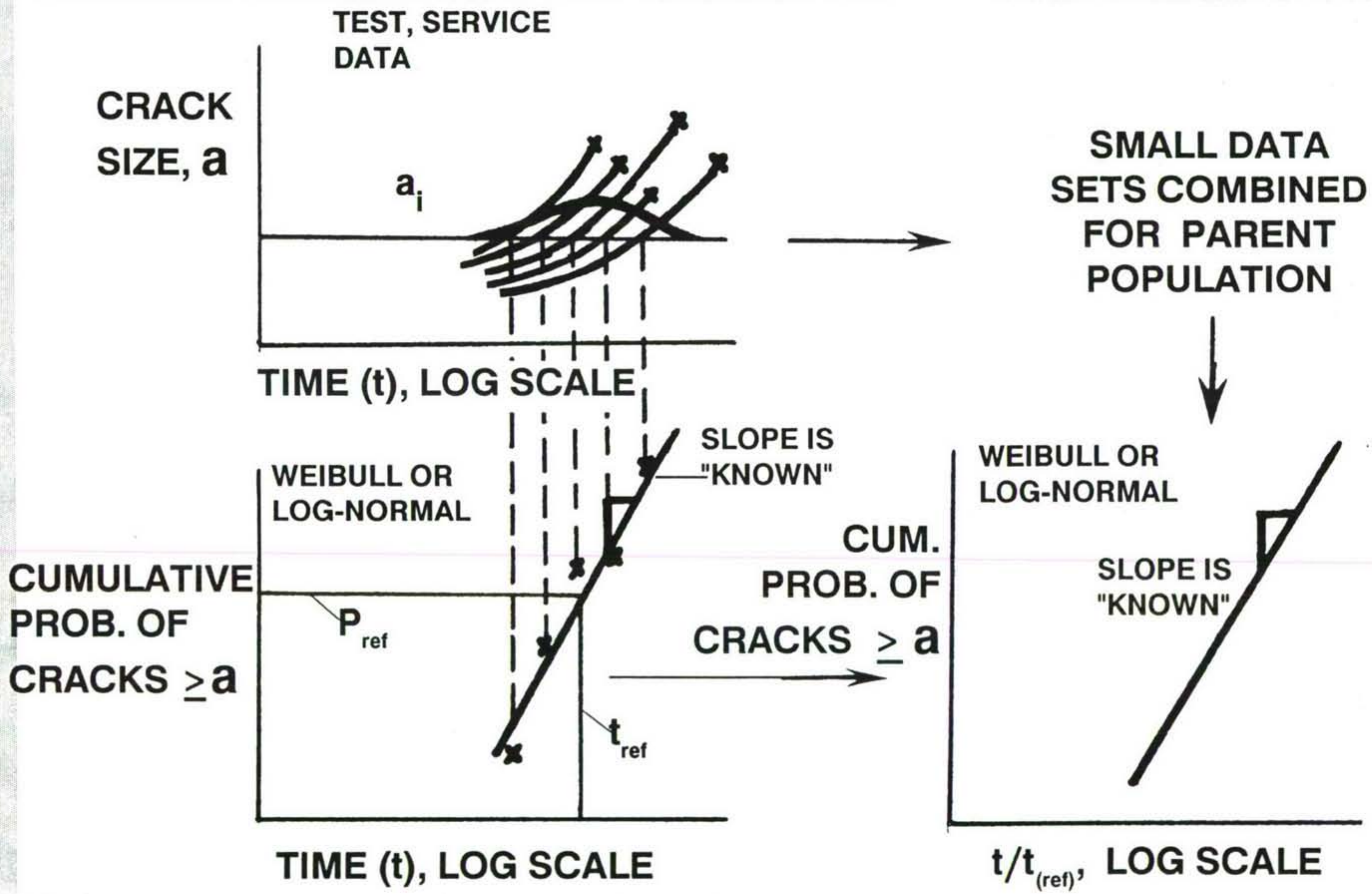
STRATEGY FOR CRACK POPULATION ESTIMATES:

1. DEVELOP "KNOWN" STATISTICAL PROPERTIES FROM HISTORIC DATA BASE

2. RELATE "KNOWN" STATISTICS TO CURRENT APPLICATION

3. ESTIMATE CRACK POPULATION AT TIME OF INSPECTION

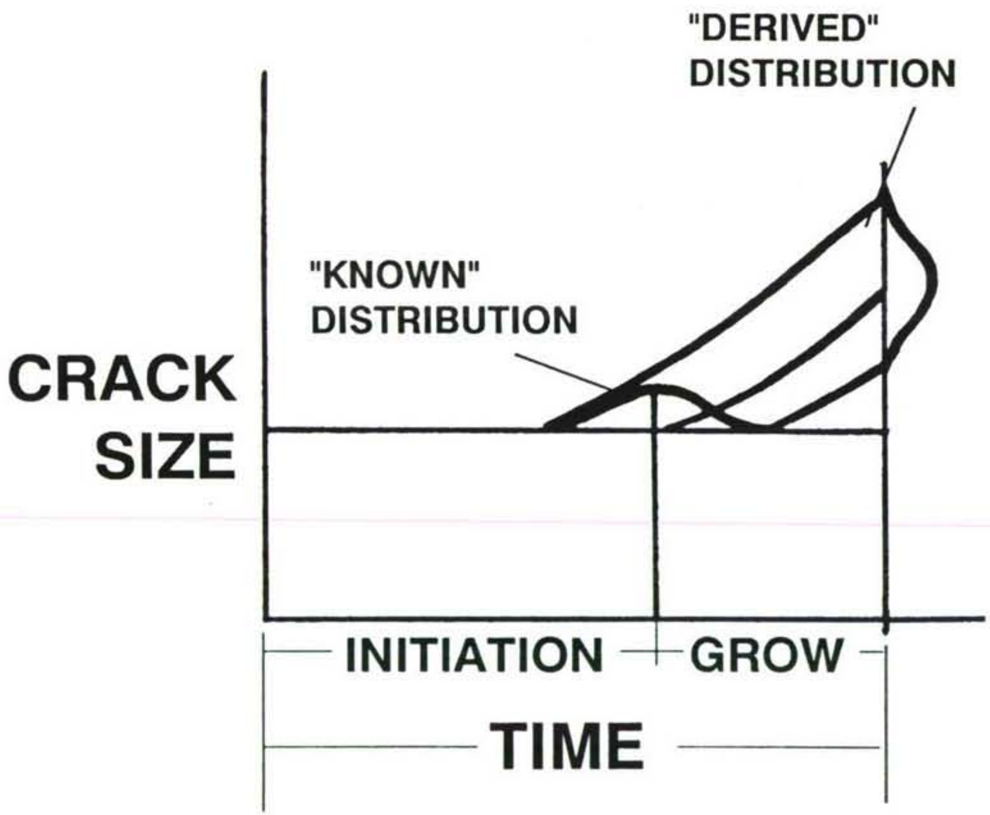
1) DEVELOP "KNOWN" STATISTICAL PROPERTIES



SAMPLE DATA BASE FOR "KNOWN" STATISTICAL PROPERTIES

REFERENCE	LOG-STANDARD DEVIATION	WEIBULL SHAPE PARAMETER	COMMENTS
1	0.14	4.0	2,000 DATA POINTS, 11,000 SPECIMENS, ALUMIINUM ALLOYS, LAB DATA
2	0.104	-----	120 AIRCRAFT COMPONENTS
3	----	4.5 SHORT LIFE 3.5 LONG LIFE	MANUFACTURER, EXTEN-SIVE SUBCOMPONENT AND ELEMENT DATA BASE
4	0.14 ADD 0.06 LOAD VAR.	-----	MANUFACTURER, SUPPLE-MENTAL INSPECTION RATIONALE (RISK ANAL)
5	0.15 AVG	-----	SERVICE EXPERIENCE 10 A/C COMPONENTS, 7 WING, 2 FUS, 1 FIN

2) RELATE "KNOWN" STATISTICS TO CURRENT APPLICATION



INITIATION

FATIGUE ANALYSIS

CRACK GROWTH FROM "EQUIVALENT INITIAL CRACKS"

TEST, SERVICE EXPERIENCE

GROWTH

LEFM ANALYSIS

TEST, SERVICE EXPERIENCE

AND ASSUME

SAME a vs N

DISTRIBUTION, "KNOWN", RANDOM

THERE ARE THREE PRINCIPAL ELEMENTS:

1. ESTIMATES OF CRACK POPULATIONS

\> 2. <u>ESTIMATES OF INSPECTION PROGRAM EFFECTIVENESS</u>

3. CRITERION FOR ACTION: SAFETY, ECONOMICS, FLEET READINESS

INSPECTION EFFECTIVENESS ELEMENTS

$$P_D = P_c \times P_d \times P_h$$

P_D = PROBABILITY OF DETECTING A CRACK OF SIZE **a** IN ONE INSPECTION

P_c = PROBABILITY OF A CRACK OF SIZE **a** BEING PRESENT

P_d = PROBABILITY OF DETECTING A CRACK OF SIZE **a** IF IT IS PRESENT AND THE INSPECTION IS PERFORMED AS INTENDED

P_h = PROBABILITY THAT THE INSPECTION WILL BE PERFORMED AS INTENDED

THE PROBABILITY OF DETECTION P_d HAS TWO ELEMENTS:

1. "KNOWN" STATISTICS BASED ON APPLICABLE DATA BASE

2. REFERENCE DETECTABLE SIZE BASED ON CONDITIONS OF INSPECTION i.e. LIGHT, PAINT, SEALANT, ETC.

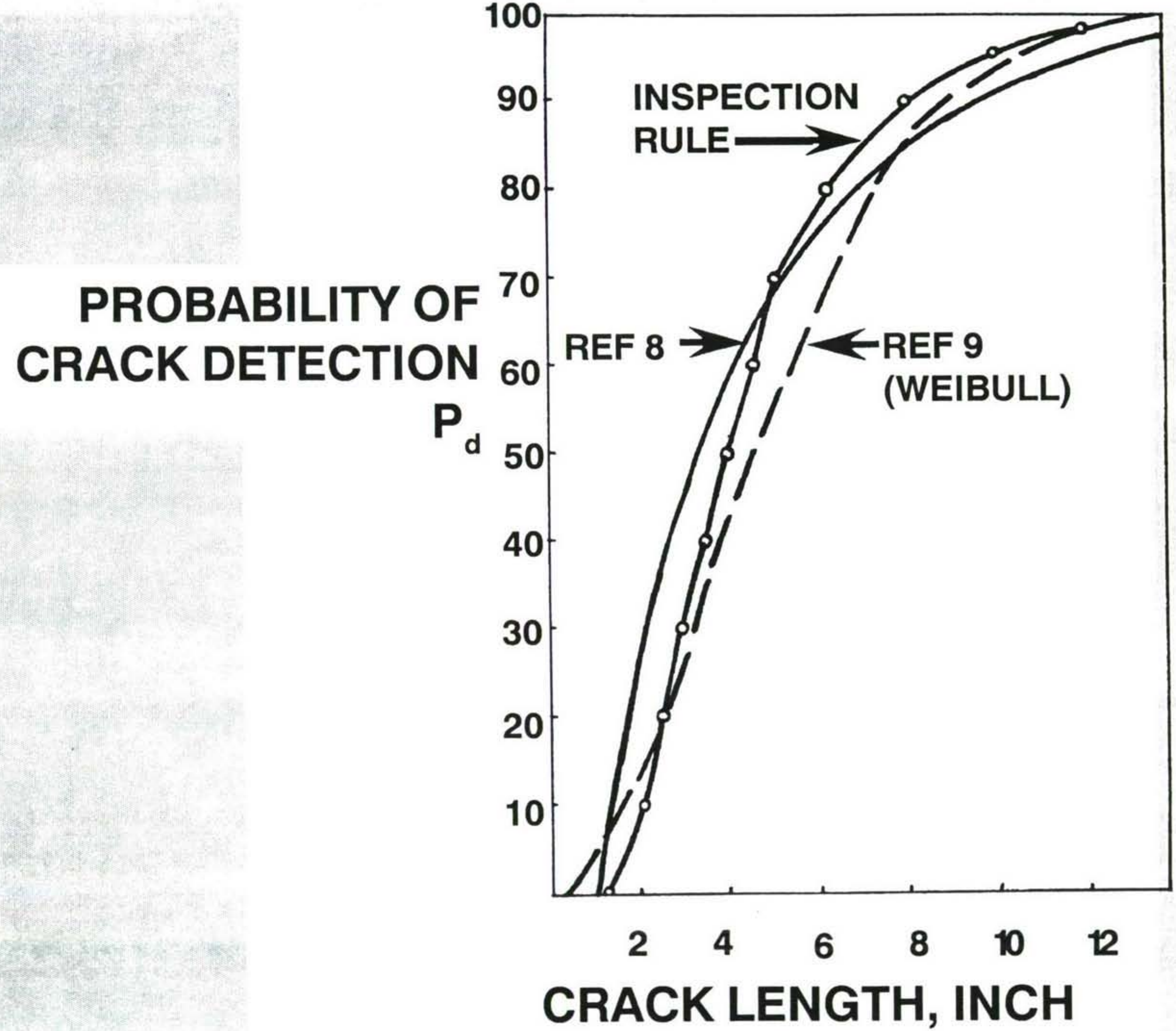
ESTIMATING PROBABILITY OF DETECTION (P_d)

EXAMPLE OF "KNOWN" STATISTICS -- DETECTABILITY "RULE"

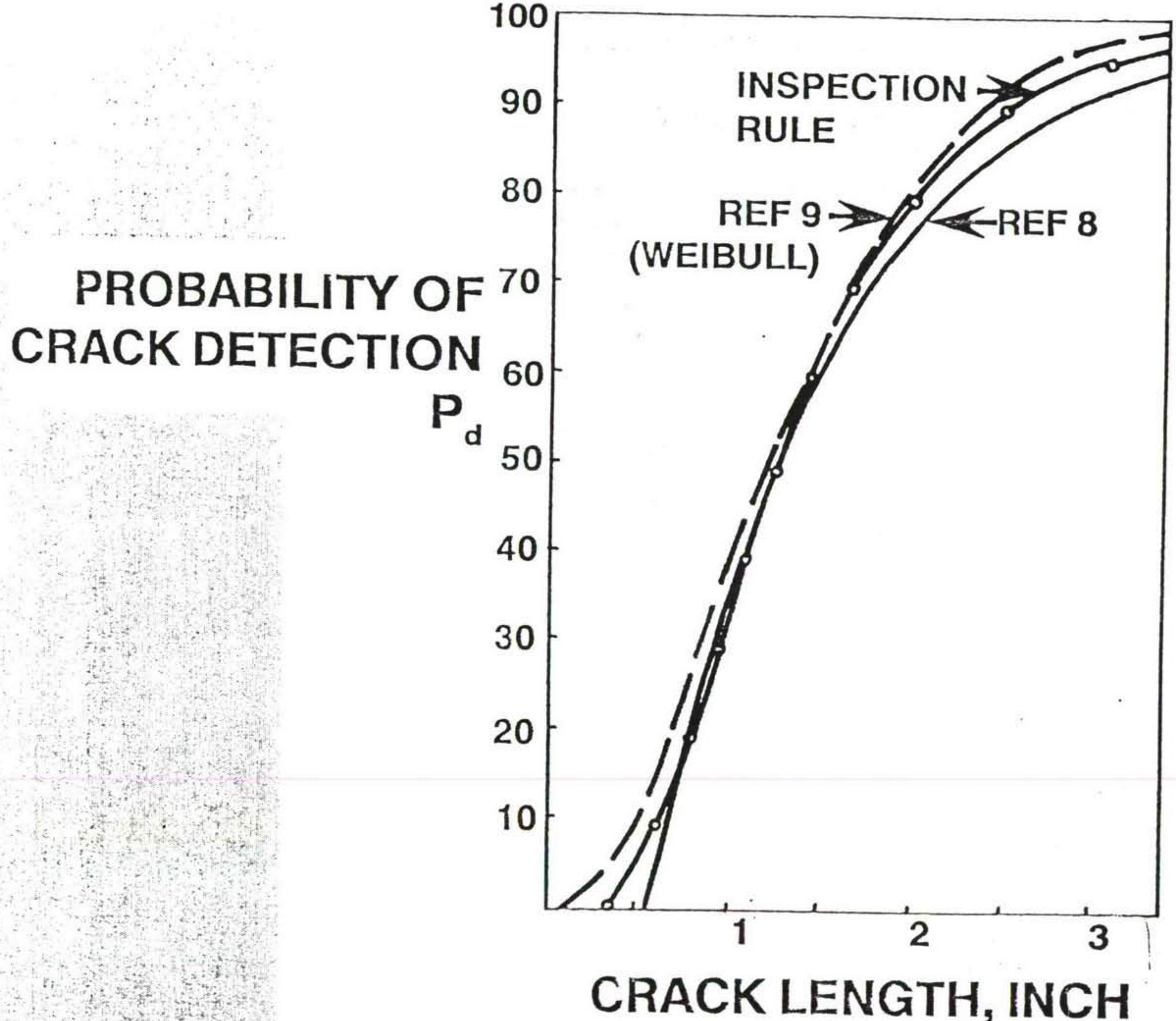
> IF A CRACK OF A GIVEN SIZE CAN BE FOUND 50 PERCENT OF THE TIME, THEN:

> ONE-HALF THAT SIZE WILL BE FOUND 10 PERCENT OF THE TIME

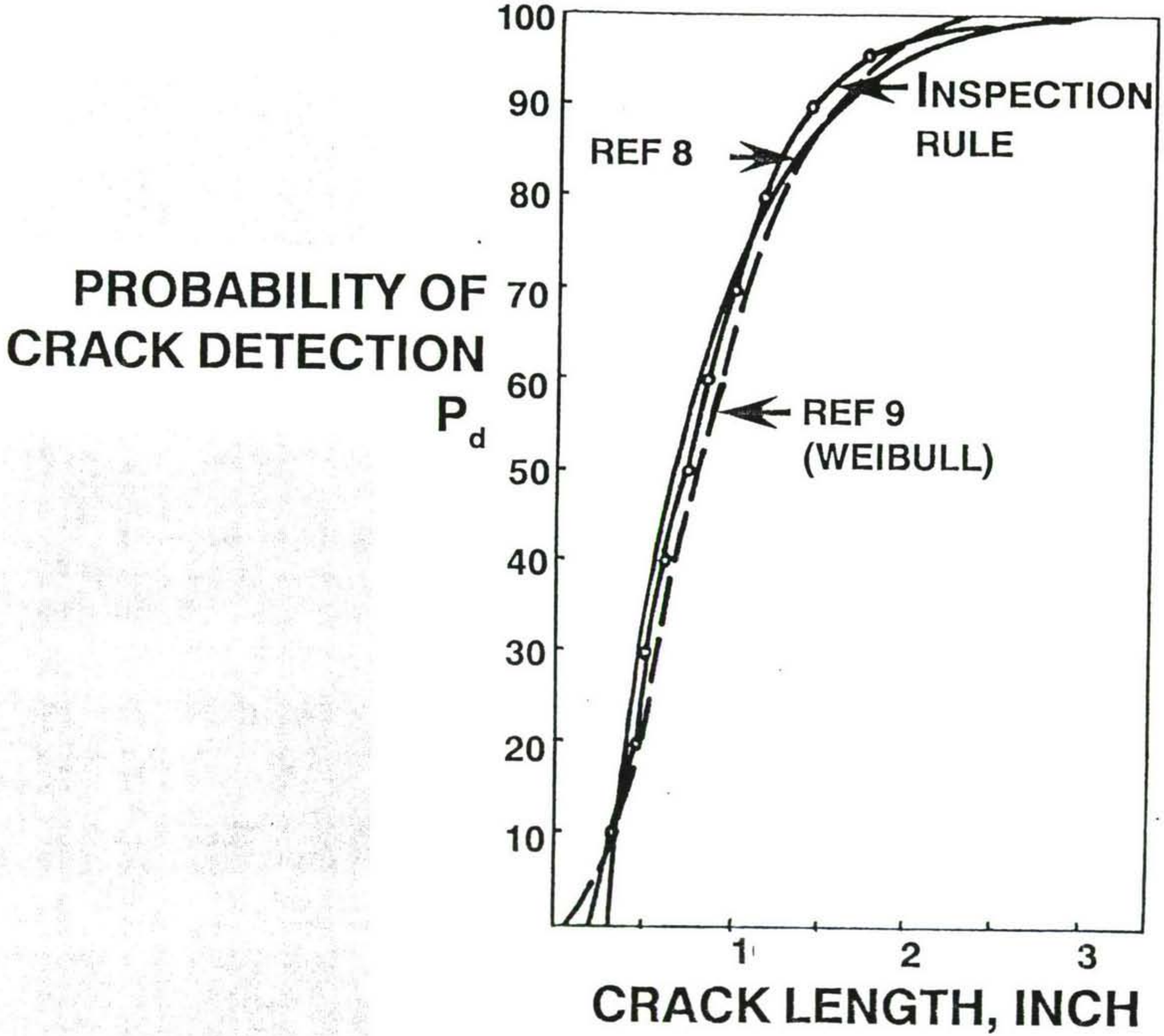
> TWO TIMES THAT SIZE WILL BE FOUND 90 PERCENT OF THE TIME



GENERAL VISUAL INSPECTIONS



SURVEILLANCE INSPECTIONS



DETAILED INSPECTIONS

2) DETECTABLE "RULE" RELATED TO INSPECTION TYPES

INSPECTION TYPE	DETECTION PROB. (P_d) SIZE, INCHES 10%	50%	90%	95%	COMMENT
GENERAL VISUAL	2	4	8	10	REF 8 & 9
CLOSE VISUAL	0.75	1.25	2.5	3.1	REF 8 & 9
DIRECTED VISUAL	0.37	0.75	1.5	1.9	
NDI	.0125	.025	.05	.062	USAF
CLOSE VISUAL	0.5	1.0	2.0	2.5	USAF DEPOT LEVEL

ESTIMATE "HUMAN FACTOR" (P_h) FROM REF. 8

TYPE OF INSPECTION	HUMAN FACTOR (P_h)
SURVEILLANCE, GENERAL AREA	0.46
SURVEILLANCE, LOCAL AREA	0.64
DETAILED DIRECTED AT SSI	0.90
SPECIAL DETAILED, NDI	0.99

THERE ARE THREE PRINCIPAL ELEMENTS:

1. ESTIMATES OF CRACK POPULATIONS

2. ESTIMATES OF INSPECTION PROGRAM EFFECTIVENESS

\> 3. <u>CRITERION FOR ACTION: SAFETY, ECONOMICS, FLEET READINESS</u>

EXAMPLE - CRITERIA FOR FORCE MANAGEMENT

SAFETY: THE PROBABILITY OF FAILURE SHOULD NOT EXCEED XXX

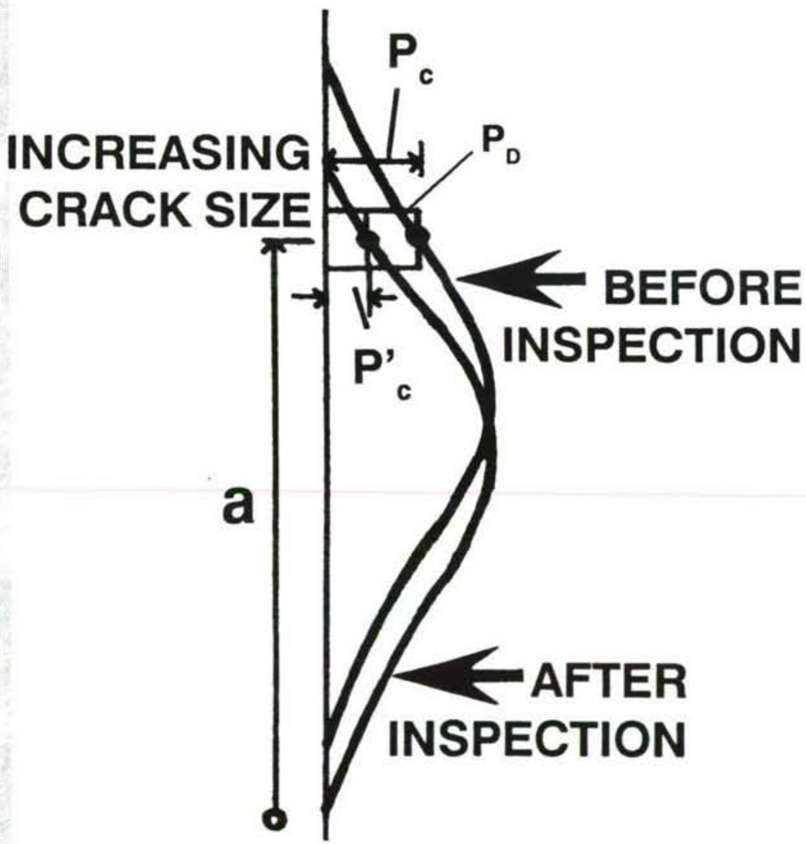
REQUIRES DETERMINATION OF REMAINING STRENGTH OF CRACKED STRUCTURE AND LOAD PROBABILITIES

ECONOMICS: THE PROBABILITY OF CRACKS ABOVE AN EASILY REPAIRED SIZE SHOULD NOT EXCEED XXX

FLEET READINESS THE PROBABILITY OF AIRCRAFT BEING OUT OF SERVICE FOR CRACK REPAIR SHOULD NOT EXCEED XXX

CALCULATIONS FOR P_D APPLIED TO CRACK SIZE DISTRIBUTION

(USING HISTOGRAM APPROXIMATION)



P_c = PROBABILITY OF CRACK OF SIZE a BEFORE INSPECTION

P_D = PROBABILITY OF CRACK OF SIZE a BEING DETECTED BY INSPECTION

P'_c = PROBABILITY OF CRACK OF SIZE a BEING MISSED

$P'_c = P_c - P_D$ where $P_D = P_c \times P_d \times P_h$

> LARGE CRACKS ARE LESS LIKELY TO BE PRESENT AFTER AN EFFECTIVE INSPECTION

CAPABILITITES AND LIMITATIONS

BEHAVIOR	CAPABILITY TO ESTIMATE	LIMITATIONS (MOSTLY DATA BASE)
"KNOWN" INITIATION STATISTICS	GOOD TO FAIR	POOLED DATA BASE ADEQUATE FOR TREND ANALYSIS
CRACK INITIATION	GOOD TO FAIR	DEPENDENT ON DATA BASE FOR SPECIAL APPLICATION - TYPICALLY WITHIN FACTOR OF 2
CRACK GROWTH	GOOD TO FAIR	DEPENDENT ON DATA BASE - LIMITED DATA TO SELECT RATE DISTRIBUTION MODEL
INSPECTION EFFECTIVENESS	FAIR TO POOR	DATA IS LIMITED FOR STATISTICS AND REFERENCE SIZES

RISK ANALYSIS STRATEGY

> KEEP ASSUMPTIONS SIMPLE

>TRY TO BOUND REAL PROBLEM

> COMPARE TO AVAILABLE EXPERIENCE

>USE RESULTS JUDICIOUSLY

EXAMPLE RISK ASSESSMENT OF MULTIPLE SITE DAMAGE

ASSUMPTIONS

> LAP JOINT IN A 2024 T3 ALUMINUM FUSELAGE

> "KNOWN" STATISTICS: LOG-STANDARD DEVIATION = 0.15

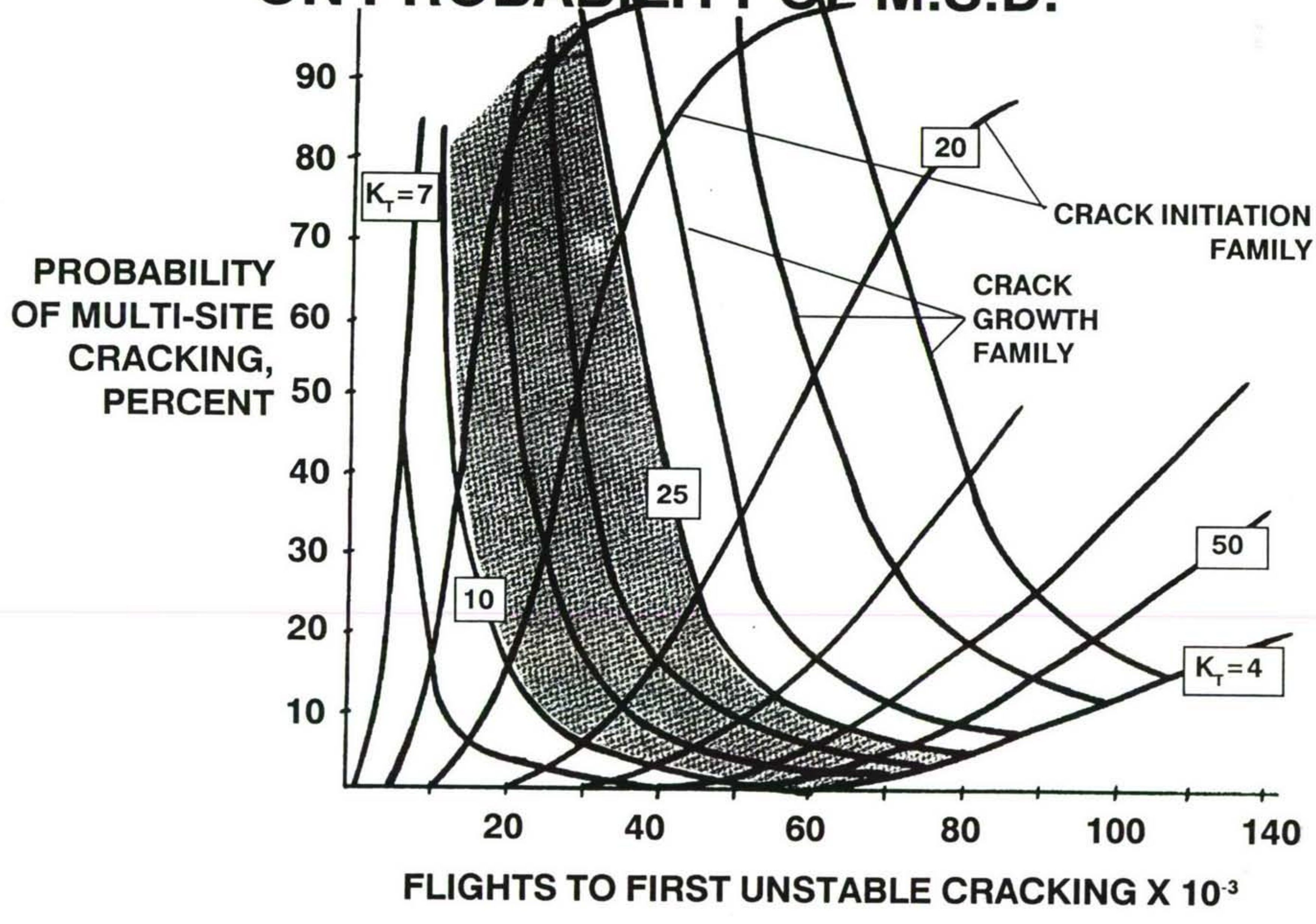
> EXAMINE K_T RANGE FROM 4 TO 7

> EXAMINE STRESS RANGE FROM 14 ksi TO 18 ksi

RISK CRITERION: 0.1% PROBABILITY OF FIRST CRACK EXTENDING TO NEXT FASTENER HOLE

QUESTION WHEN THERE IS 0.1% PROBABILITY OF A CRACK EXTENDING HOLE-TO-HOLE, WHAT ARE THE TRENDS FOR PROBABILITY OF CRACKS IN OTHER HOLES ALONG THE FASTENER LINE?

INFLUENCE OF CRACK INITIATION & GROWTH ON PROBABILITY OF M.S.D.



EXAMPLE RISK ASSESSMENT -FLEET CRACKING

ASSUMPTIONS

> A FLEET OF 100 A/C WITH TWO POTENTIAL CRACK SITES PER A/C

> AIRCRAFT AGE IS DISTRIBUTED BETWEEN 40,000 & 60,000 FLIGHTS

> MEAN FATIGUE LIFE FOR DETAIL IS ESTIMATED, 120,000 FLIGHTS

> "KNOWN" STATISTICS, LOG-STANDARD DEVIATION = 0.15

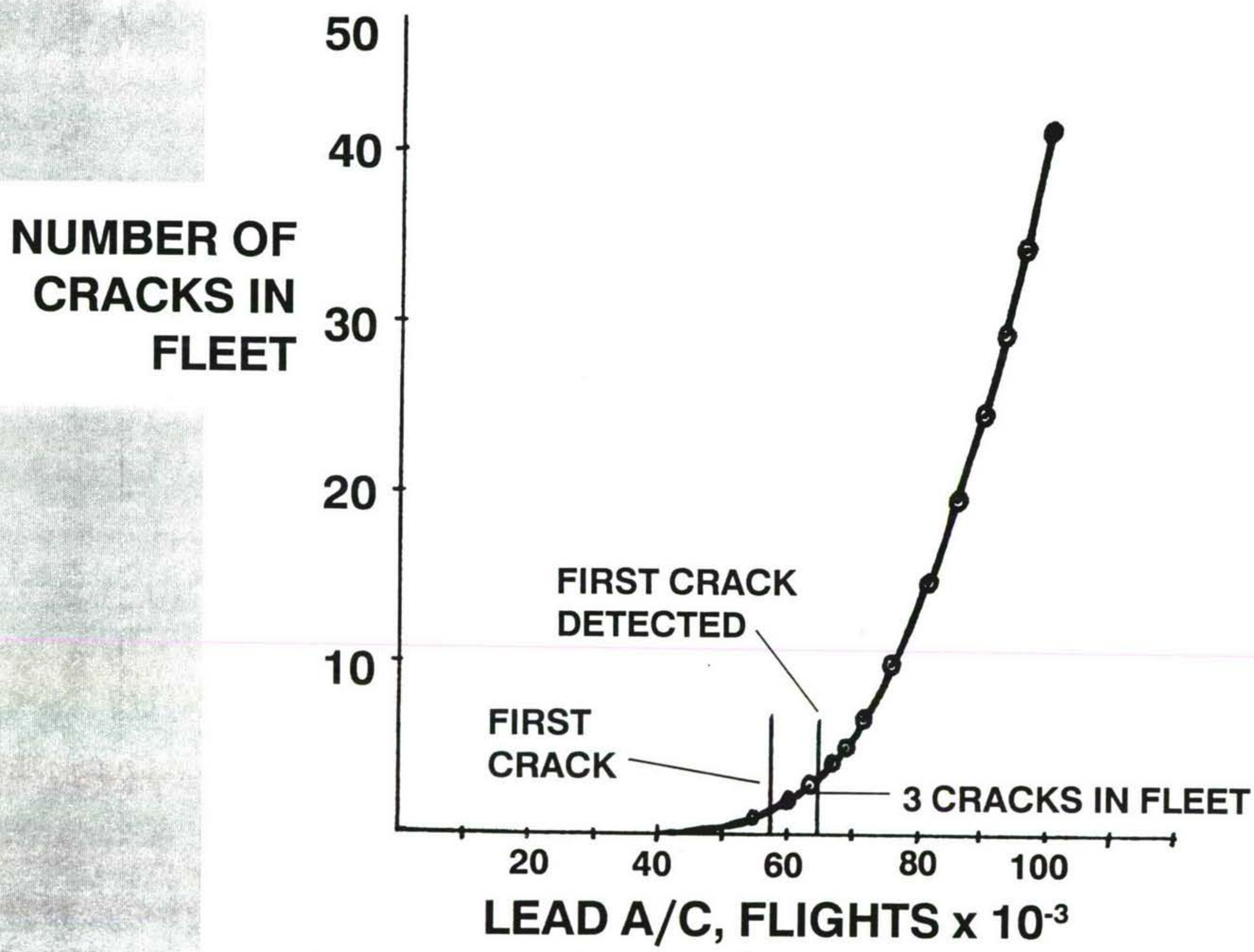
> CRACK GROWTH FROM 0.1 INCH INITIATION TO UNSTABLE AT 5 INCH IS 20,000 FLIGHTS

> INSPECTIONS ARE ULTRASONIC AT 2,000 FLIGHT INCREMENTS

QUESTION

WHAT WILL BE THE LIKELY FLEET CRACKING, CRACK DETECTION EXPERIENCE?

LIKELY FLEET CRACKING EXPERIENCE



FINAL COMMENTS

IF ONE ACCEPTS THE QUALITATIVE NATURE OF STRUCTURAL FATIGUE RISK ASSESSMENTS AND IS WILLING TO KEEP THE ASSUMPTIONS RELATIVELY SIMPLE (THUS UNDERSTANDABLE), THE RESULTING ANALYSES CAN BE INFORMATIVE AND OF VALUE TO PLANNING STRUCTURAL FATIGUE INSPECTIONS AND MAINTENANCE.

PROBABILISTIC STRUCTURAL ANALYSIS METHODS (PSAM) FOR AEROSPACE SYSTEM COMPONENTS

Hal Burnside
Tom Cruse

Division of Engineering and Materials Sciences
Southwest Research Intitute
San Antonio, Texas

Presented at the 1989 USAF Structural Integrity Program Conference
San Antonio, Texas
December 5-7, 1989

NASA

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

PROBABILISTIC STRUCTURAL ANALYSIS METHODS FOR SELECT SPACE PROPULSION SYSTEM STRUCTURAL COMPONENTS

NASA PROGRAM MANAGER
DR. CHRISTOS C. CHAMIS
NASA LEWIS RESEARCH CENTER

SwRI PROGRAM MANAGER
DR. THOMAS A. CRUSE
SOUTHWEST RESEARCH INSTITUTE

875260-102

PRESENTATION OVERVIEW

- Why Probabilistic Design/Analysis?
- Probabilistic Finite Elements
- Probabilistic Analysis Methods

  -Monte Carlo
  -Fast Probability Integration
- Validation Studies

  -Beam Analysis
  -Fracture Mechanics Crack Growth
- Verification Studies
- Future Work

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WHY DO PROBABILISTIC DESIGN?

- HOW SAFE IS IT?
- HOW WILL IT PERFORM?
- WHAT IS MY CONFIDENCE?
- HOW CAN I MAKE IT MORE RELIABLE?

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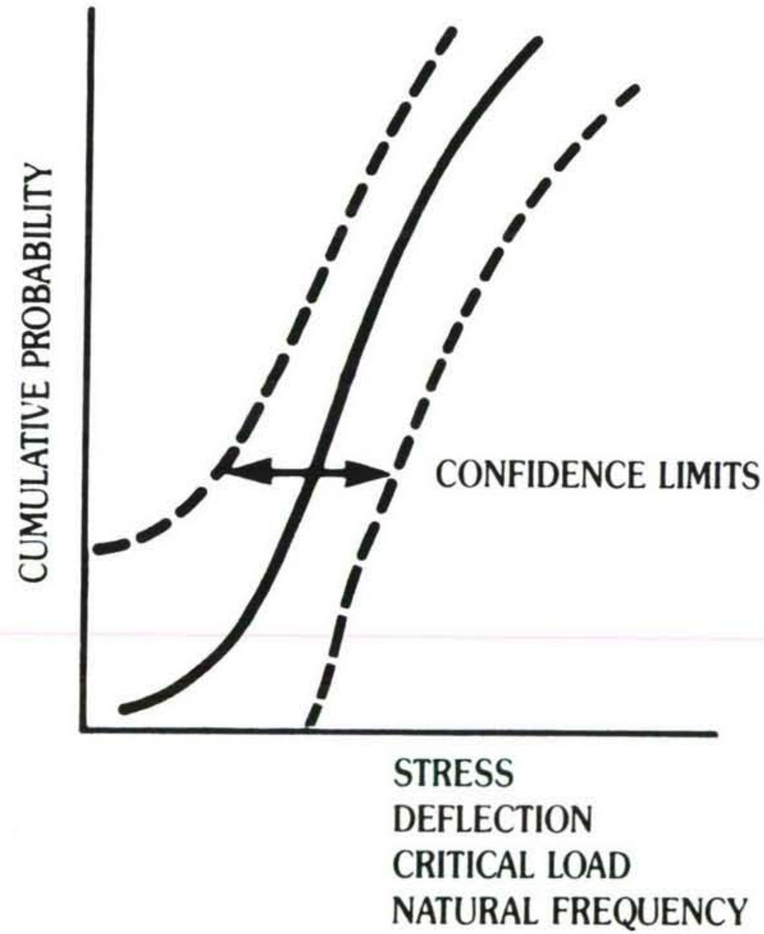
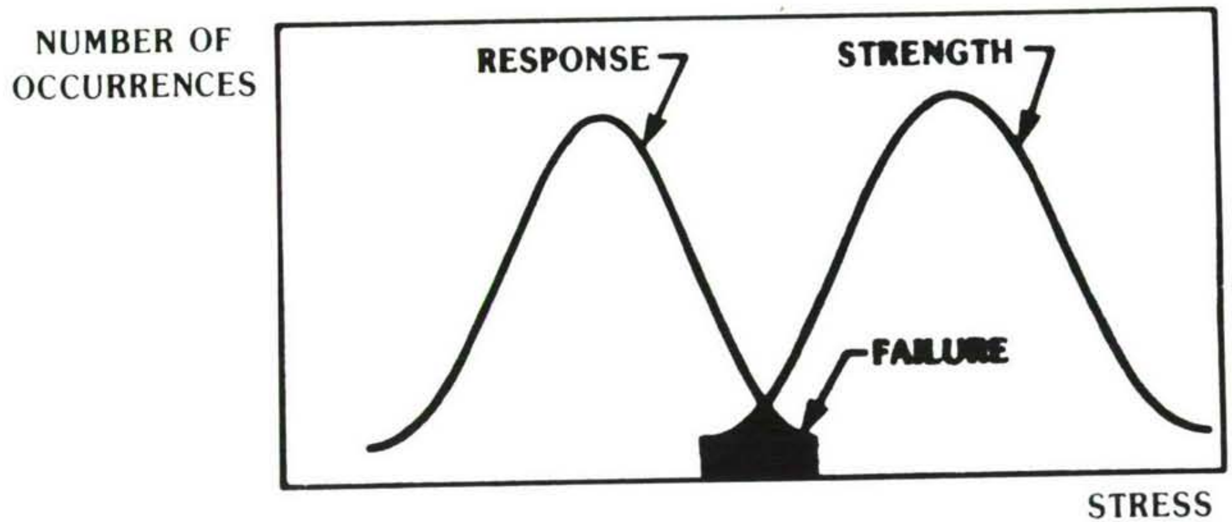
PROBABILISTIC DESIGN METHODS WILL SIMULATE "REAL WORLD" STRUCTURAL RESPONSES



DUE TO DESIGN UNCERTAINTIES:

- LOADING
- MATERIAL BEHAVIOR
- GEOMETRY, TOLERANCES
- BOUNDARY CONDITIONS

RISK ANALYSIS COMPARES RESPONSE TO EXPOSURE

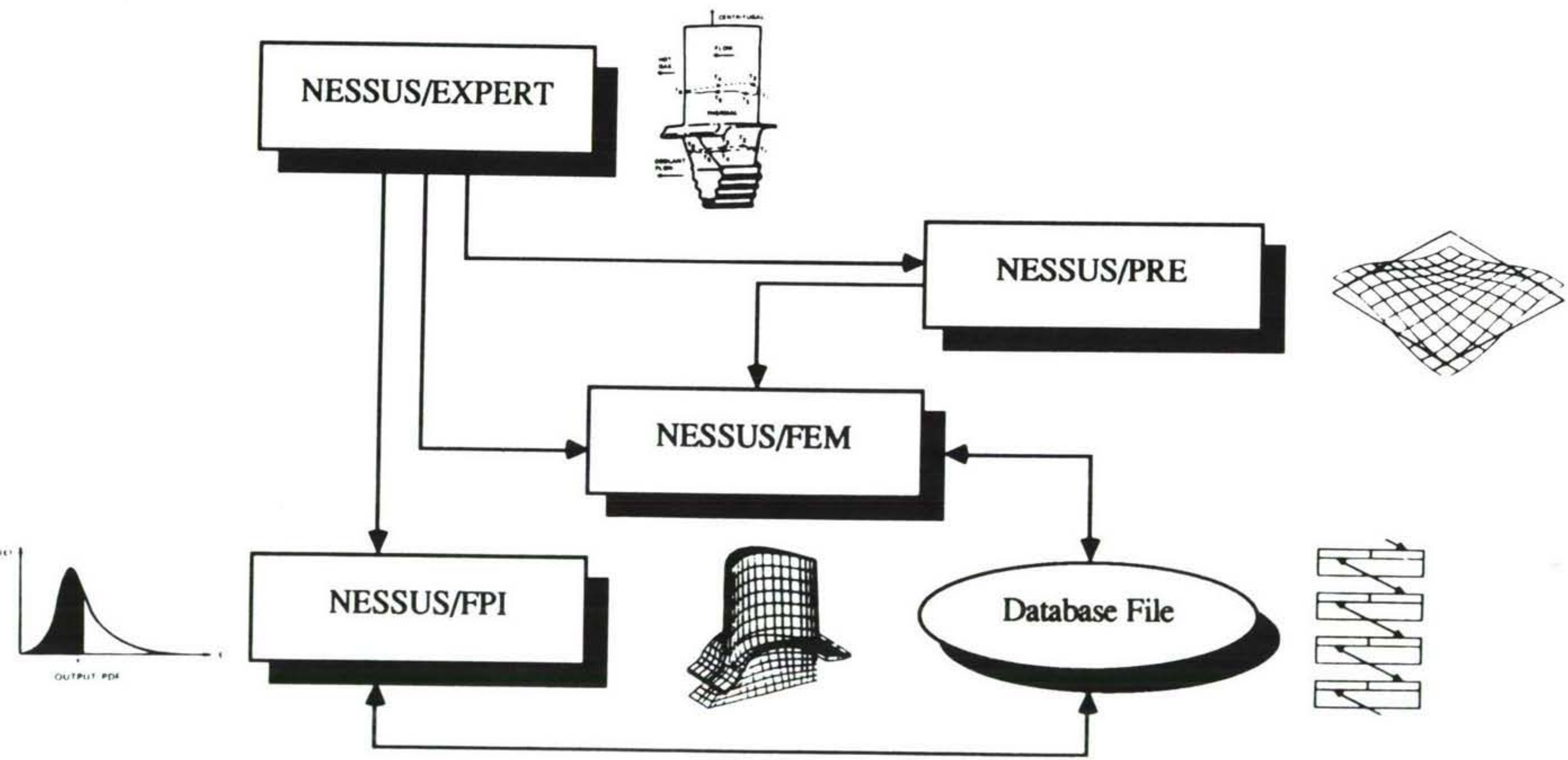


FAILURE = JOINT DISTRIBUTION OF RESPONSE AND STRENGTH

RELIABILITY ESTIMATION METHODS INTEGRATED WITH FINITE ELEMENT ANALYSIS

- USER DEFINES UNCERTAIN DATA IN MODEL
- FEM MODELS PREDICT DESIGN SENSITIVITY
- RELIABILITY METHODS COMBINE UNCERTAINTY AND SENSITIVITY DATA
  - FAST PROBABILITY INTEGRATION (FPI)
  - MONTE CARLO SIMULATION (MC)

AN OVERVIEW OF THE NESSUS CODE
Five Major Software Modules Working Together



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PROBABILISTIC FEM USES FIRST ORDER RANDOM VARIABLE PERTURBATIONS

- RANDOM VARIABLES (X)
  - PROPERTIES
  - GEOMETRY

STANDARD FEM (0^{TH} ORDER; 1^{ST} MOMENT)

$$[K]\ \{u\} = \{F\}$$

PROBABILISTIC FEM (1^{ST} ORDER; 2^{ND} MOMENT)

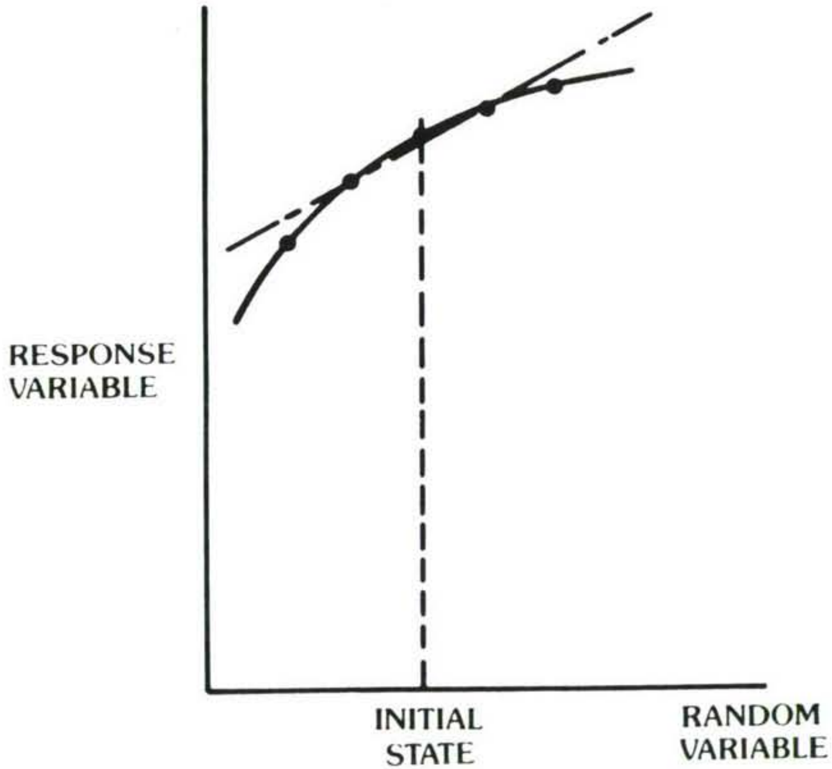
EXPECTED SOLUTION:

$$E\{u\} = \{u_o\} + [K_0]^{-1}(E\{F\} - \{F_o\}) - [K_o]^{-1}\left[\frac{\partial K}{\partial x}\right](E\{x\} - \{x_o\})\ \{u_o\}$$

SOLUTION VARIANCE:

$$\sigma^2\{u\} = ([K_0]^{-1}\ \sigma\{F\})^2 + ([K_o]^{-1}\left[\frac{\partial K}{\partial x}\right]\{u_o\})^2\ \sigma^2(x)$$

NESSUS/FEM GENERATES RESPONSE MODEL BY EFFICIENT PERTURBATION ANALYSIS



- INDEPENDENT RANDOM VARIABLES
  - GEOMETRY
  - MATERIAL PROPERTIES
  - BOUNDARY CONDITIONS
- ITERATIVE SOLUTION ALGORITHM
  - SMALL PERTURBATIONS
  - RETAIN INITIAL STIFFNESS MATRIX
  - MODIFY RIGHT-HAND SIDE ONLY
- RESPONSE SURFACE FITTING
  - LEAST SQUARES ERROR
  - LINEAR OR QUADRATIC

875260-6

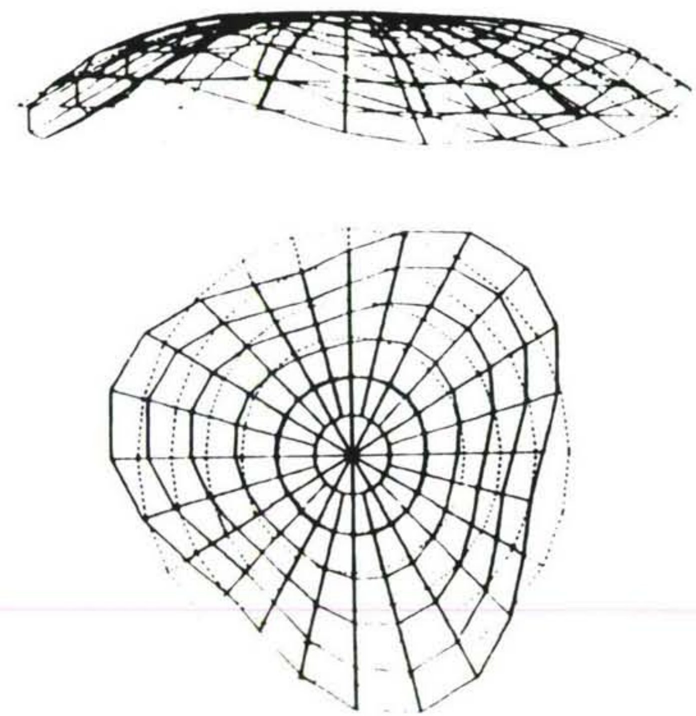
SPATIALLY CORRELATED RANDOM VARIABLE FIELDS ARE DECOUPLED IN A PRE-PROCESSING OPERATION

Random field variables may be

- **Uncorrelated**
- **Partially correlated**
- **Fully correlated**

Random variables may include

- **Geometry**
- **Material properties**
- **Point loads**
- **Pressure loading**
- **Thermal loading**
- **Body force**
- **Anisotropy orientation**



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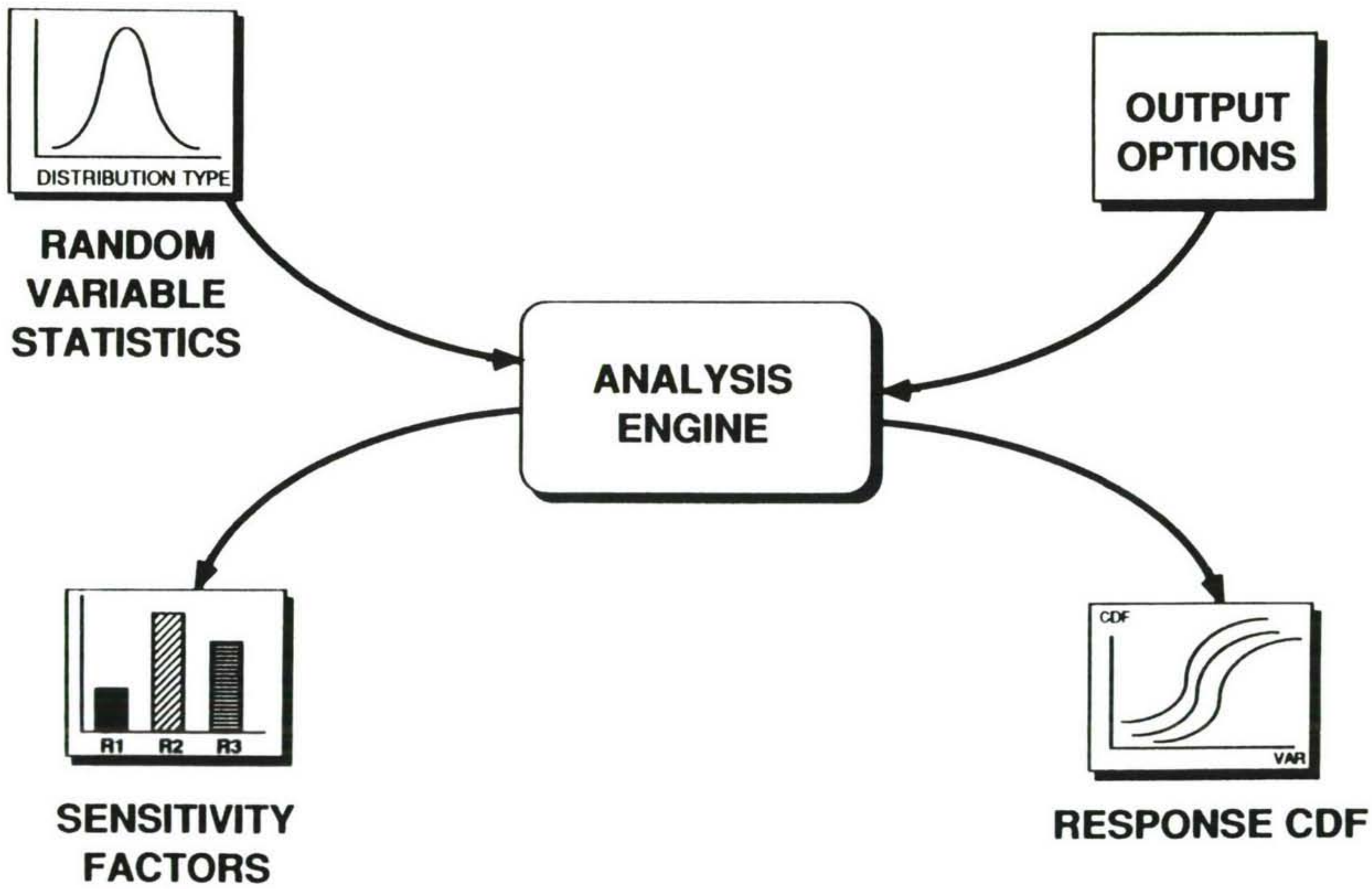
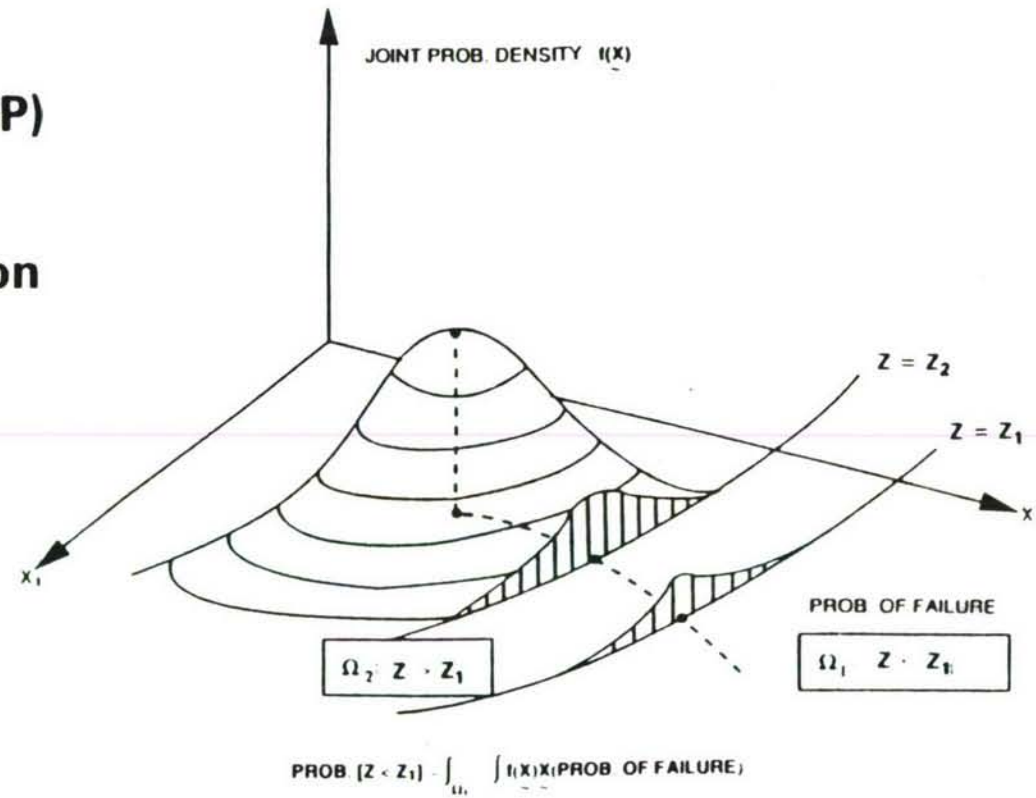
FPI OUTLINE


DISTRIBUTION TYPE
RANDOM
VARIABLE
STATISTICS
OUTPUT
OPTIONS
ANALYSIS
ENGINE
R1
R2
R3
SENSITIVITY
FACTORS
CDF
VAR
RESPONSE CDF

Establish CDF Using Fast Probability Integration Algorithm

- Slice Total Probability Volume Into n Pieces
- Identify Most Probable Point (MPP) For Each Objective Function Z_i
- Establish Quadratic Approximation At MPP
- Solve the Simplified Probability Integration Problem



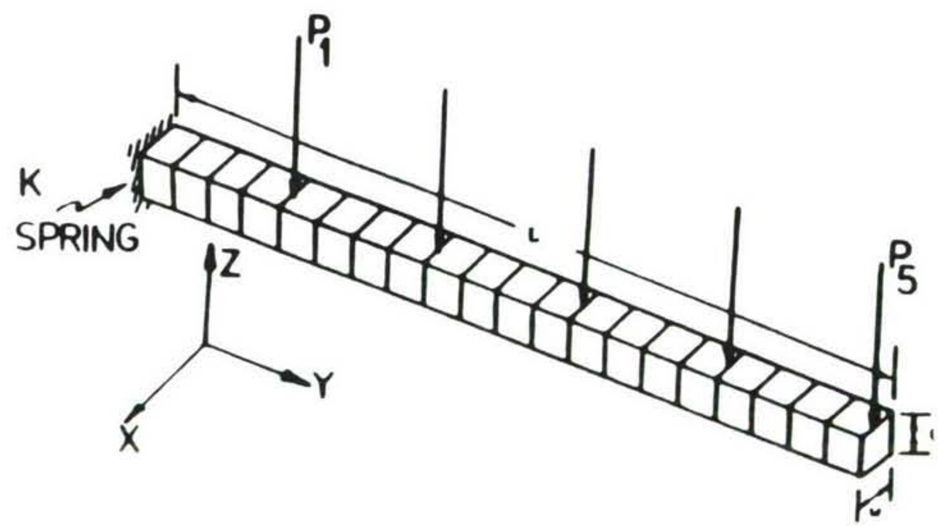
PSAM NESSUS VALIDATION

Case 1 - Static Analysis of Cantilever Beam with Correlated Loadings

Problem Definition

Random Variables

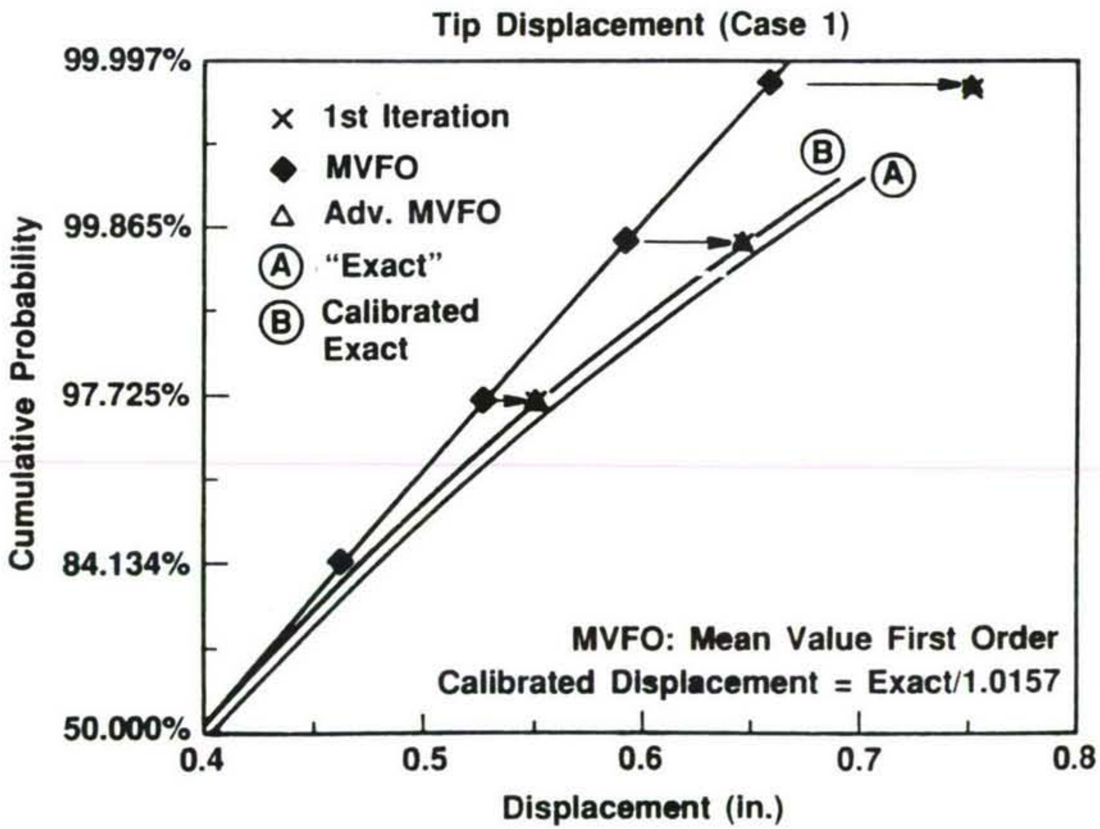
- Correlated Loads
- Young's Modulus
- Length
- Thickness
- Width
- Base Spring Stiffness



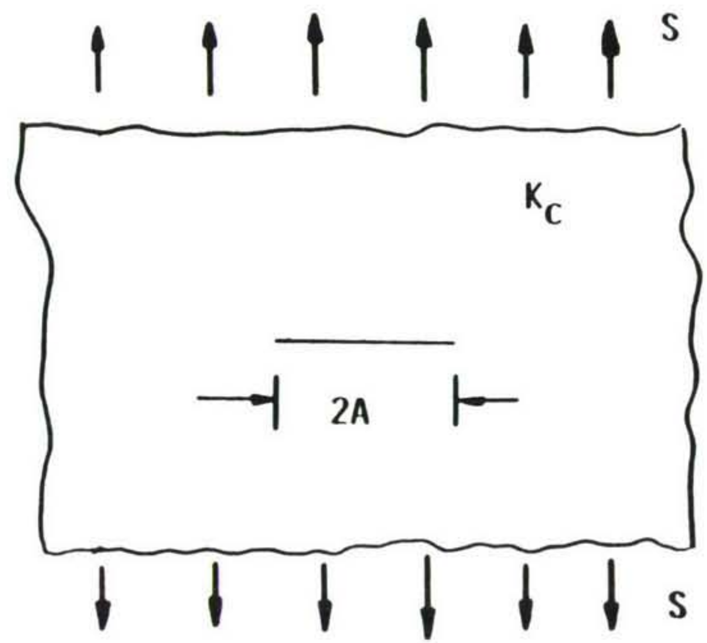
PSAM NESSUS Validation

Case 1–Static Analysis of Cantilever Beam with Correlated Loadings

Results



FRACTURE MECHANICS CRACK GROWTH



CRACK GROWTH LAW: $\frac{da}{dN} = C(\Delta K)^m$ [Paris]

STRESS INTENSITY FACTOR: $\Delta K = \Delta S \sqrt{\pi a}$

RANDOM VARIABLES: INITIAL CRACK LENGTH (A_I)

PARIS COEFFICIENT (C)

PERFORMANCE FUNCTION FOR LIFE

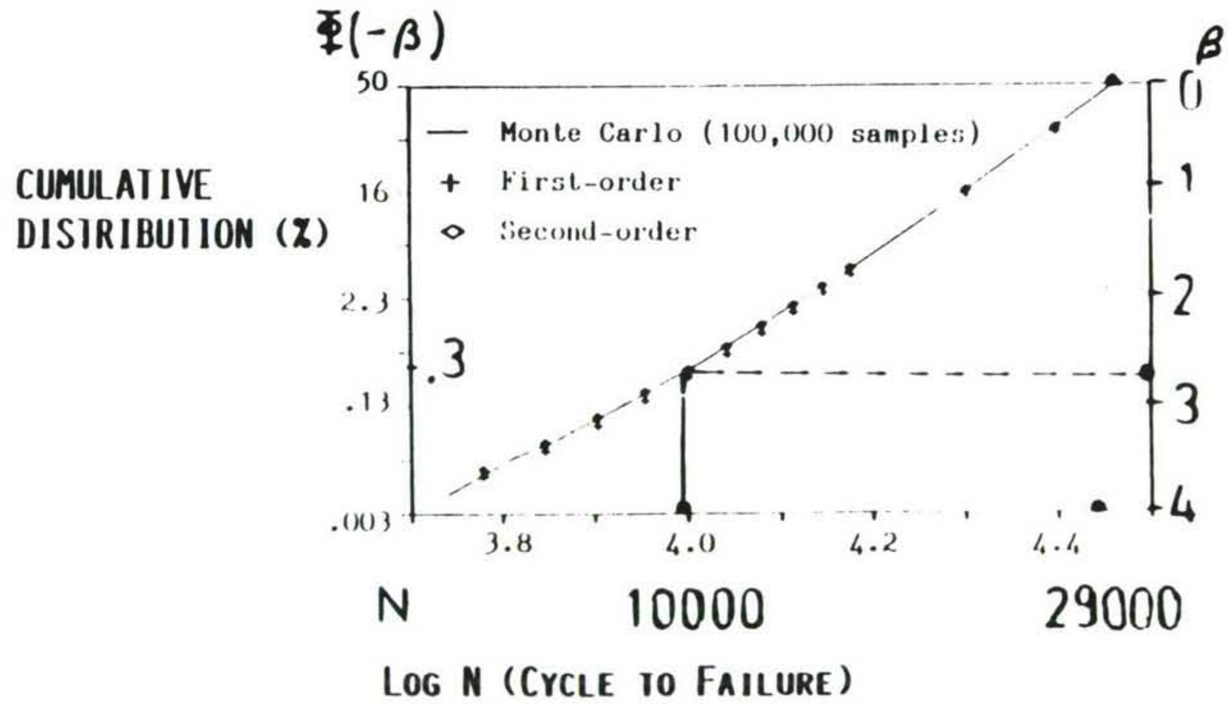
$$N(X) = \int_{a_i}^{a_f} \frac{1}{c(\Delta S\sqrt{\pi a})^m} da$$

$$= \left[\frac{1}{\pi c(\Delta S)^2}\right] \ln \left(\frac{a_f}{a_i}\right)$$

PARAMETER DATA

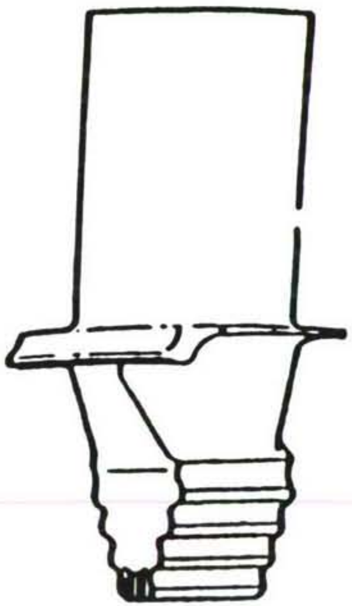
PARAMETER	DISTRIBUTION TYPE	MEDIAN	COV
A_I	LOGNORMAL	0.000127 M	0.5
C	LOGNORMAL	1.E-8	0.3
ΔS	DETERMINISTIC	345 MPA	0
M	DETERMINISTIC	2	0
A_F	DETERMINISTIC	0.00127 M	0

FPI PROVIDES CUMULATIVE DISTRIBUTIONS FOR LIFE

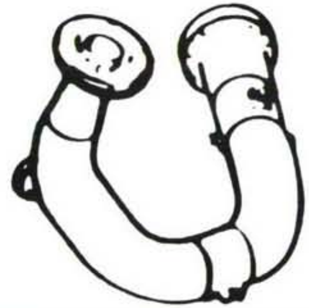


VERIFICATION ANALYSES OBJECTIVE

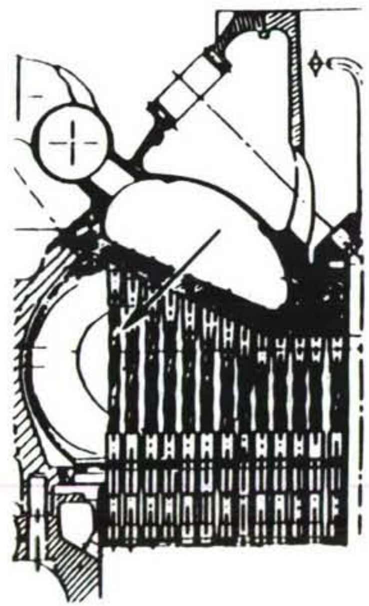
TO APPLY THE METHODS DEVELOPED TO THE ANALYSIS OF TYPICAL SPACE PROPULSION SYSTEM COMPONENTS



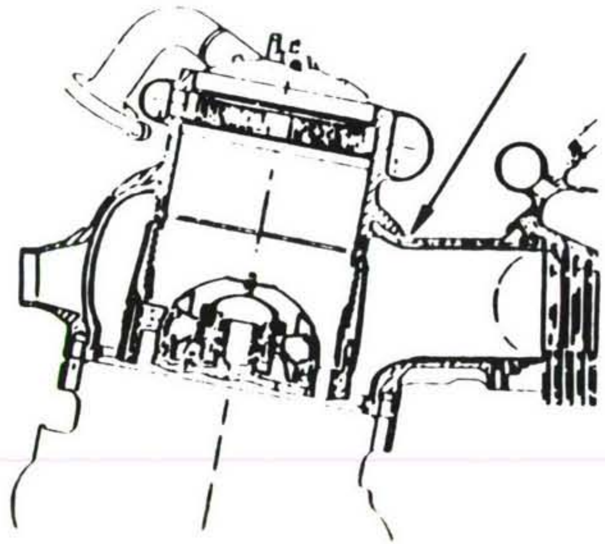
TURBINE BLADE



HIGH PRESSURE DUCT



LOX POST



TRANSFER TUBE LINER

PROBABILISTIC STRUCTURAL ANALYSIS METHODS

PSAM PROGRAM EXTENDED FIVE YEARS

- **Preliminary Test Release of NESSUS Software Made**

  -Workshop for users held at LeRC
  -Numerous Aerospace firms attended
  -Commercial support to NESSUS/SwRI announced

- **Extended Program Focuses on Reliability Issues**

  -NESSUS software to be expanded
  -Additional Probabilistic Technology to be developed
  -Annual code updates planned

- **Five Year Plan details**

-FY90:	**Component reliability with SOA damage models**
-FY91:	**System reliability and Bayesian capability**
-FY92:	**Progressive damage modeling; Certification**
-FY93-94:	**System/Component Health Monitoring**

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1989 USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE

SMART STRUCTURES APPLICATION TO AIRCRAFT STRUCTURAL INTEGRITY PROGRAMS

by

J. Mohammadi, J. A. Cicero and K. Kasai

SEI

SYSTEMS & ELECTRONICS, INC.
190 Gordon Street
Elk Grove Village, IL 60007
(708) 228-0985 FAX (708) 228-1164

5-7 December 1989

SMART STRUCTURES APPLICATION TO AIRCRAFT STRUCTURAL INTEGRITY PROGRAMS

1.0 INTRODUCTION

In recent years there has been a growing interest in the application of smart structures to such areas as energy control and monitoring of buildings, automated emergency alarm systems, structural health monitoring, etc. A smart structure contains two essential parts: (i) a sensing module, and (ii) an artificial intelligence (AI) module. The sensing module includes a system of sensors that continuously collects data on the status of a structure (i.e., internal strains) and transfers the information to the AI module for processing, evaluation and decision making. Smart structures can be applied to the aircraft structural integrity evaluation process. In such application, the smart structure can be designed for a specific task. An example of this is a system designed to monitor the fatigue crack and crack growth in the critical structural components of an aircraft and make recommendations on inspection demands of the aircraft. In a more broad and, perhaps, more idealized case, the smart structure can be designed to monitor the state of the internal strain on all aircraft components, make the necessary evaluation of the aircraft's structural integrity, and recommend specific measures to maintain this integrity. The advantage of a smart structure system over the current structural integrity program is that the smart structure can continuously monitor the structure and can conduct rapid structural, statistical and decision making analyses as needed.

The sensing module is responsible for data gathering in the smart structure. Thus it should be reliable, properly installed and calibrated, and durable in the load environment experienced by aircraft. Current sensing module technology includes conventional strain gages, load cells and acoustic emission systems. The latter has been utilized in applications such as flaw detection in metals. A new comer in the field of sensor technology is the optical fiber. With proper design, the optical fiber has potential for use as a sensing module in data gathering applications. The AI module contains many software units designed for specific tasks. A main program controls the functions of these units, maintains a data-base, generates information and makes decisions on the status of the structure based on the incoming data.

This paper reviews the availability of the technology to develop an airframe smart structure. Current systems used in structural integrity programs are reviewed and their potential for inclusion during the smart structure fabrication process is discussed. Different sensor types are examined and their application in the smart structure is explained. Sensors for which information on performance and capability does not exist are identified and discussed.

2.0 REVIEW OF EXISTING TECHNOLOGY

A smart structure, as defined here, refers to a monitoring mechanism capable of sensing the performance of an engineering system and making the necessary decisions so that the system's continuous and reliable service is maintained. Smart structures (also known as intelligent structures) have now been used in such applications as monitoring and control of energy in buildings, security and alarm systems, preventive maintenance management, etc. [1]. When applied to structural tracking of an aircraft, a smart structure will continuously monitor and control the state of health of the structure. This is accomplished by conducting tasks such as strain data gathering, damage analysis, maintenance and repair decisions of the aircraft components, advice on the flight conditions, etc.

Current structural tracking of airframe systems is accomplished through the Aircraft Structural Integrity Program (ASIP), and the Navy Aircraft Structural Integrity Program (NASIP), Refs. [2] and [3]. The two programs set guidelines for the life-cycle management of aerospace structures for the Air Force and Navy, respectively. Issues such as design, development, testing, use, inspection, maintenance and retirement of airframe structures are covered under these programs. One specific element in aircraft life-cycle management includes gathering, interpreting and analyzing aircraft strain data and then evaluating the current health of an aircraft. Current technology achieves these objectives with computerized systems that gather data via strain gages mounted at fatigue-critical locations in an aircraft. In a system developed by Systems & Electronics, Inc., the data gathering process is controlled by a microprocessor-based recorder which is also capable of performing limited "on-site" data analysis. Systems such as this have been successfully used aboard the Air Force and Navy aircraft for strain and other flight parameter acquisition.

In the area of data management, rapid data analysis and decision making, several new advances have occurred. New computer hardware and AI software allow for quick on-site data evaluation, data reduction, data interpretation and knowledge base maintenance. Such programs can effectively be used in building the needed AI module in the smart structure.

The current data acquisition technology, however, needs additional development before it can be applied to building a smart structure. An ideal smart structure is expected to continuously monitor the state of the aircraft's internal stresses. The hardware needed to store, manage, and process the vast amount of structural data is currently under development. New developments in the sensor technology as well as the computer hardware and software are essential in building a smart structure.

3.0 SENSOR TYPES

As was described earlier, a smart structure contains a sensing module and an artificial intelligence module. Sensors that have been used in the airframe structure are described in this section.

3.1 Fiber Optics

The application of fiber optics to the telecommunication industry is well established. The basis of fiber optic communication is the optical fiber. This is a thin flexible glass or plastic cable through which light is transmitted. As a transmission link, a fiber connects a transmitter and a receiver. The central part of the transmitter is a light emitting diode (LED) or an injection laser diode (ILD) which converts electrical signals into light signals. An oscillator drives the source that changes the digital logic input voltage into a modulated light output. The receiver contains a photodiode capable of converting the light back into electrical signals, and an output circuit, which amplifies and reshapes the signal into a measurable form [4,5].

In recent years, the optical fiber as a means to measure strains in engineering materials has attracted researchers in the field of experimental stress analysis. With any deformation in the form of severe bending, tension or contraction experienced by a fiber, the intensity and/or phase of the transmitted light is changed and the results appear in the form of a shift in the output signal. This can effectively be used as a means to measure strain variations in metallic and composite structural materials.

In a recent laboratory investigation conducted by Systems & Electronics, Inc. (SEI) on fiber optics, a number of fibers were tested for their potential as a means to measure strains. The tests showed that:

- Optical fiber is a potential sensing device in strain measurements

- The reliability of measured strains depends primarily on the fiber diameter, its length, its bond to the structure under test and the sensitivity of the photodiode receiver.

Thus with a proper design (both for the electronics and the fibers) it is possible to optimize the sensitivity of the optical fiber for maximum strain measurement accuracy. In terms of the application to structural components, optical fibers are ideal for composite materials where a reliable bond strength between the composite and the embedded fibers can be maintained during the material straining. However, fibers need to be tested for their durability in a repeated load environment. Under a severe loading/unloading condition, the potential to develop a rupture

in the fiber needs to be investigated.

The specific application of the optical fiber in a smart structure and the tests conducted by the authors on the potential use of the fiber as the sensing module are explained later in this paper.

3.2 <u>Acoustic Emission</u>

Acoustic emission analysis has been used extensively in metal fracture mechanics. The acoustic emission analysis is based on the fact that short acoustic impulses are emitted during a deformation process in the metal or during a discontinuity in the form of structural crack and crack growth [6,7]. With a sensitive transducer (e.g., a piezo-electric transducer) to measure and detect impulses generated during the acoustic emission process, any deformation in the material can be measured reliably and effectively. This can be done for nearly all materials including metal [8] and composites [6].

Figure 1 shows the schematic diagram of a simple acoustic emission measurement system. The sensors are mounted on the structural component. The emitted impulses are then measured and converted into data. Figure 2 shows the process of signal distortion due to acoustic propagation and transducer characteristics. When applied to airframe structures, the acoustic emission system should be carefully designed with respect to both the type of sensor used and the filtering process. Because of its previous application to metallic specimens, it is reasonable to believe that the acoustic emission system is more suitable for use in metallic airframe components. The specific application of acoustic emission analysis to airframe structures made of composite materials, requires further investigation. Also, additional studies are needed to ascertain optimal pulse generation schemes and optimal reliability issues before the acoustic emission system can be used in an airframe smart structure.

3.3 <u>Strain Gages</u>

The use of strain gages in strain data acquisition of airframe systems is well established. Systems & Electronics, Inc. has used and is currently using strain gages as sensors in the Navy test aircraft. The mechanism of strain measurement in this type of sensor is well known and can be found in Refs. [9] and [10]. Strain gages can be used in a smart structure. However, their useful life and their efficiency are issues that may impose certain limitations on their applicability to the sensing module in a smart structure. With this regard, the development of new techniques for quick installation and replacement should be considered and researched further.

Based on the available information on the three potential sensors described in Sections 3.1, 3.2 and 3.3, Table I was

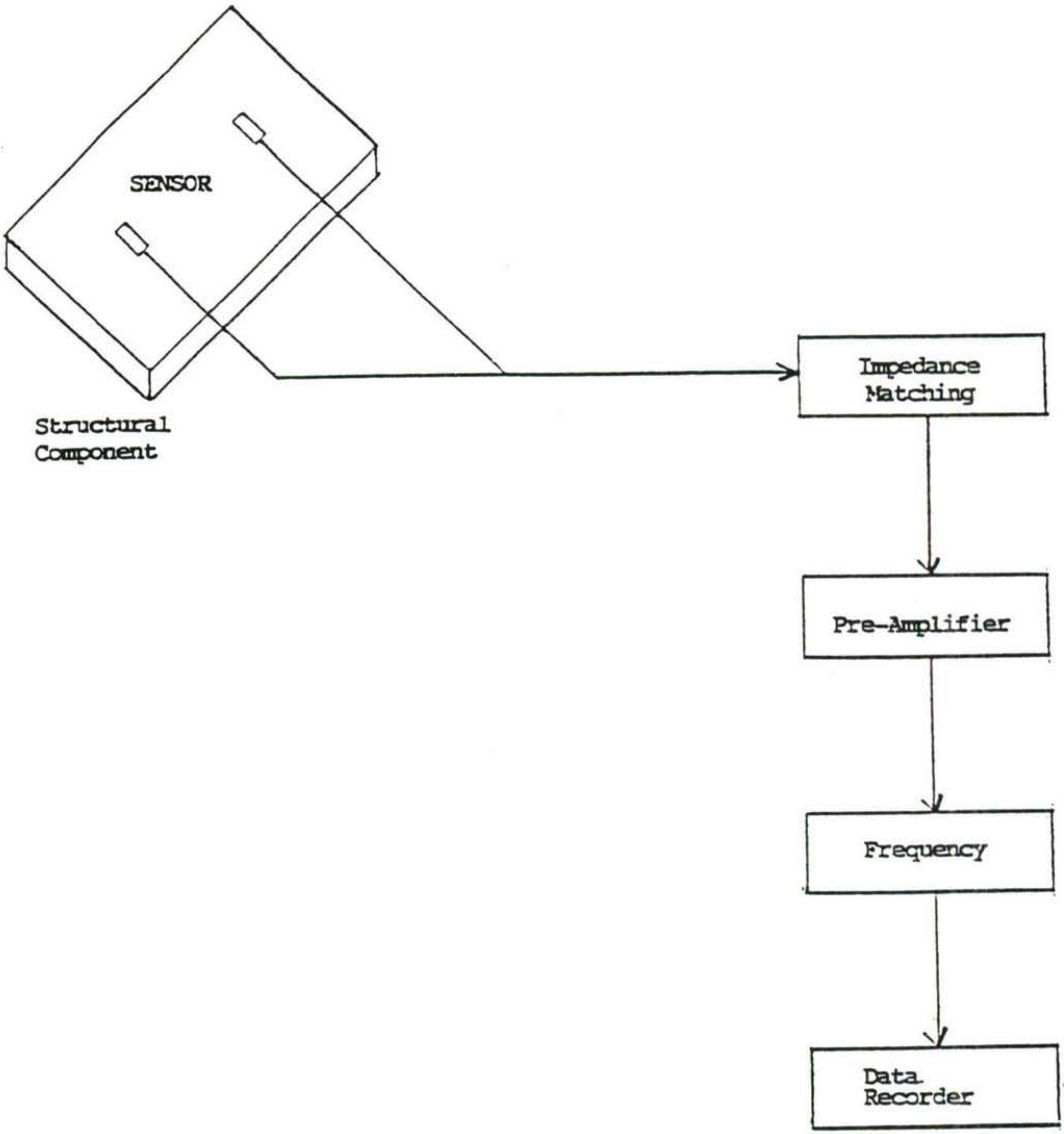

Fig. 1 - Acoustic Emission Measurement System

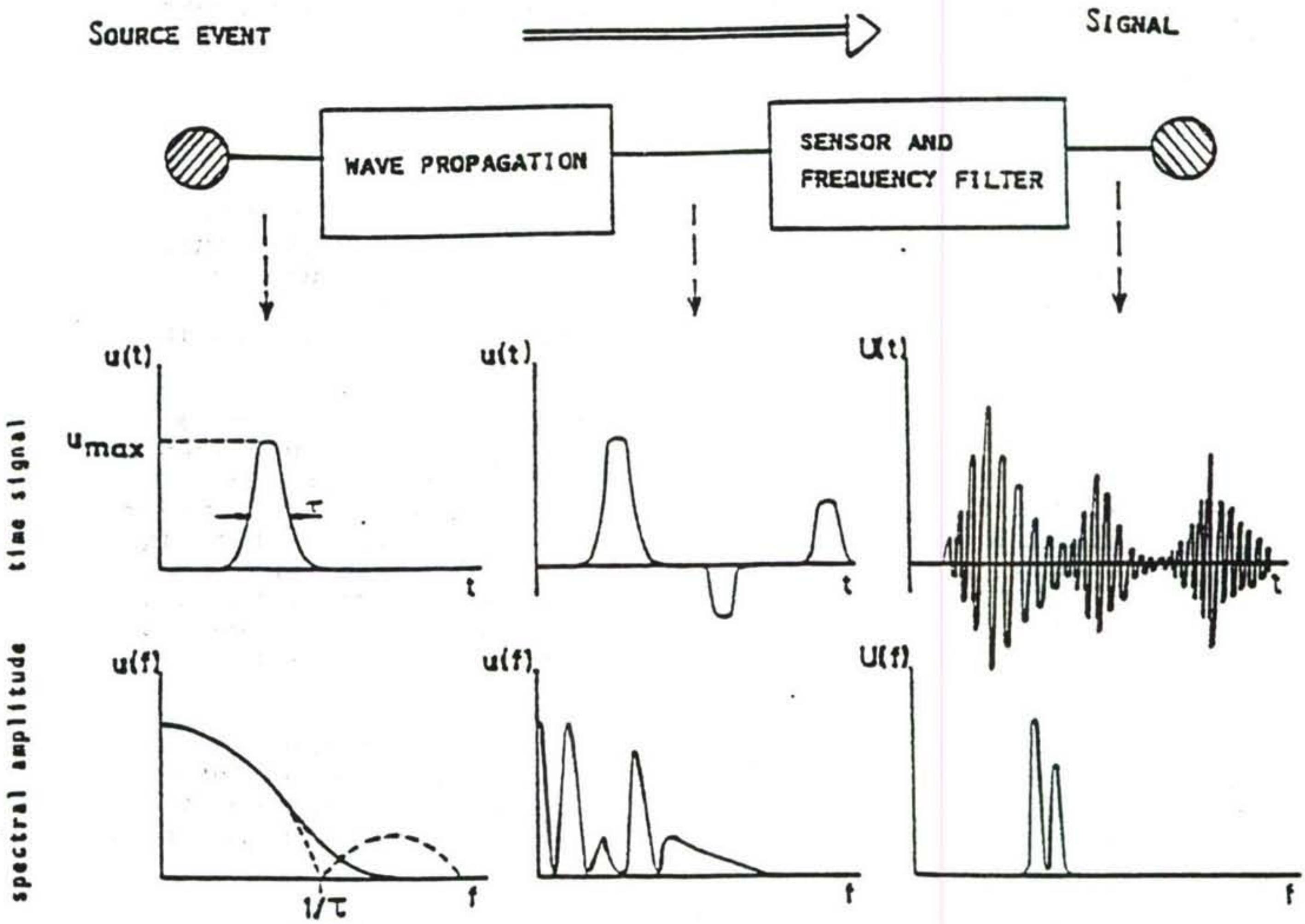

Fig. 2 - Signal Distortion Due to Acoustic Propagation and Transducer Characteristics (Ref. [6])

prepared. Areas where additional studies need to be conducted are identified and reported in Table I. These studies will be needed to determine the usefulness of these sensors in a smart structure.

4.0 THE FIBER OPTIC SMART STRUCTURE SYSTEM

4.1 Fiber Characteristics

The purpose of this fiber optic smart system is to determine the mean time to service and the remaining life of a structure. The system uses an optical fiber embedded in a structure to measure the dynamic strain in the structure. The basic measurement technique correlates measured transmission loss in the optical fiber with strain in the structure.

Fiber loss can occur for several reasons. The first type of fiber loss is due to signal dispersion. Dispersion occurs when a light pulse is spread out as it travels down the fiber. The result is that the signal at the output of the fiber has a wider pulse width than the signal at the input of the fiber. At a high enough frequency, consecutive signals are merged into one, resulting in information loss. In the fibers we tested, there was no measurable dispersion at 1000 Hz. This is significant because it indicates that a structure can be sampled 1000 times per second without dispersion loss.

The second type of fiber loss is due to absorption. Impurities in glass absorb light energy, turning photons into heat. Absorption is very predictable and therefore, the smart system can easily compensate for this expected transmission loss.

The third type of fiber loss is due to scattering. Scattering results from imperfections in the basic structure of the fiber. Unintentional variations in density and fiber geometry occur during the manufacturing process. The losses due to scattering are also very predictable and easily compensated for.

Fiber loss can also occur from micro-bending. Micro-bending can occur when an external force is applied to the structure. The result is that there is an abnormal attenuation of the signal at the output of the fiber. The micro-bending phenomenon generates the transmission loss that is measured by our system. This loss is eventually translated into structure strain. Under test conditions, we were able to show that there was a direct correlation between the amount of tension applied to a specimen and the transmission loss at the output of the fiber.

4.2 The Electro-Optic Interface: Transmitters and Receivers

As previously mentioned, two of the most popular optical fiber light sources are (LEDs) and (ILDs). (See [4] and [11] for more information.) Both sources have their advantages and

TABLE I

COMPARISON AMONG POTENTIAL SENSORS

METHOD	PERFORMANCE DATA	DESIGN REQUIREMENT FOR SMART STRUCTURES	SENSITIVITY	APPLICABILITY	USEFUL LIFE	OTHER PROPERTIES
Fiber Optics	Limited Lab. Data	*	*	In Composites mainly	*	Multi applic
Acoustic Emission	Mainly in metal crack detection	*	*	In metals mainly can be used for composites also	*	Surface Mounting
Strain Gages	Laboratory & Field	*	Relatively Good	All materials	Limited in exiting sensors	Surface Mounting

* Additional Investigations are required to further explore these factors.

disadvantages. The ILD and LED are both solid state devices that can be easily fabricated. Although the ILD can be modulated at a significantly higher rate than the LED, we chose the LED as the source for our transmitter for several reasons. First, the modulation rate of the LED (beyond 20 MHz) is more than adequate for our purposes. Second, the LED is more reliable under adverse environmental and vibrational conditions. Third, it is easier to interface a fiber cable to an LED source than to a ILD source. (Although the coupling efficiency between an LED and a fiber is only about 2%, there are roughly 50 independent positions that can achieve this coupling efficiency. While the coupling efficiency of an ILD can be as high as 50%, the alignment is much more critical.) Fourth, the LED transmitter and receiver are extremely simple designs. Fifth, the LED has an expected lifetime that is ten times longer that the ILD (i.e., an expected lifetimes of 1,000,000 hours vs. 100,000 hours).

4.3 The Smart Structure Design

The Fiber Optic Smart Structure Sensing System is shown in Figure 3. First, an oscillator is used to generate the electrical pulses that are amplified and converted to light pulses. The oscillator ultimately determines the rate at which the strain of the structure is sampled. The transmitted light pulses are measured before they enter the fiber. This transmitted signal is compared with the receive signal to determine the strain in the structure. Next, the light pulse travels along the fiber that is embedded in the structure. Notice that the fiber originates and terminates at the transmitter/receiver board. This is done intentionally to compensate for any signal variations that might result from environmental changes. Since both the transmitter and receiver are in the same environment all environmental problems are eliminated.

The received light pulse is converted into an electrical pulse. The amplitude and width of this pulse are compared with the transmitted pulse. Any amplitude degradation and/or pulse width expansion are translated by the system into strain. This process is repeated up to 1000 times per second.

The strain data collected from the Smart Sensor System is used to build a statistical database. Also, the data is sent to the on-board artificial intelligence (AI) module in which an expert system is used to interpret the data. The expert system will predict the mean time to service and the remaining life of the aircraft.

5.0 APPLICATION TO THE AIRCRAFT LIFE-CYCLE MANAGEMENT

The preceding sections described the mechanics of a simple fiber optic smart structure. In a much larger scale, the smart structure is expected to handle many more tasks as described below.

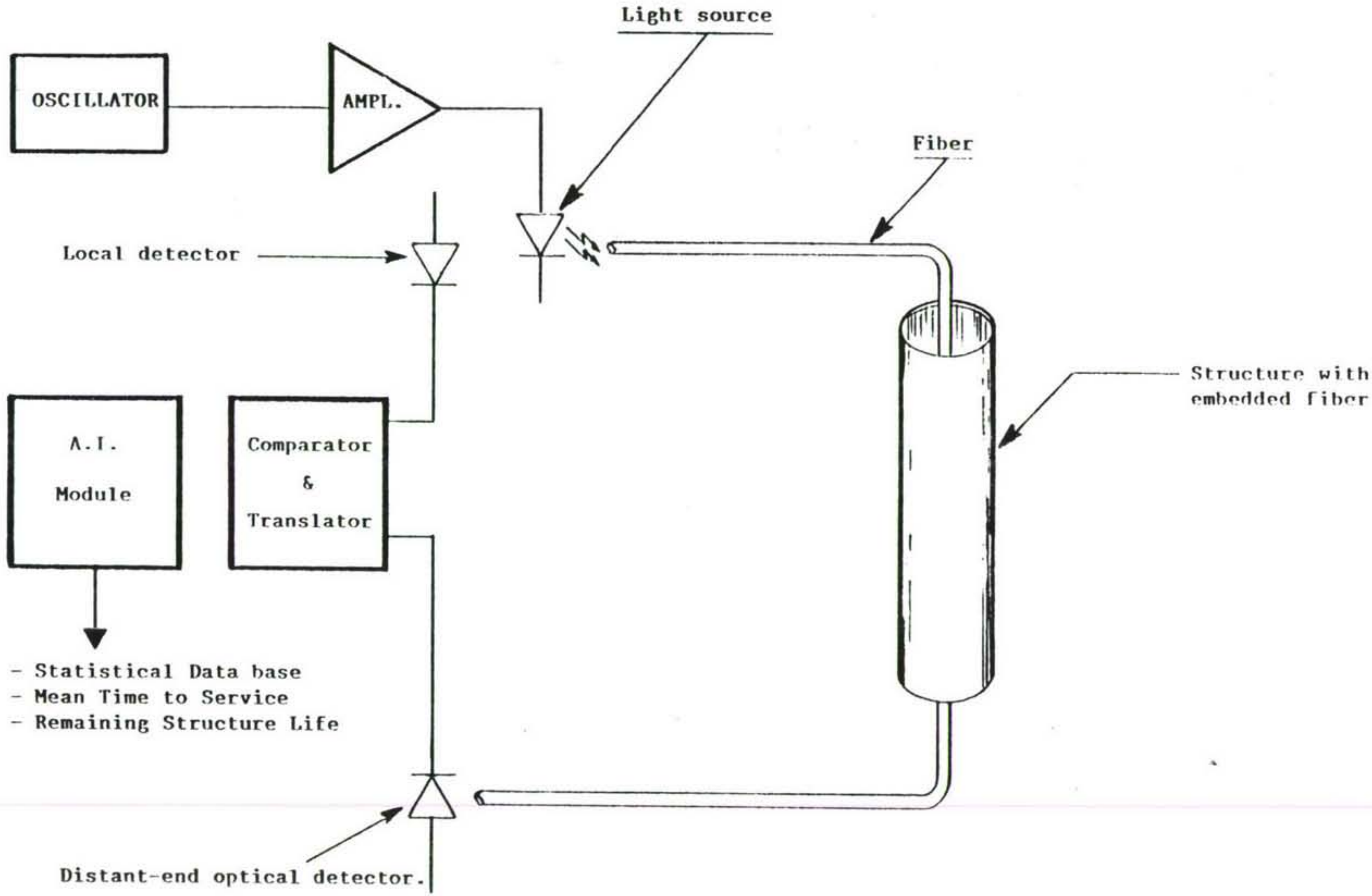

Figure 3 - A Fiber Optic Smart Structure Sensing System.

- o Advanced warning on the condition of a severely stressed component in the aircraft;
- o Advanced structural analysis (such as the finite element method) to determine the extent of damage that occurred on the critical components, the damage growth and the damage transfer to other components;
- o Calculation of remaining useful life of a critical component using statistical analyses and expected damage growth data;
- o Development of summary reports on the distribution of incoming data (i.e., internal stress conditions) for observation, examination and/or further analyses; and,
- o Predicting the maintenance demands of the aircraft, and periodically updating the maintenance and repair records as they become available.

Ideally, the smart structure, as described above, will have the general organizational diagram shown in Fig. 4. The AI module continuously communicates with the data bus and sensing module, and consults with the knowledge-base for making necessary decisions on the data received from the sensing module. Through its many units, the AI module can perform the many tasks described above. The organizational diagram in Fig. 4 depicts the base-line architecture in an ideal airframe smart structure designed for structural integrity programs. This provides a permanent built-in health-monitoring system which can improve performance of the aircraft, survivability, flight safety and life cycle cost. Problems associated with fatigue and fracture, which normally occur abruptly and without advanced warning can effectively be controlled and predicted using many special features of the smart structure shown in Fig. 4.

The capability of the smart structure to conduct rapid structural analyses is especially helpful in maintaining the history of crack development and crack growth as the flight activities of the aircraft advance. The incoming flight data can be used to predict the crack growth and to generate advanced warning should a crack becomes critical. Methods for investigating the crack growth using the statistics of stresses induced in structural components can be used for this purpose. The fatigue properties of specific critical components of the aircraft are then needed and must be in the knowledge base of the program for the analysis. The procedure for evaluating damage growth and predicting the remaining useful life of the aircraft are shown in Fig. 5. As seen in Fig. 5, the accumulated input data from the aircraft is used to establish the stress histories required for crack growth analysis of aircraft critical structural components. Computer programs capable of conducting such advanced probabilistic analyses are the essential elements of the smart structure AI module.

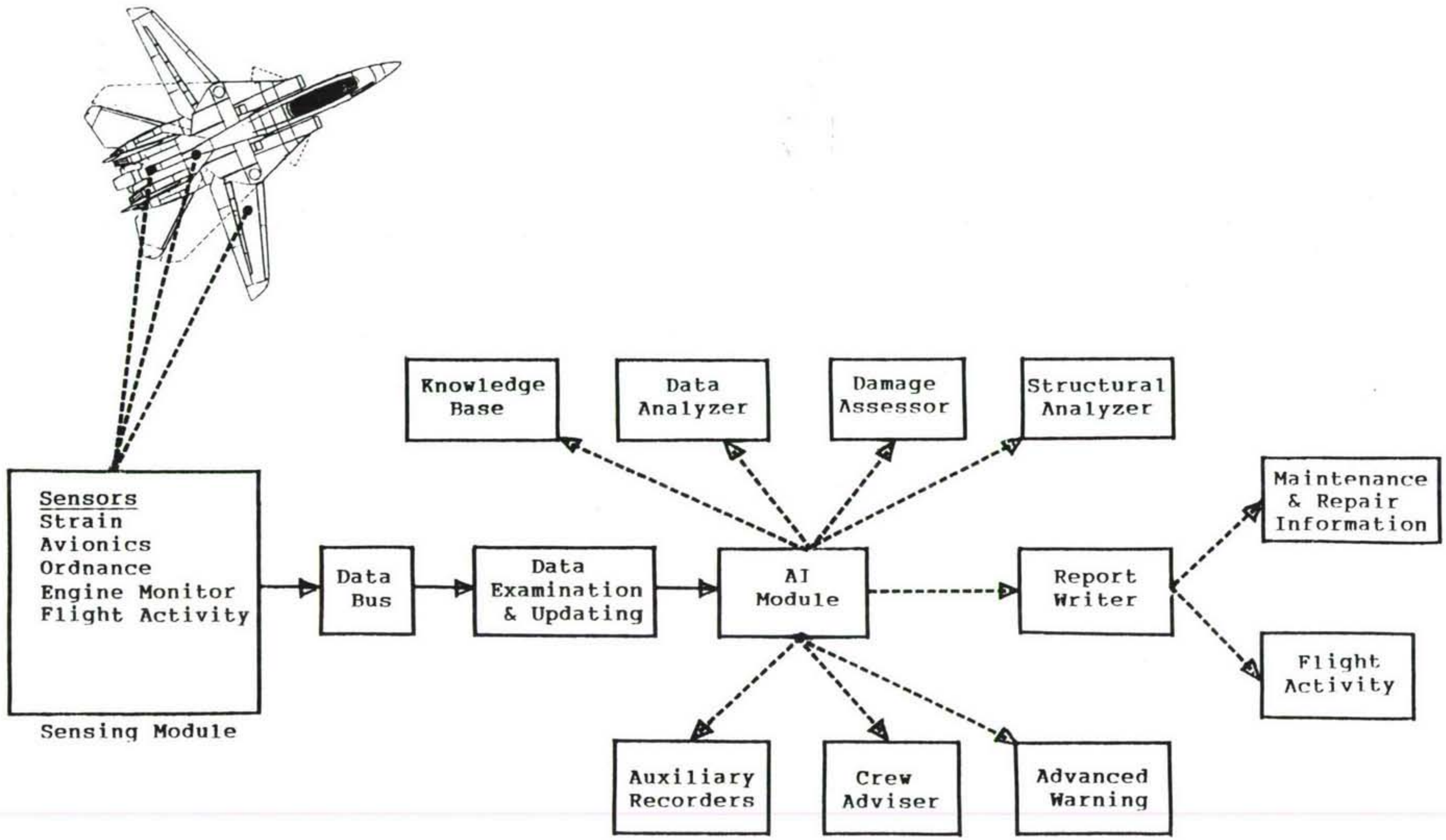

Fig. 4 - Smart Structure Architecture

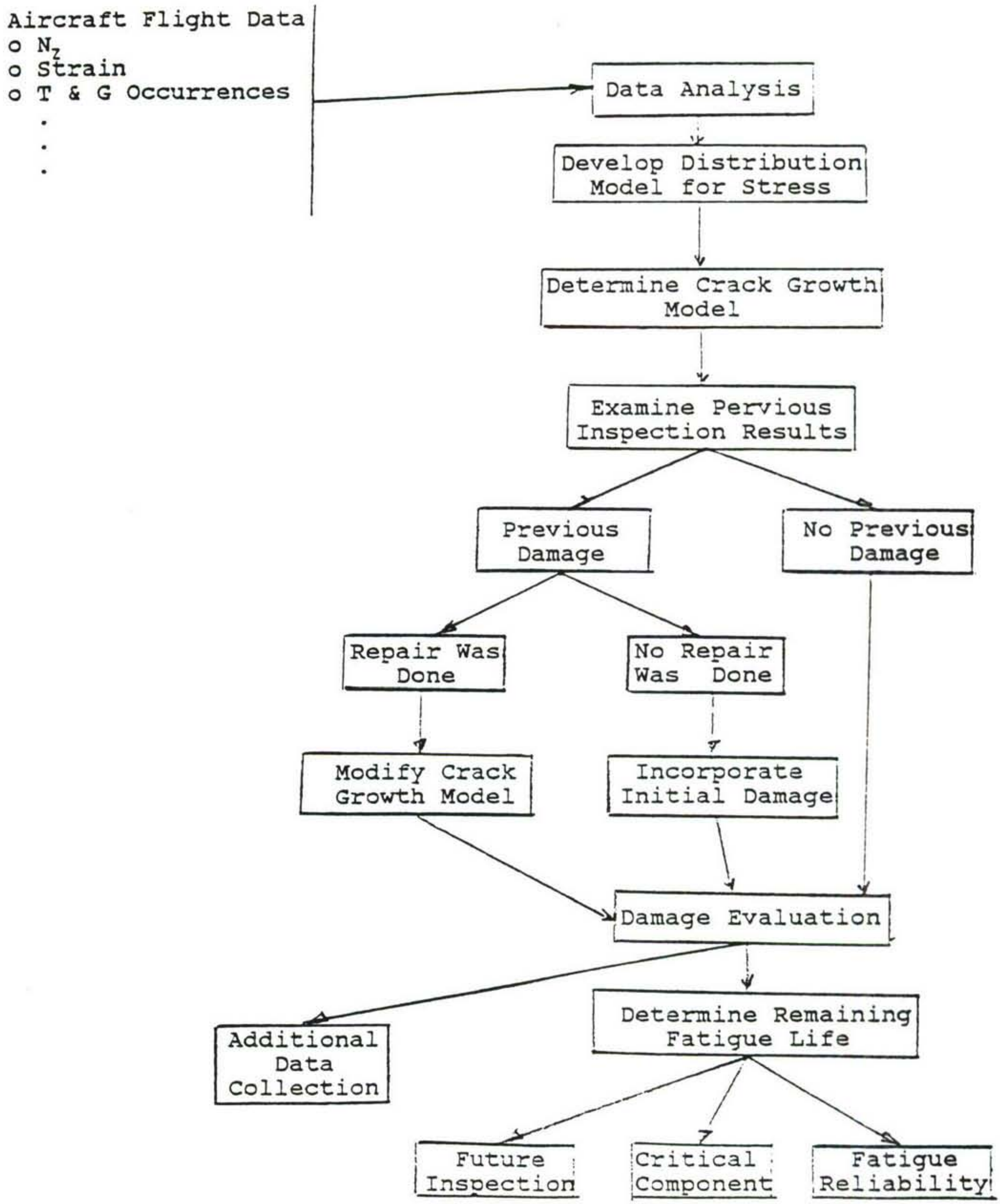

Fig. 5 - Aircraft Structural Components Fatigue Analysis

6.0 CONCLUSIONS

The application of the smart structure concept to the aircraft structural integrity programs was reviewed. Different types of sensors with potential to be used as the sensing module of an ideal airframe smart structure were identified. The advantages of each sensor type and its specific application to the type of material were examined and reported. A sample fiber optic smart structure made of simple components and electronics was discussed. Problems associated with light transmission, attenuation and loss were examined and reported. The basic requirements of a more comprehensive smart structure capable of monitoring the status of the internal strain in the aircraft structural components were described. The need for additional studies both in the area of sensor technology and artificial intelligence computer programming was described. The advantages of current methods of aircraft structural tracking and their use in building the first generation smart structures were also examined and discussed in the paper.

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1989 USAF Structural Integrity Program Conference

ANALYSIS OF CRACKING IN THE 66 PERCENT WING SPAR OF THE T-38 AIRCRAFT

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San Antonio, Texas
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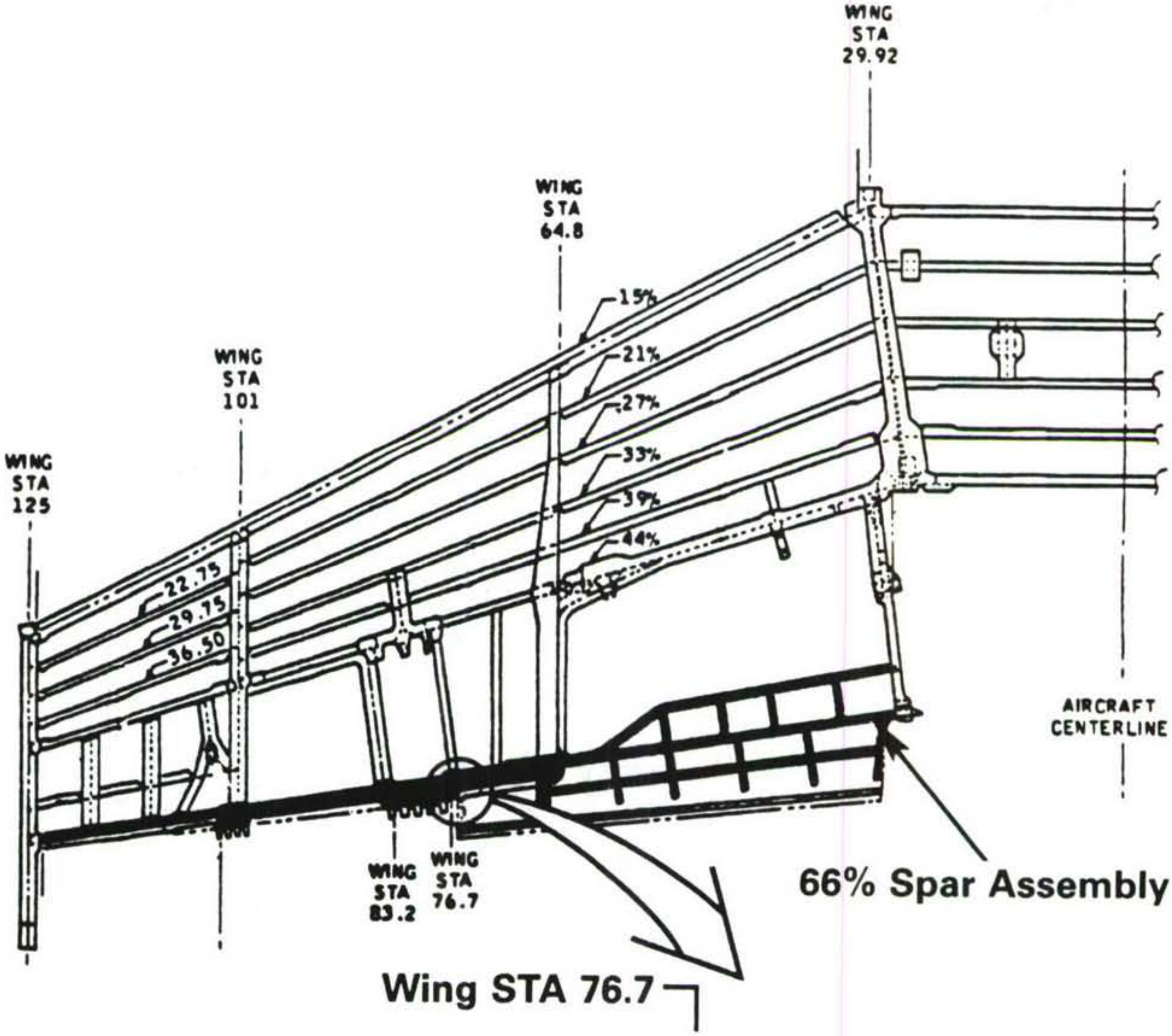
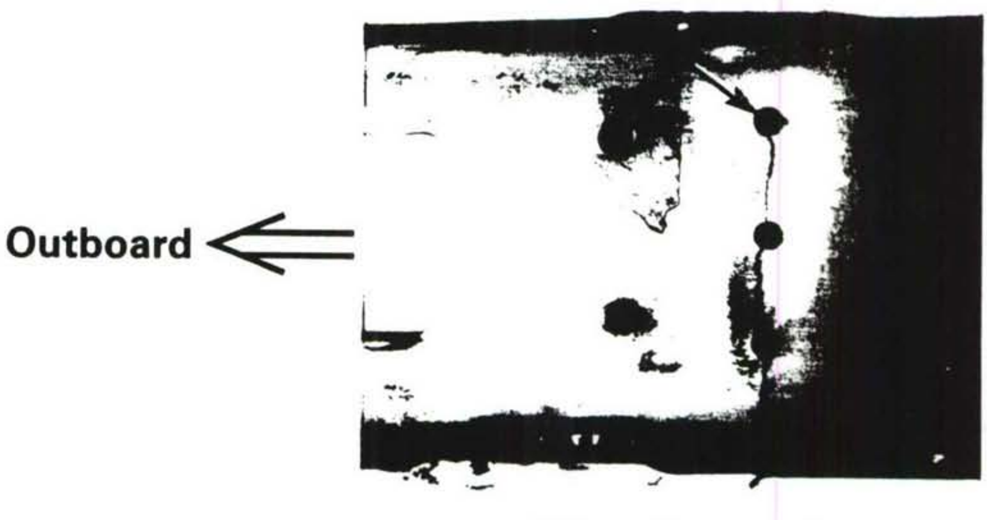
T-38 66% WING SPAR ANALYSIS

INTRODUCTION

- **STRUCTURAL FAILURE EVALUATION**
  - PREDICTION
  - SOLUTION
    - Material Mod
    - Geometric Mod
- **TECHNOLOGY SYMBIOSIS**
  - OPERATIONAL FLIGHT MEASURED DATA
  - STATISTICAL/PROBABILITY ANALYSIS
  - FRACTURE MECHANICS

T-38 66% WING SPAR ANALYSIS

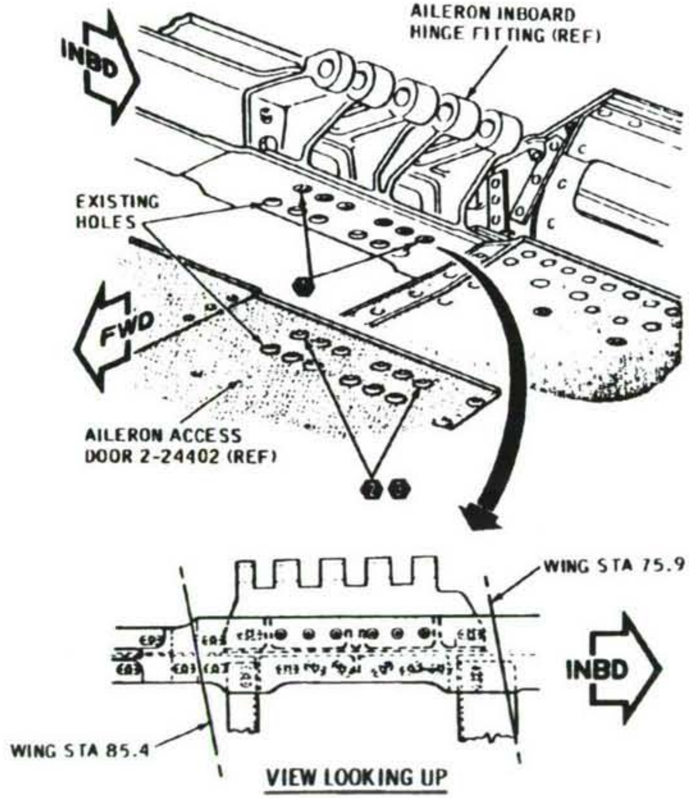




View Forward

T-38 66% WING SPAR ANALYSIS

T-38 66% SPAR STRAIN GAGE



T-38 66% WING SPAR ANALYSIS

USAF CONCERNS

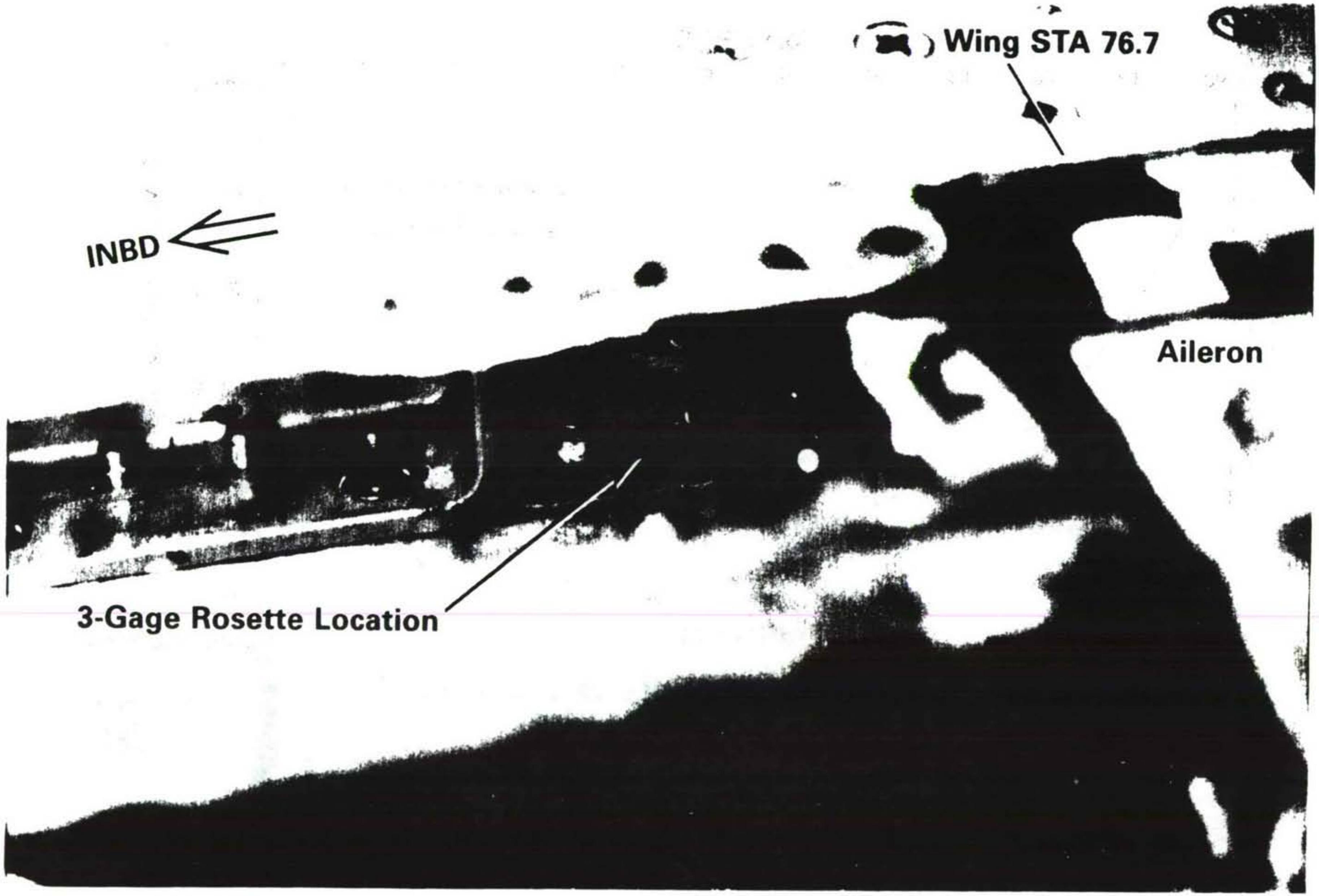
- **CRACKED STRUCTURE**
  - NOT SAFETY OF FLIGHT
  - MAINTENANCE REQUIREMENT
- **COSTS**
  - INSTALLATION
  - PARTS

T-38 66% WING SPAR ANALYSIS

T-38 MANEUVER SPECTRA UPDATE

- COLLECT MANEUVER SPECTRA AND OPERATIONAL USAGE PROFILE DATA
- MEASURE STRAIN LEVELS AT CRITICAL A/C STRUCTURE
- MEASURE SURFACE DEFLECTIONS

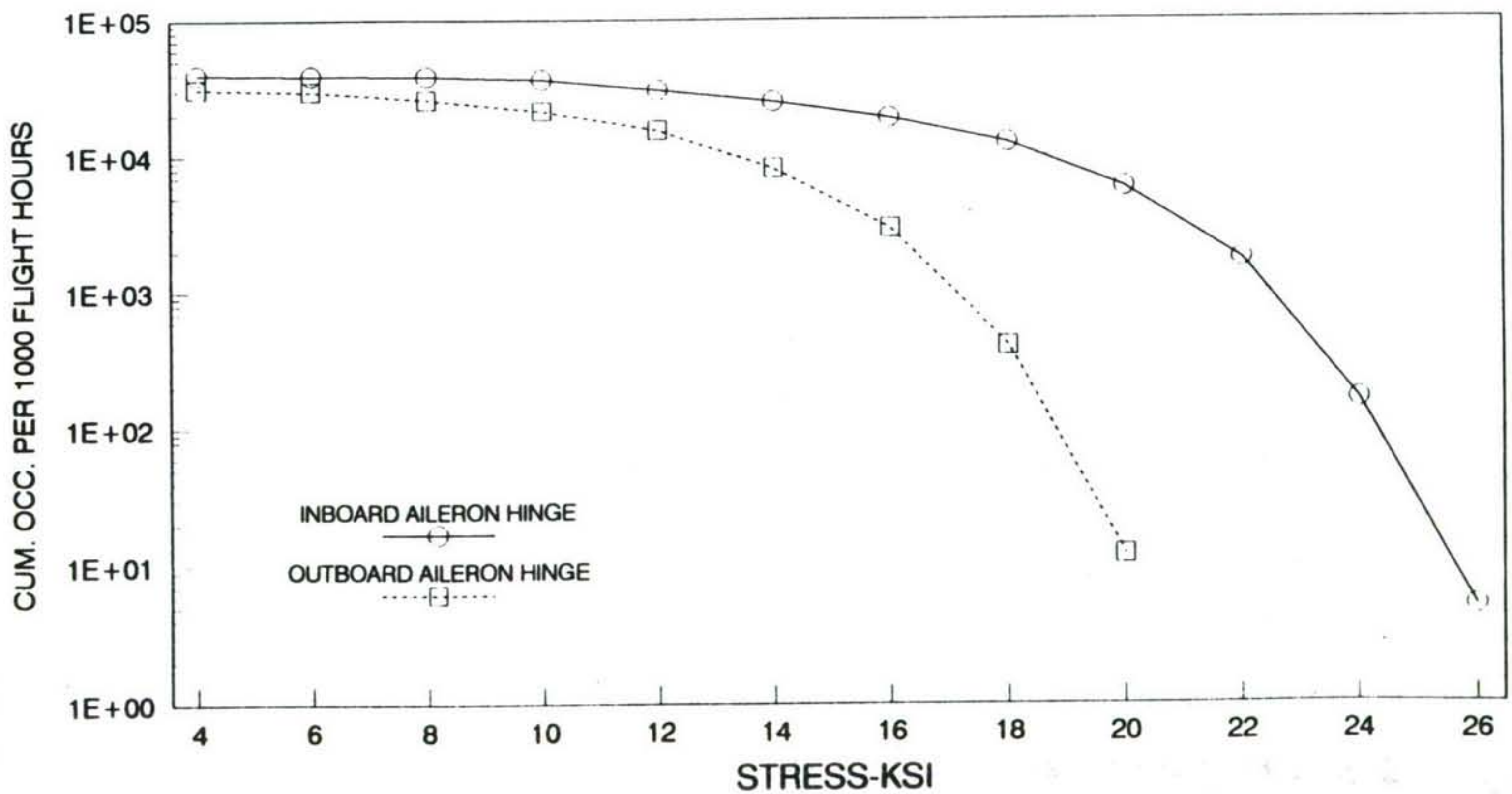
T-38 66% WING SPAR ANALYSIS



T-38 66% WING SPAR ANALYSIS

T-38 66% SPAR STRESS SPECTRUM

LIF USAGE



T-38 66% WING SPAR ANALYSIS

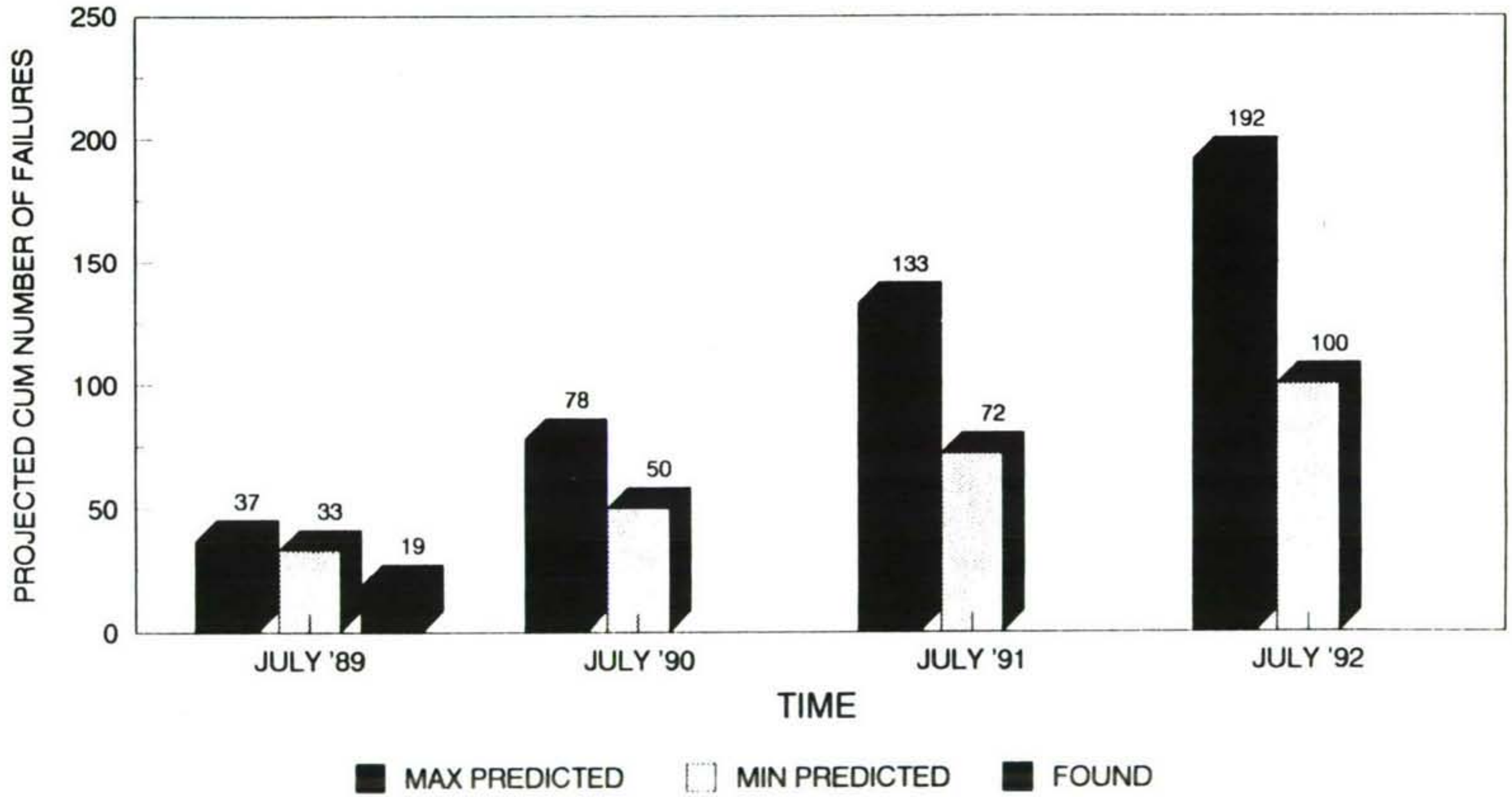
FAILURE PROBABILITY ANALYSIS
LEAD-IN-FIGHTER USAGE

- **PERIODIC INSPECTIONS**
  - 450 FLIGHT HOURS
  - X-RAY
- **AS OF JULY 1989**
  - 19 LIF FAILURES
  - 7 LEFT, 12 RIGHT
  - 1731 - 3040 FLIGHT HOURS
- **WEIBULL ANALYSIS**
  - COMMON MODE FATIGUE FAILURE
  - TIME OF SERVICE
    - Failed
    - Unfailed
  - ASSUMED PROJECTED USAGE OF 25 FLT. HRS PER MONTH

T-38 66% WING SPAR ANALYSIS

PROJECTED T-38 66% SPAR FAILURES

LIF USAGE



T-38 66% WING SPAR ANALYSIS

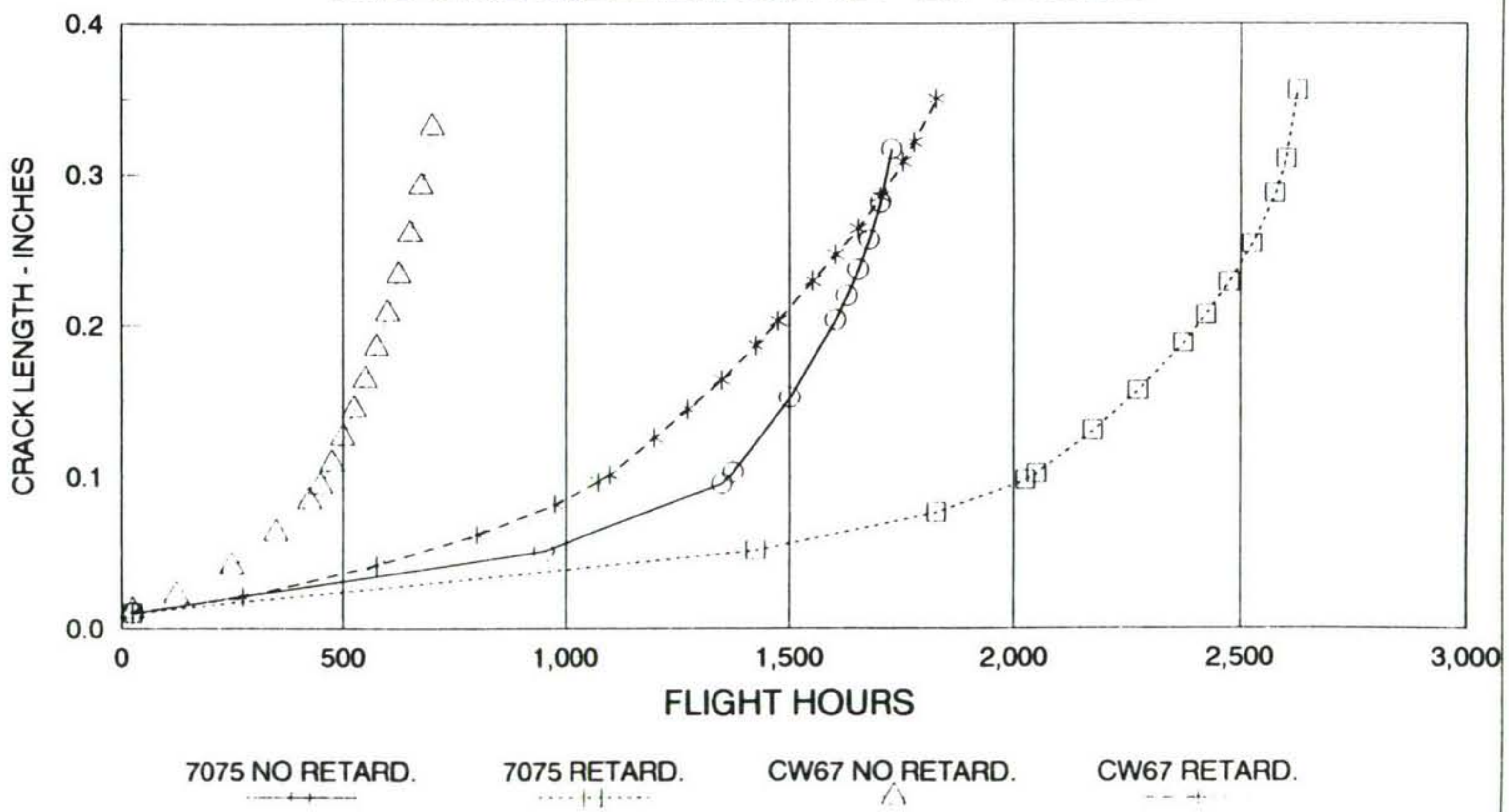
CRACK GROWTH ANALYSIS

- **MATERIAL COMPARISON ONLY**
  - CURRENT MATERIAL
  - POWDER METAL ALLOY (CW67)
- **CRACK GROWTH MODEL**
  - FASTENER HOLE IN LOWER SPAR FLANGE
  - QUARTER CIRCULAR INITIAL FLAW
  - TABULAR CRACK GROWTH RATE DATA
    - Northrop Reports NOR 88-28, 88-61, 89-55
    - R = 0.1
  - WITH/WITHOUT RETARDATION
- **FLIGHT MEASURED STRESS**

T-38 66% WING SPAR ANALYSIS

CRACK GROWTH RESULTS

INBOARD AILERON HINGE - LIF USAGE



T-38 66% WING SPAR ANALYSIS

CONCLUSIONS

- **EXTENSIVE LIF FLEET SPAR FAILURES ANTICIPATED**
- **STRUCTURAL REDESIGN/MOD REQUIRED**
- **LARGE ATC USAGE FLEET THREATENED**
  - 679 AIRCRAFT (1358 SPARS)
  - CURRENT ATC USAGE FAILURES AS OF OCT. 1989
    - 3339 Flight Hours (Williams AFB)
    - 4619 Flight Hours (Vance AFB)

T-38 66% WING SPAR ANALYSIS

RECOMMENDATIONS

- **CONTINUED USE OF TECHNOLOGY SYMBIOSIS**
  - ATC FLIGHT MEASURED DATA (ONGOING)
  - USE FRACTURE MECHANICS TO ESTABLISH SEVERITY RELATIONSHIP WITH LIF USAGE
  - ATC FAILURE PROBABILITY DISTRIBUTION
    - LIF Inspection Experience
    - Analytical Comparison To Severe Usage Environment (LIF)
    - Weibull Analysis

Report WU/CCM - 89/2

NONLINEAR MODELS FOR FASTENED STRUCTURAL CONNECTIONS BASED ON THE P-VERSION OF THE FINITE ELEMENT METHOD

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Israeli Air Force/Washington University

B. A. Szabó,

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December 1989

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December, 1989

1989 USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE

SAN ANTONIO, TEXAS

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NONLINEAR MODELS FOR FASTENED STRUCTURAL CONNECTIONS BASED ON THE P-VERSION OF THE FINITE ELEMENT METHOD

J. Bortman and B. A. Szabó

Washington University

1. INTRODUCTION

Structural fatigue cracks are initiated and propagated in areas of high stress concentration, such as fastened joints. Often one or more fastener holes are sites of crack initiation. Hence, a large amount of effort is devoted in the aeronautical industry to the evaluation of the fatigue life of fastened joints. The first step in this analysis is determination of the load distribution between the fasteners and the stress field. Because of the complexity of the problem, certain modeling assumptions have to be made. In finite element models fasteners are usually idealized as one-dimensional springs or rigid links which connect nodal points between two elastic bodies. The elastic bodies are usually idealized as membranes or plates. While such practices are intuitively plausible, they are inconsistent with formulations based on the principle of virtual work, and are therefore conceptually incorrect. As a result, it will be demonstrated that the computed fastener forces and the stresses in the vicinity of the fasteners are entirely discretization-dependent. This gives the motivation to formulate a new model.

An efficient and convenient technique is therefore suggested for modeling load transfer through fasteners, based on the p-version of the finite elements. The interaction between the fastener and the two-dimensional elastic body are

modeled by normal displacements imposed on distributed springs. Friction is imposed as a weak condition (external tractions). Each fastener is represented by a nonlinear relation between the transferred force and the relative displacements. This relation may be obtained from a detailed three-dimensional analysis or from tests. After condensing out all linear degrees of freedom, the nonlinear equations are solved.

2. PROBLEM DEFINITION AND IDEALIZATION

The aim of this study is to investigate the structural and strength analyses of two flat, finite width, sheets connected by n fasteners and loaded by in-plane tractions (see for example Figure 1). The sheets may include cracks and are restricted to be isotropic or orthotropic with specified material properties. Each fastener may be of a different geometry or material type and may be installed with a different clearance. Friction exists between the fasteners and the hole surface. The loading, either by application of forces, or imposition of displacements, may result in nonlinear response which will be considered.

These joints are complicated structural assemblies, detailed analysis of which requires consideration of a three dimensional problem involving nonlinear effects such as contact, friction, material properties and mode of installation. Although it is possible to construct such a model, the large amount of computational effort (CPU time and memory space) required makes it prohibitive, especially in the case of a joint that includes many fasteners. Instead, a more realistic approach based upon justifiable engineering simplification is employed, and only in the final step, when a detailed local analysis is needed, should construction of a fully three-dimensional analysis be considered.

In the first idealization step it is assumed that the stress, strain and displacement fields within the sheets are two-dimensional (i.e. plane stress). This assumption is questionable for the region near the sheet boundaries where clamping forces and out of plane deformation sometimes occur. This effect will be taken into account partially when defining the fastener stiffness which is assumed to be highly influenced by three-dimensional properties. Nevertheless, the model is proper for those cases where these displacements are small. Based on finite element studies it is assumed that, as a result of stress concentration and bearing loads, yielding occurs in the plate in the neighborhood of the fasteners. The overall behavior of the plate is almost linear and only over the bore are the displacements nonlinear (when the load is increased). This nonlinear effect will be taken into account in formulating the fastener element. It is further assumed that the friction between the fastener and the hole face is restricted to the "slip-region" where $|\tau_t| = \mu|\sigma_n|$. This assumption is based on the results presented in [1]* where, in practical cases, the "no-slip" zone was found to be restricted to a very small range (less than $5°$ of the hole perimeter) which will be neglected in the current study.

The fastener itself is assumed to behave as a special connector between two different holes; one in the upper plate and the other in the lower plate. The stress distribution inside the fastener is not of interest. The force transferred by the fastener is assumed to be a nonlinear function of the relative displacements of the upper and lower plates. The nonlinear function describing the force transferred by the fastener depends strictly on material and geometrical properties. The

* The numbers in parentheses in the text indicate references in the Bibliography.

fastener is assumed to be initially at the center of the hole, and so the clearance is defined as the initial radial gap.

As a result of tests, a general concept of the action of fastened joints was formulated. The behavior, under load, of a perfectly fabricated, two-rivet joint (see Figure 2) may be considered in four stages. In the first stage, static friction prevents slip. In the second stage, the load is greater than the static friction resistance and the joint slips until the rivets come into bearing. In the third stage, rivets and plates deform elastically so that the load-relative displacement relation is almost linear. In the fourth stage, yielding of plates or rivets, or both occurs and the relation becomes nonlinear until plastic failure occurs. It was also noted that the range of load over which the first stage extends is affected by the amount of friction between the plates.

3. THE CURRENT MODELING PRACTICE

3.1 The "Line-Spring" Model

The most common approach used in the aeronautical industry is based on the assumption that the fastener behaves like a spring. In that case, the fastener is idealized as a one-dimensional spring which connects two nodal points of two different two-dimensional elastic bodies. As previously stated, this approach is conceptually incorrect. When the line spring is attached to a two-dimensional domain, it introduces a point load into the two-dimensional domain. Such a load causes the total potential energy to be unbounded from below. Hence, it is not possible to use this model when the formulation is based on the principle of virtual work. In order to examine the error that is introduced through use of this modeling approach, the following test case was solved. The computations

were performed with the computer program *ADINA* [2] which is based on the h-version of the finite element method.

A two-dimensional plane stress rectangular plate is loaded in plane by a uniform load (see Figure 3). Two fasteners are simulated by two springs of the same stiffness ($K1$ and $K2$), which are fixed on one side and connected to the plate on the other side. The plate dimensions are given in Figure 3. Using symmetry, only half of the model was considered. Four different meshes were constructed:

a) 16 linear elements with 49 degrees of freedom.

b) 100 linear elements with 231 degrees of freedom.

c) 400 linear elements with 861 degrees of freedom.

d) 1600 linear elements with 3321 degrees of freedom.

Each two-dimensional mesh was changed three times to obtain three different element sizes next to the springs by controlling a parameter Q, which is defined as the relation between the element size next to the right spring ($K2$) divided by the remote element size. The three different Q values are: unity (no distortion), 0.5 and 0.1. A typical mesh is presented in Figure 4. In the next step two different Q levels were defined, one for each spring: $Q1 = 0.01$ and $Q2 = 10.$, such that the first fastener is surrounded by small elements and the second by large elements. The results are presented in Figure 5. Reducing the size of the element next to the second spring ($K2$) appears to reduce the amount of load transferred by that spring. In this case no convergence was obtained, which confirms the theoretical expectations. By changing the mesh, the relation $F2/F1$ is changed from 1.7 to 0.9, a difference of almost 200%! It is concluded that the "line spring" model of the h-version gives poor results and should not be used to simulate fasteners before more research is conducted to define its capabilities.

3.2 The "Distributed-Spring" Model

Very often the fasteners are assumed to behave as a finite number of radial links. In order to study the accuracy of that model (i.e. how many degrees of freedom are required to well represent the stress distribution around the fastener hole) and in order to compare the h-version approach with the p-version approach, two two-dimensional models were built. First, using a p-version code, *PROBE* [3] and second, using a h-version code *ADINA* [2]. In both cases the same test problem was solved: A two-dimensional plane strain rectangular plate that includes a hole was loaded in plane by a uniform load applied to the plate edge and reacted into a pin-load (see Figure 6). Using symmetry, only half of the model was considered. The same material data that were used in the line spring test case are used again. The commercial finite element code *PROBE* [3] was used to construct a p-version model. The model includes eight elements and was run for $p = 1$ to $p = 8$. The mesh is presented in Figure 7. The pin reaction is simulated by a normal distributed spring. The results that were obtained using this model are presented in Figure 8.

The computer program *ADINA* [2] was used to construct an h-version model of the same problem presented in Figure 6. The fastener was modeled as "concentrated links". Four different meshes were built:

a) 16 linear elements with 48 degrees of freedom.

b) 100 linear elements with 260 degrees of freedom.

c) 400 linear elements with 883 degrees of freedom.

d) 1600 linear elements with 3362 degrees of freedom.

Two typical meshes are shown in Figure 9. All meshes were geometrically graded toward the hole. Two important issues were raised during the study: how should one choose the stiffness of the individual link? and what is the amount of error

introduced by a wrong choice? Therefore, two cases were considered. In the first, $K \cdot n$ was kept constant where K is the individual link stiffness and n is the number of links. In the other case K was kept constant. In both cases the maximum stress was plotted against the number of degrees of freedom as shown in Figure 8. The *PROBE* results are also presented in this Figure and comparison of the results make it clear that the h-version model converges relatively slowly. Nonetheless, both the h-version and the p-version converge to the same value. It is concluded that the "distributed links" model of the h-version converges slowly relative to the p-version. Therefore, with these problems it is more efficient to employ the p-version approach.

4. MATHEMATICAL FORMULATION

In the idealization process two different groups of structural elements are considered. First, the two sheets are defined as the first domain (Ω_1) in which the linear two-dimensional elasticity relations are valid. Second, the fasteners (Ω_2) are described by a combination of the elastic foundation theory with nonlinear relations (see Figure 10). Finally, the overall model is assembled and solved. The following sections include a description of the equations in each part of the model.

4.1 The Two-Dimensional Domain (Ω_1)

Since conventional two-dimensional p-version finite element formulation is employed, only basic theory is reviewed based on [4]. The domain (Ω_1) is divided into finite elements through a meshing process. Then the polynomial basis functions are defined on a standard element. These elements are then mapped by appropriate mapping functions onto the "real" elements. The displacement

components (u_x and u_y) can be expressed in terms of the shape functions $\Phi_i(x,y)$:

$$u_x(x,y) = \sum_{i=1}^{n} a_i \Phi_i(x,y) \tag{1}$$

$$u_y(x,y) = \sum_{i=1}^{n} a_{n+i} \Phi_i(x,y) \tag{2}$$

where a_i are the amplitudes of the basis functions. In the current study, the computer program *PROBE* is used to analyze the two-dimensional domain. In that case, hierarchic shape functions based on the Legendre polynomials and the blending function method are used for the mapping. Then by defining:

$$[K] \stackrel{\text{def}}{=} \iint_{\Omega 1} \left(([D]\{\Phi\})^T [E][D]\{\Phi\} \right) t\,dx\,dy \tag{3a}$$

and

$$\{r\} \stackrel{\text{def}}{=} \int_{\Gamma_T^n} \{\Phi\}^T \{T_n\} t\,ds \; + \; \int_{\Gamma_T^t} \{\Phi\}^T \{T_t\} t\,ds, \tag{3b}$$

where $[E]$ is the matrix of the material constants, Γ is the boundary, and where [D] is an operator matrix:

$$[D] \stackrel{\text{def}}{=} \begin{bmatrix} \frac{\partial}{\partial x} & 0 \\ 0 & \frac{\partial}{\partial y} \\ \frac{\partial}{\partial y} & \frac{\partial}{\partial x} \end{bmatrix}, \tag{4}$$

the virtual work relation becomes:

$$[K]\{a\} = \{r\}, \tag{5}$$

where $[K]$ is the unconstrained stiffness matrix. After imposing the kinematic boundary conditions and obtaining the constrained stiffness matrix, equations (5) are solved for $\{a\}$. As the p-version approach is employed, this procedure is repeated eight times with different hierarchic shape functions, which makes it possible to check for convergence of the required parameters.

4.2 The Fastener ($\Omega 2$)

The essential issues in modeling the fasteners are: first, the transferred load needs to be distributed over the bore in a realistic way, and second, the magnitude of that load should be related to the relative displacement (between the upper and lower plates) by a specified equation that is obtained from a test or a calculation. So, in order to formulate the fastener relations, these two concerns are involved. First, distributed springs are attached to the perimeter of the hole on one side and to a rigid disc on the other (see Figure 10). The assumption that this modeling technique allows for the correct stress distribution was verified by solving representative test cases. This modeling procedure is used for both plates. Second, a nonlinear relation between the relative displacements of the upper and the lower discs to the load transferred by the fastener is specified.

The distributed spring may be used to connect two different two-dimensional domains. In our case, they are used to connect the two-dimensional domain, ($\Omega 1$), to the rigid disk. Following relation is specified at each point of the fastener hole boundary Γ_s (Γ_s is the contact line between ($\Omega 2$) and ($\Omega 1$)):

$$K_{nn}(\Delta_n - u_n) = T_n \tag{6}$$

where K_{nn} is the distributed spring constant, u_n and Δ_n are the plate boundary and the rigid disc normal displacements, respectively. T_n is the normal traction acting on the hole perimeter. Relations (3) are modified to account for the spring constraint. With these modified definitions, the rest of the formulation presented before remains the same.

Consider a single fastener that connects two plates together. The relation between the relative displacement, δ, of the upper and the lower rigid discs and the load transferred, F, is assumed to be a known nonlinear equation. Two

different force-displacement relations are considered. First, a three parameter model:

$$F(\delta) = \begin{cases} 0 & \text{for } \delta \le \delta_0 \\ A(\delta - \delta_0)^n & \text{for } \delta > \delta_0 \end{cases} \tag{7}$$

where A and n are calibration constants (in practical cases n is limited to be in the range of $0.4 \le n \le 1.0$), δ_0, the third constant, is associated with the initial clearance between the fastener and the hole. This model cannot describe the behavior of load-displacement relation in the neighborhood of δ_0 well enough since this is a singular point; in other words, the derivative of the load with respect to the relative displacement is not defined at that point (see Figure 11a). So, a second approach is suggested, a six parameter model that is valid for $\delta > 0$:

$$F(\delta) = F_0 \frac{(\delta - \delta_1)(\delta - \delta_2)\delta}{(\delta_0 - \delta_1)(\delta_0 - \delta_2)\delta_0} + F_1 \frac{(\delta - \delta_0)(\delta - \delta_2)\delta}{(\delta_1 - \delta_0)(\delta_1 - \delta_2)\delta_1} + F_2 \frac{(\delta - \delta_0)(\delta - \delta_1)\delta}{(\delta_2 - \delta_0)(\delta_2 - \delta_1)\delta_2} \tag{8}$$

where F_0, F_1, F_2, δ_0, δ_1 and δ_2 are the six calibration constants (see Figure 11b). The advantage of this approach is that the function is smooth for all $\delta > 0$ and hence doesn't have singular points.

4.3 Friction

Friction causes shear traction on the bore. Since the contact stress, σ_{rr}, can be often described as a cosine function, it is assumed only for friction calculations that the contact stress is represented by a cosine-distribution. Under this assumption the load transferred by the fastener, F, can be related to the friction amplitudes (Λ). This relation can be developed as follows:

$$F = td \int_0^{\frac{\pi}{2}} \sigma_{rr} \cos\theta \, d\theta \tag{9}$$

where d is the hole diameter, θ is a local coordinate defined in Figure 12. Substituting the assumed relation $\sigma_{rr} = \sigma_0 \cos\theta$ (where σ_0 is a constant) for σ_{rr} yields:

$$F = \sigma_0 \frac{td\pi}{8} \tag{10}$$

and similarly for the frictional load:

$$\sigma_{r\theta} = \mu\sigma_{rr} = \mu\sigma_0 \cos\theta \tag{11}$$

where Λ is defined to be the frictional amplitude $\Lambda \stackrel{\text{def}}{=} \mu\sigma_0$. Combining this definition with (10) yields the rquired relation:

$$\Lambda = \frac{8\mu}{td\pi} F. \tag{12}$$

4.4 Construction of the Set of Equations

4.4.1 Data management

Most of the components of the model behave in a linear manner, the plates ($\Omega 1$) and the distributed springs which are part of ($\Omega 2$). Hence, it was decided to condense out the linear degrees of freedom and then to solve for the nonlinear degrees of freedom separately, employing an iterative process.

In the first step, the two plates are divided into a finite element mesh. Distributed springs are attached to the fastener hole boundaries. These models are used to obtain the linear coefficients that describe the linear part of the model. All rigid disks and loaded boundaries are fixed, only one rigid disc of the i^{th} hole is constrained to move a unit displacement. The reactions acting on each of the holes and on the boundaries are extracted using a superconvergent technique. In that way two coefficient matrices are obtained: $S_{uu}(i,j)$ and $S_{dd}(i,j)$ where the subscripts u and d stand for the upper and lower plates, respectively, j is the fastener hole number in which a unit displacement was imposed and i is the index of the fastener hole for which the transferred load was extracted (see Figure 13).

It is assumed that the external load is applied to the lower plate. So, two other matrices are constructed, $S_{ee}(i,j)$ and $S_{ed}(i,j)$, to represent the relation between the lower plate displacements and the external loads. Based on the

above definitions the following relations may be written for an arbitrary external loading vector, $\{P_e\}$, where frictional forces are omitted:

$$\{F_u\} = [S_{uu}]\{U_u\} \tag{13}$$

$$\begin{Bmatrix} \vec{P}_e \\ \vec{F}_d \end{Bmatrix} = \begin{bmatrix} [S_{ee}] & [S_{ed}] \\ [S_{de}] & [S_{dd}] \end{bmatrix} \begin{Bmatrix} \vec{D}_e \\ \vec{D}_d \end{Bmatrix} \tag{14}$$

where $F_i = F_{d|i} = F_{u|i}$ is the magnitude of the force transferred by the i^{th} fastener, $U_i = U_{u|i}$ and $D_i = D_{d|i}$ are the displacements of the upper and the lower rigid discs, respectively, $(i = 1, \ldots, n)$. $P_j = P_{e|j}$ and $D_{e|j}$ are the applied force and the displacement at the j^{th} external boundary $(j = 1, \ldots, k)$. The sub-matrices $[S_u] = [S_{uu}]$, $[S_{ee}]$, $[S_{ed}] = [S_{de}]$ and $[S_d] = [S_{dd}]$ are symmetric and positive definite.

In order to extract the friction coefficients a cosine distributed tangential traction is imposed on each fastener hole face while the rigid disks and the loaded boundaries are held fixed. The spring reactions are extracted and two coefficient matrices are constructed: $R_{uf}(i,j)$ and $R_{df}(i,j)$, where R stands for the reactions, subscripts u and d denote the upper and lower plates, respectively, i represents the index of the loaded hole and j represents the index of the hole for which the reaction is extracted. In each case the external forces are extracted and stored in a matrix $R_{ef}(i,j)$ where i represents the index of the loaded hole and j the index of the boundary for which the load is extracted.

Consider a hypothetical case where only frictional forces are acting on the plates while the fasteners (i.e. the rigid discs) and the external boundaries are fixed. In this case the following relations can be written:

$$\{F\} = [R_{uf}]\{\Lambda_u\} \tag{15}$$

$$\begin{Bmatrix} \vec{P} \\ \vec{F} \end{Bmatrix} = \begin{bmatrix} [R_{ef}] \\ [R_{df}] \end{bmatrix} \{\Lambda_d\} \tag{16}$$

where $\Lambda_{u|i}$ and $\Lambda_{d|i}$ are the friction amplitudes of the upper and the lower plates, respectively, at the i^{th} hole.

4.4.2 Modifying the equations to include friction

Consider the superposition presented in Figure 14 (the superposition procedure is valid as the plate itself is linear). The lower plate solution is obtained by superimposing case (a) where the frictional load is set to zero with case (b) where the fastener displacements ($\vec{D}$) and the external displacements ($\vec{D}_e$) are set to zero and frictional load is applied. Based on the above arguments relations (13) and (14) are modified to include friction.

First, the external load is considered:

$$\{P\} = \{P_a\} + \{P_b\} \tag{17}$$

substituting relations (14) and (16) into (17),

$$\{P\} = [R_{ef}]\{\Lambda_d\} + [S_{ee}]\{D_e\} + [S_{ed}]\{D\}. \tag{18}$$

Second, the fastener load is considered:

$$\{F\} = \{F_a\} + \{F_b\} \tag{19}$$

substituting (14) and (16) into (19),

$$\{F\} = [R_{df}]\{\Lambda_d\} + [S_{de}]\{D_e\} + [S_d]\{D\} \tag{20}$$

Also recall (12), which in our case can be written for the i^{th} fastener as:

$$\Lambda_{d|i} = \frac{8\mu_i}{t_i d_i \pi} F_i \qquad (no\ summation). \tag{21}$$

In this equation, d_i and t_i are the hole diameter and plate thickness, respectively, of the i^{th} fastener. At this point it is convenient to introduce the following definitions:

$$\{\hat{P}\} \stackrel{\text{def}}{=} [S_{de}][S_{ee}]^{-1}\{P\} \tag{22a}$$

$$[\hat{S}_d] \stackrel{\text{def}}{=} [S_d] - [S_{de}][S_{ee}]^{-1}[S_{ed}] \tag{22b}$$

$$[\hat{R}_{df}] \stackrel{\text{def}}{=} [R_{df}] - [S_{de}][S_{ee}]^{-1}[R_{ef}] \tag{22c}$$

$$Q_d(i,j) \stackrel{\text{def}}{=} \frac{\pi t_i d_i}{8\mu_i} I(i,j) - \hat{R}_{df}(i,j) \tag{22d}$$

$$[\tilde{S}_d] \stackrel{\text{def}}{=} \Big[[I] + [\hat{R}_{df}][Q_d]^{-1}\Big][\hat{S}_d] \tag{22e}$$

and

$$\{\tilde{P}\} \stackrel{\text{def}}{=} \Big[[I] + [\hat{R}_{df}][Q_d]^{-1}\Big]\{\hat{P}\} \tag{22f}$$

where $I(i,j)$ is the unit matrix (equals unity for $i = j$ and zero, otherwise). With these definitions, after eliminating the unknowns $\{\Lambda\}$ and $\{D_e\}$, relations (18), (20) and (21) can be written as:

$$\{F\} = [\tilde{S}_d]\{D\} + \{\tilde{P}\} \tag{23}$$

which may be considered as the "friction modification" of relation (14). The same process is repeated for the upper plate which yields a similar modification to the "non-friction" relations:

$$\{F\} = [\tilde{S}_u]\{U\}. \tag{24}$$

4.4.3 Forming the nonlinear equations

It is again convenient to introduce the following definitions:

a. The relative displacement, $\{\delta\}$:

$$\{\delta\} \stackrel{\text{def}}{=} \{U - D\} \tag{25}$$

b. The combined stiffness matrix, $[S_{ud}]$:

$$[S_{ud}] \stackrel{\text{def}}{=} \Big[[\tilde{S}_u]^{-1} - [\tilde{S}_d]^{-1}\Big]^{-1} \tag{26}$$

c. The fastener stiffness matrix, $[b(\bar{\delta})]$ which is a diagonal matrix:

$$b(i,j) \stackrel{\text{def}}{=} \begin{cases} F_i/\delta_i & \text{for } i = j \\ 0 & \text{for } i \neq j \end{cases} \tag{27}$$

where the transferred force F_i is a nonlinear function of a single variable, δ_i. Combining (23) and (24) with the above definitions (25), (26) and (27) yields the final set of nonlinear equations:

$$\Big[[b(\bar{\delta})] - [S_{ud}]\Big]\{\delta\} - [S_{ud}][\tilde{S}_d]^{-1}\{\tilde{P}\} = \{0\} \tag{28}$$

which can be solved for $\{\delta\}$ and finally, using (27), the load distribution $\{F\}$ is obtained.

The classical Newton-Raphson procedure for solving the system of nonlinear algebraic equations (28) fails to converge in case the solution of one of the fasteners, δ_i, is in the neighborhood of its initial clearance, $\delta_{0|i}$ (see Figure 11a). Hence, the *Hybrid-method* that was developed by M. Powel, [5] is employed. This method showed better convergence rate for most cases that were considered.

5.0 MODELING ASSUMPTIONS - DISCUSSION

In the suggested model, the interaction between the plate and fastener is modeled by distributed normal springs. It is relatively easy to verify that this approach is valid for an axisymmetric case. For example, it is obvious that the elastic solution for a plane stress disk installed in a circular plate with no clearance and the solution of a similar case where the inner disk is replaced by distributed normal springs fixed in the normal direction are equivalent, provided that the spring stiffness, $k = k_{nn}$, is chosen to be:

$$k = \frac{2}{1 - \nu_D} \frac{E_D}{d} \tag{29}$$

where E_D and ν_D are the inner disk Young's modulus and Poisson's ratio respectively, and d is the disk diameter.

A further investigation of the behavior of the distributed springs, is undertaken. An infinite plate with a circular hole into which an elastic circular disk has been inserted, is loaded by a unidirectional tension (see Figure 15). Both the disk and the plate are assumed to be in a state of plane stress. The radii of the disc and of the hole are the same before deformation. It is further assumed that the plate is in full contact with the disc along the common boundary and that friction is neglected. The solution to be derived shows however that the traction, T_r, is positive on part of the boundary. This departure from physical reality is cancelled when an axisymmetric compression of sufficient magnitude is superimposed on this solution. Based on the exact solution [6], the nondimensional contact stress, $\bar{\sigma}_{rr}$, over the hole perimeter is:

$$\bar{\sigma}_{rr}(\theta) = A_a + B_a \cos 2\theta \tag{30}$$

where $\bar{\sigma}_{rr} = \dfrac{4\sigma_{rr}}{\sigma_0}$ and σ_0 is the remote stress. A_a and B_a are material and geometrical parameters which are presented graphically in Figure 16 for different values of $\beta = \dfrac{E_D}{E_P}$ where $\nu_P = \nu_D = 0.3$. One can easily verify that for two extreme cases: a) $E_D \to 0$ and b) $E_D \to \infty$ this solution (30) is exact.

The inserted disk is then replaced with normal distributed springs as presented in Figure 15b. A solution for that case was not found in the literature, hence it is solved using the Airy stress function method. In that case the nondimensional radial stress $\bar{\sigma}_{rr}$ at $r = d/2$ can be expressed as:

$$\bar{\sigma}_{rr}(\theta) = A_b + B_b \cos 2\theta \tag{31}$$

where A_b and B_b are material and geometrical parameters and also are functions of k, the stiffness of the distributed spring. If k is chosen to match the axisymmetric

case as given by relation (29), the distributed springs solution (31) and the exact solution for the two-dimensional disk (30) agree. Figure 16 presents a comparison between A_a and A_b and between B_a and B_b for different values of β. It is concluded that the distributed spring model is a good idealization for a two dimensional disk in the cases considered.

In the case above the pin is not directly loaded since no closed form analytical solution is available. Test data, hence, is used to verify the model for this class of problems. Nisida et al. [7] obtained the stress distribution in a plate due to a loaded pin of the same material as the plate. A finite plate model of diallyphthalate (DAP) of dimensions 200 x 200 x 5.34 mm, Young's modulus E=244 kg/mm² and Poisson's ratio of $\nu = 0.41$, has at its center a circular hole of 20 mm in diameter, into which a close-fitting annular disk is inserted (see Figure 17). It is supported at one side and loaded in the other direction by a pin which just fits the annular disk. An interferometric method is used to measure the stress field in the plate. In [7] the frictional force is not measured, but it is noted that since σ_1 and σ_2 do not coincide with σ_{rr} and $\sigma_{\theta\theta}$, respectively, it is clear that some frictional force is acting on the boundary between the hole and the pin. The results obtained for σ_{rr} and $\sigma_{\theta\theta}$ are presented in Figure 18.

A p-version finite element model is constructed to describe the geometry presented in Figure 17 using *PROBE*. The plane stress formulation is used with the material parameters as specified above. Due to symmetry only a half model is constructed. The model is loaded by a uni-directional tension in the upper edge which is opposed by the distributed springs over half of the hole face (to represent the contact zone between the pin and the plate). Fourteen elements were used with polynomial levels $p = 1$ to 8 (for $p = 8$, 920 degrees of freedom are used). The mesh is presented in Figure 19. Three iterations were necessary to locate the

ends of the contact boundary. That is, the spring region in tension was iteratively removed. First, the stress distribution (σ_{rr} and $\sigma_{\theta\theta}$) on the contact boundaries was obtained for the case where friction is neglected, that is the transverse shear along the contact boundary is set to zero. Examining the results in Figure 18 shows good agreement between test results and the current results for σ_{rr} but there is a slight difference in $\sigma_{\theta\theta}$. It is assumed that the difference is due to the fact that frictional stresses are not included in the numerical analysis. Imposing a cosine distributed shear stress, equivalent to $\mu \cong 0.15$ ($|\sigma_{rr}| = \mu|\sigma_{r\theta}|$), improves the stress distribution in relation to the test results (see Figure 18).

6. COMPUTATIONAL IMPLEMENTATION OF THE MODEL

6.1 Description of the System

The solution process is divided into three major steps. In the first step the linear portion of the model, the plate, is analyzed using *PROBE*. The data needed in the subsequent steps is extracted in the second step. Finally, in the third step, the nonlinear equations (28) are assembled and solved for predefined external load increments. The input for this step includes the data extracted in the previous step (e.g. $[S_u]$ and $[S_d]$) combined with the following additional information:

a) External forces data.

b) Friction data.

c) Fastener stiffness data.

d) General parameters (such as convergence criteria).

For the initial approximate solution, the user may choose one of the following algorithms:

a) An infinite stiffness for all fasteners.

b) A linear solution where the user can specify a constant value for the stiffness of all fasteners.

c) An equal force distribution between fasteners.

The proper selection of the initial solution depends on the problem, and may reduce the number of iterations necessary to obtain the desired solution. A bad decision may cause the process to diverge. The three steps and their relationships are illustrated in a flowchart shown in Figure 20.

A flowchart of the nonlinear program (step c) is presented in Figure 21: First, the basic nonlinear equations are assembled. Second the equations are modified to include friction, if necessary. Third, the initial external forces and the initial solution are calculated. Finally, the hybrid method is employed to solve the nonlinear equations. A subroutine documented in [8], which is based on the hybrid method, was incorporated into the program. If this subroutine fails to converge, then the program attempts to solve again with a different initial solution. After a few failures the process stops. In this case, the user should check for errors in the input data or reduce the external load increments.

6.2 Example - The Three-Fastener Joint

In order to study the effect of different parameters on the load distribution, and to demonstrate the different features of the model, a simple joint is considered. The problem includes three plates fastened together by six fasteners (see Figure 22). The upper and lower plates have three levels of thickness, where relevant dimensions are illustrated in Figure 22. A uni-directional load is applied to the edge of the center plate. All three plates are assumed to be in a state of plane stress and have the same Young's modulus of 10^7 psi and Poisson's ratio of 0.3. Due to symmetry only a quarter of the model is considered.

First, a p-version finite element model of the upper and center plates are constructed (see Figure 23). By solving for three different load vectors, the matrix $[S_u]$ is extracted. Then, by solving the other three different load vectors for the lower plate, the matrix $[S_d]$ is constructed. Finally, by solving for a unit external displacement, the scalar S_{ee} and the vector $\{S_{ed}\}$ are determined. The above matrices will be used in all of the following examples.

6.2.1 Linear Fasteners with no Initial Clearance

With the above data, a parametric study is conducted to determine the effect of fastener stiffness on the load distribution between three fasteners. First, a linear fastener model with no initial clearance is considered. The load transferred by each fastener is presented in Figure 24 for stiffness values between 1lb/in to 10^7lb/in, where the external load is set to 1,000 lbs. The graph may be divided into three major regions, where the stiffness values are:

a) Less than 100 lb/in - the fasteners transfer a small load which is practically negligible. In this region the fastener stiffness is small compared to the stiffness of the upper plate. So, this problem is equivalent to a case where the upper plate is not attached to the center plate at all.

b) Between 100 lb/in to 10^5 lb/in - in this transition zone, the fastener stiffness approaches values that are on the same order of magnitude as the upper plate stiffness ($\sim 10^4$ lb/in).

c) Above 10^5 lb/in - in this third region the fastener stiffnesses are much larger than that of the plates and may be considered as rigid links. Hence, the load transferred by each fastener is constant.

Note that the relative load distribution between the fasteners, does not change much with fastener stiffness. For example, the first fastener transfers between

59% (where the stiffness of the fasteners was 1lb/in) to 63% (where the stiffness was 10^7 lb/in) of the total transferred load.

6.2.2 Initial Clearance

The problem presented in Section 6.2.1 is reconsidered. This time each of the three fasteners has a constant stiffness of 1000 lb/in. It is further assumed, that only the first fastener was installed with an initial clearance, and the other two fasteners were installed with close fit. Five levels of initial clearance are considered at the first fastener:

a) Very large (infinite) - which practically implies that the first fastener is missing and the structure is linear through the considered range.

b) 0.18 inches - in this case the fastener comes into contact with the plate when the external load is approximately 2,000. lbs.

c) 0.12 inches - where the fastener comes into contact with the plate when the external load is approximately 1,300. lbs.

d) 0.06 inches - where contact occurs for external load of approximately 650. lbs.

e) No initial clearance - in that case the solution is linear through the entire range.

The external load is applied in three time steps. At each step the load is increased by 1,000. lbs to a total of 3,000. lbs. The load transferred by the first fastener is presented in Figure 25 for each of the five levels of clearance.

It can be noted that in each clearance level, the transferred load is zero until contact between the plate and the first fastener occurs, then the load becomes a linear function of time with a positive slope. So, the transferred load is a piecewise linear function. A similar behavior is observed for the other two fasteners. All solutions are bounded in between two linear solutions, which are represented in

Figure 25 by dashed lines. These lines represent the solutions of two extreme cases: a) the *infinite initial clearance* and b) the *no initial clearance.*

6.2.3 The Effect of Friction

In order to study the effect of friction, the linear case is reconsidered. All fasteners are assumed to be installed with no initial clearance and an external load of 1,000. lbs is applied in a single time step. Solutions for four levels of friction are determined:

a) $\mu = 0$, the basic case,

b) $\mu = 0.1$,

c) $\mu = 0.2$,

d) $\mu = 0.3$.

It is assumed that the coefficient of friction between the fastener and each plate is the same. The percent increase in fastener load relative to the basic case, where no friction is present ($\mu = 0$) is illustrated in Figure 26. It should be noted that friction helps in transferring the load from one plate to another, and the effect of friction increases with the transferred load. This effect is almost a linear function of μ (see Figure 26). The presence of frictional loads caused a change in the transferred load of between 5% (for $\mu = .1$) to 15% (for $\mu = .3$) and therefore in many cases friction cannot be ignored.

7. SUMMARY AND CONCLUSIONS

The problem of computing load distribution among fasteners in structural connections has been considered. It was noted that the current modeling practice, in which fasteners are typically handled by multipoint constrains, that is nodes of the finite element mesh are positioned at fastener locations and the

nodes are connected by springs, is conceptually wrong and consequently the computed forces in the fasteners are entirely discretization-dependent. An example, showing the effects of incorrect modelling practices, was presented.

With respect to the common modelling practice of using radial links attached to the bore to represent the fastener, it was demonstrated through a simple test case that with the h-version approach many degrees of freedom are required to obtain accurate stress distribution in the vicinity of fasteners. It was also demonstrated that the p-version of the finite element method converges much faster and is, therefore, much better suited for this purpose. For this reason the proposed approach to modelling fastened connections is based on the p-version of the finite element method.

The proposed model takes into account possible nonlinear response of the structure as well as friction forces. This new formulation was coded into a computer program which was used to solve the example of a three-fastener joint reported in this paper.

With respect to future developments, there are possibilities for improving the performance of extraction procedures for the stresses such that the stress concentration factor around the fastener hole will be calculated by a superconvergent scheme. Additional investigations concerned with application of the model to fastened orthotropic sheets and solution sensitivity with respect to change in fastener parameters, such as initial clearance, are currently underway and will be reported in a future paper.

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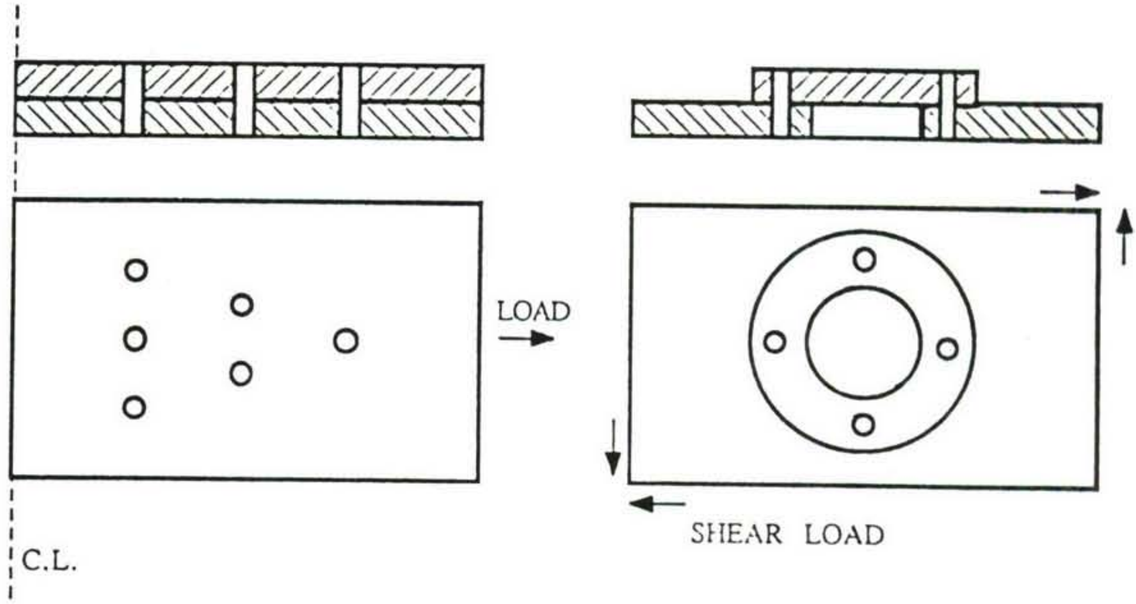


Figure 1 Typical cases.

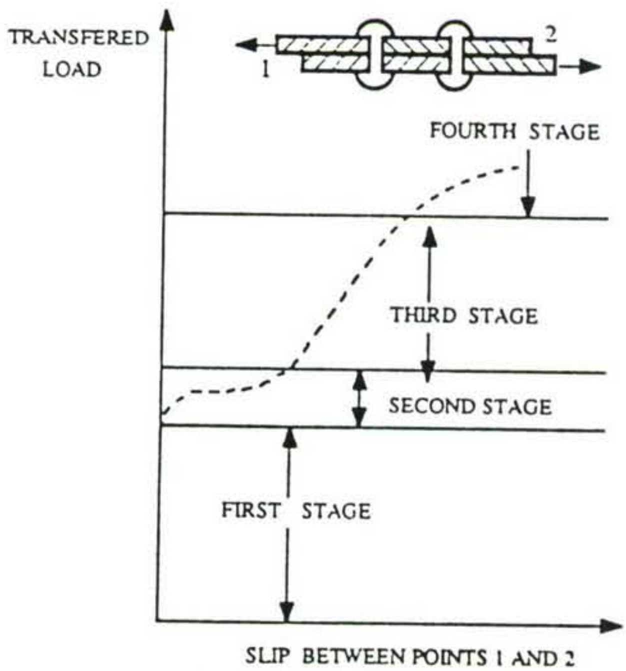
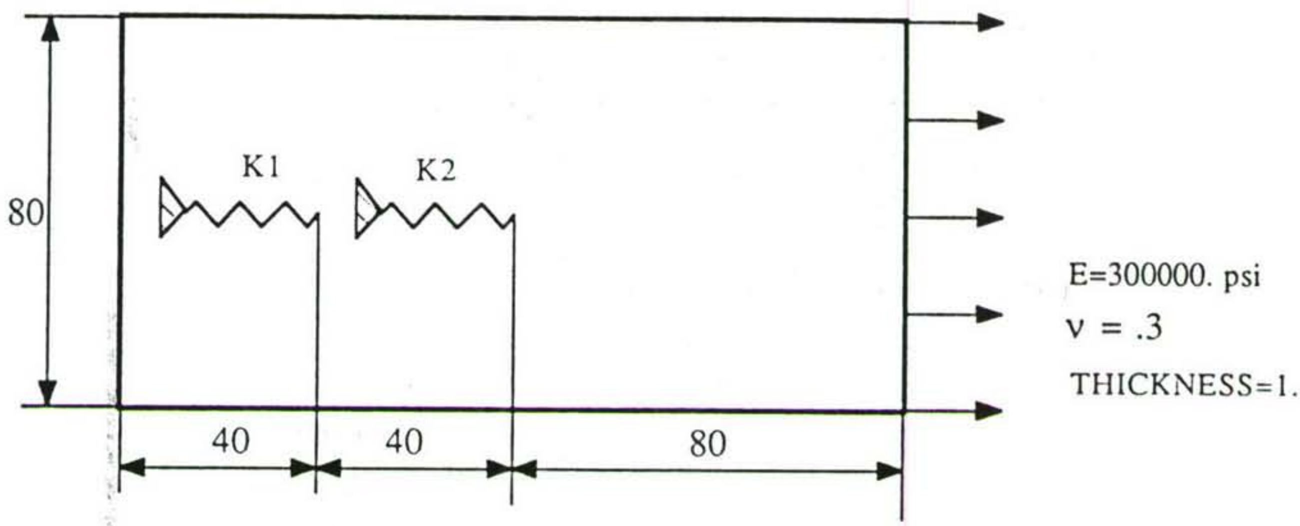


Figure 2 Action of riveted joints.



NOTE: THE DIMENSIONS ARE IN INCHES.

Figure 3 The line spring test case.

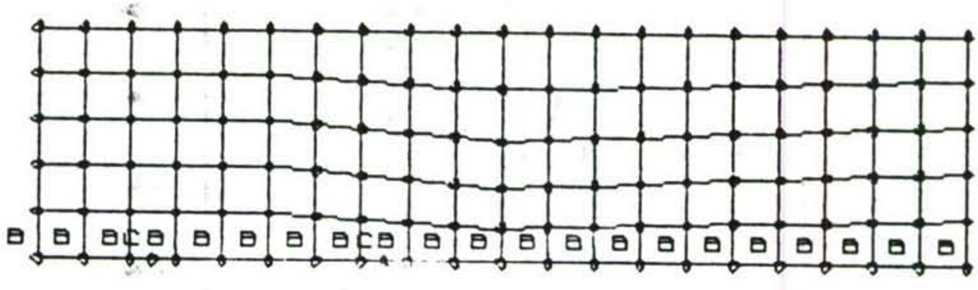


Figure 4 A Typical finite element mesh for the line spring test case ($Q = 0.5$).

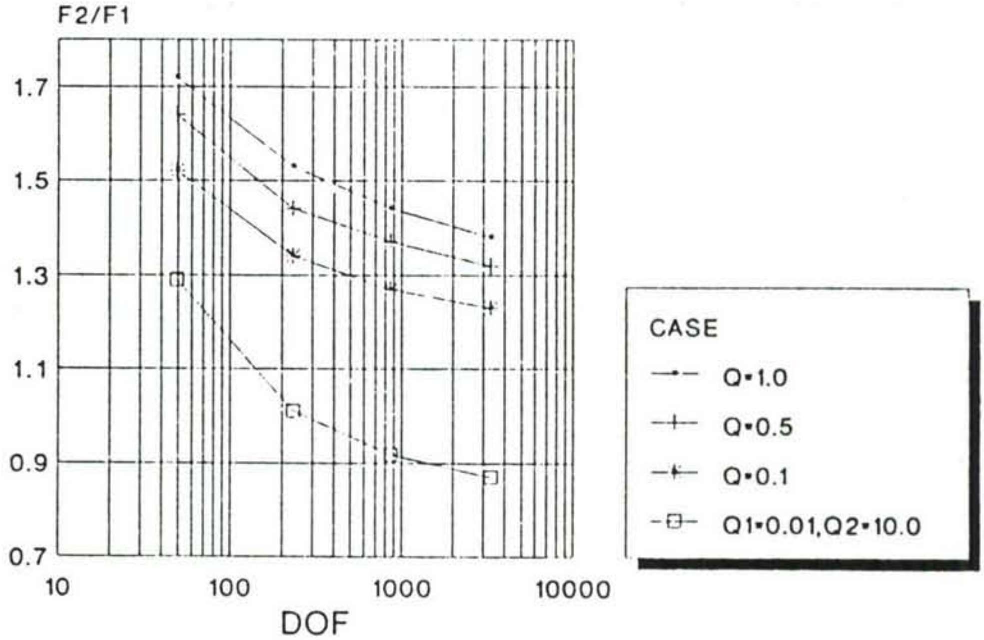


Figure 5 The load distribution between two springs-four noded elements.

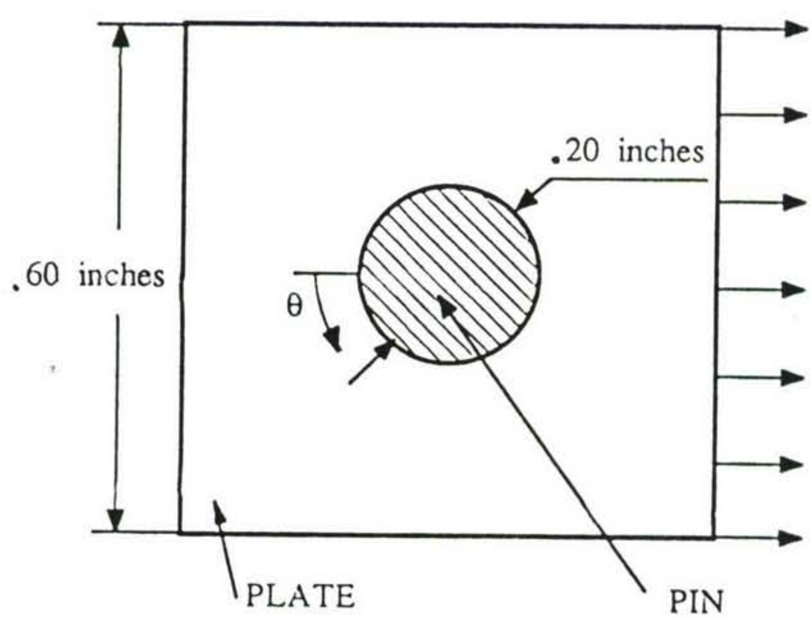


Figure 6 The distributed spring test case.

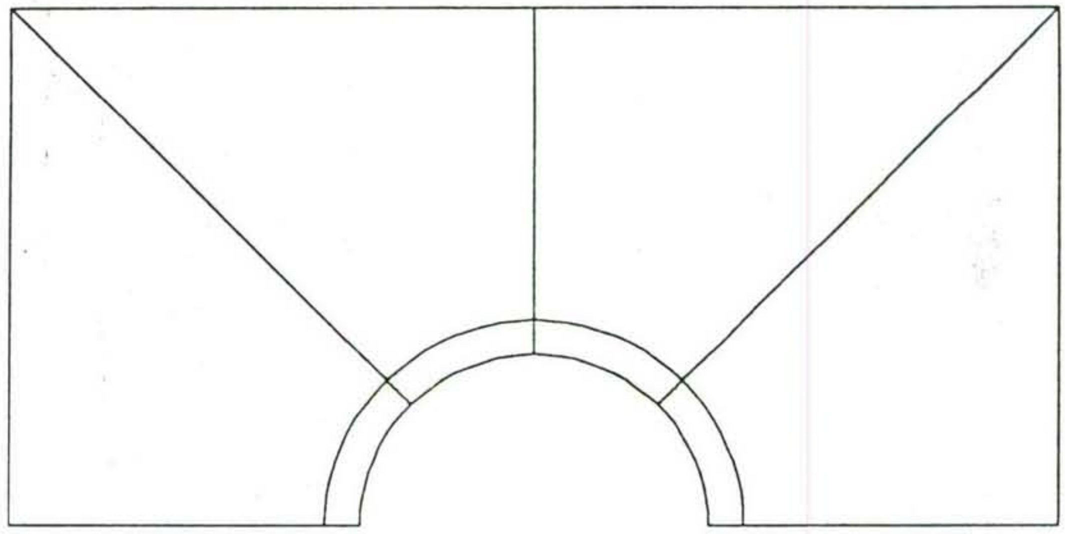


Figure 7 The p-version mesh used for the distributed spring test case.

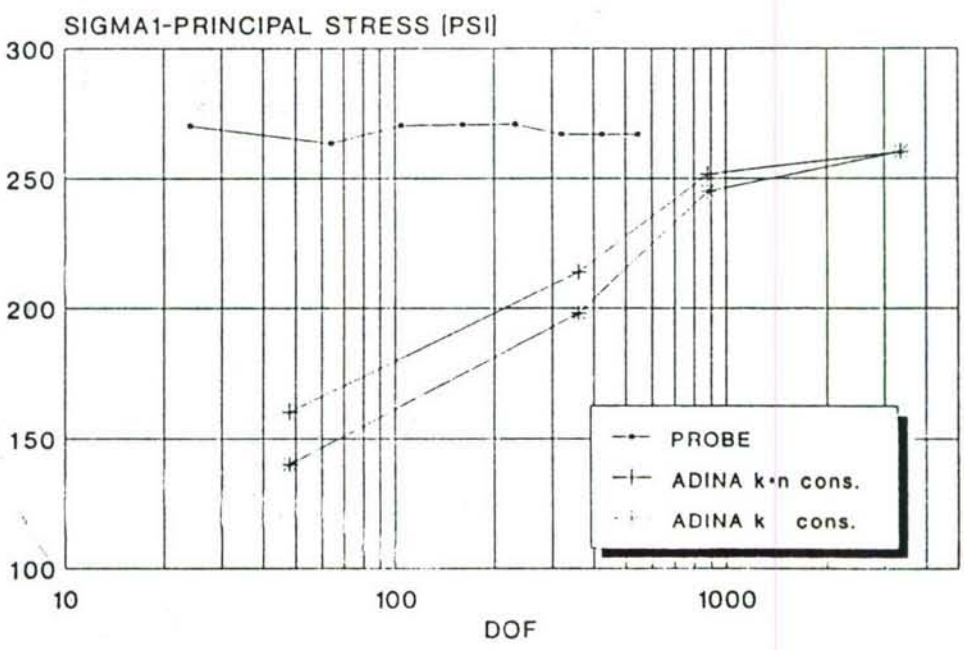


Figure 8 The maximum stress convergence - linear elements.

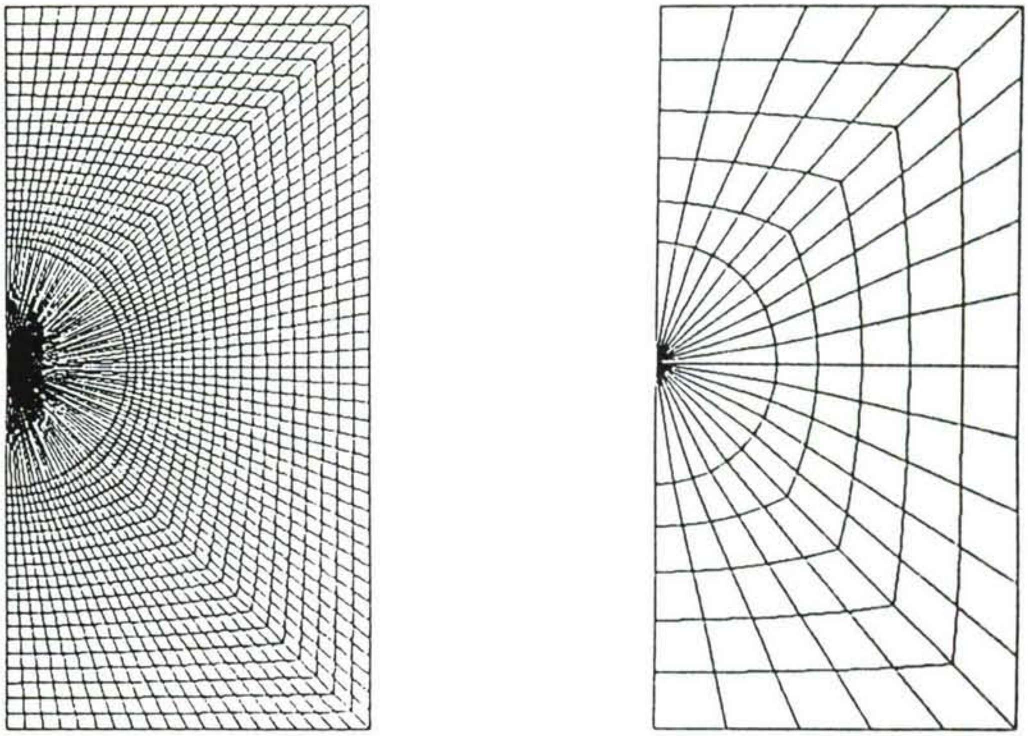


Figure 9 The "distributed links case" - Typical finite element meshes.

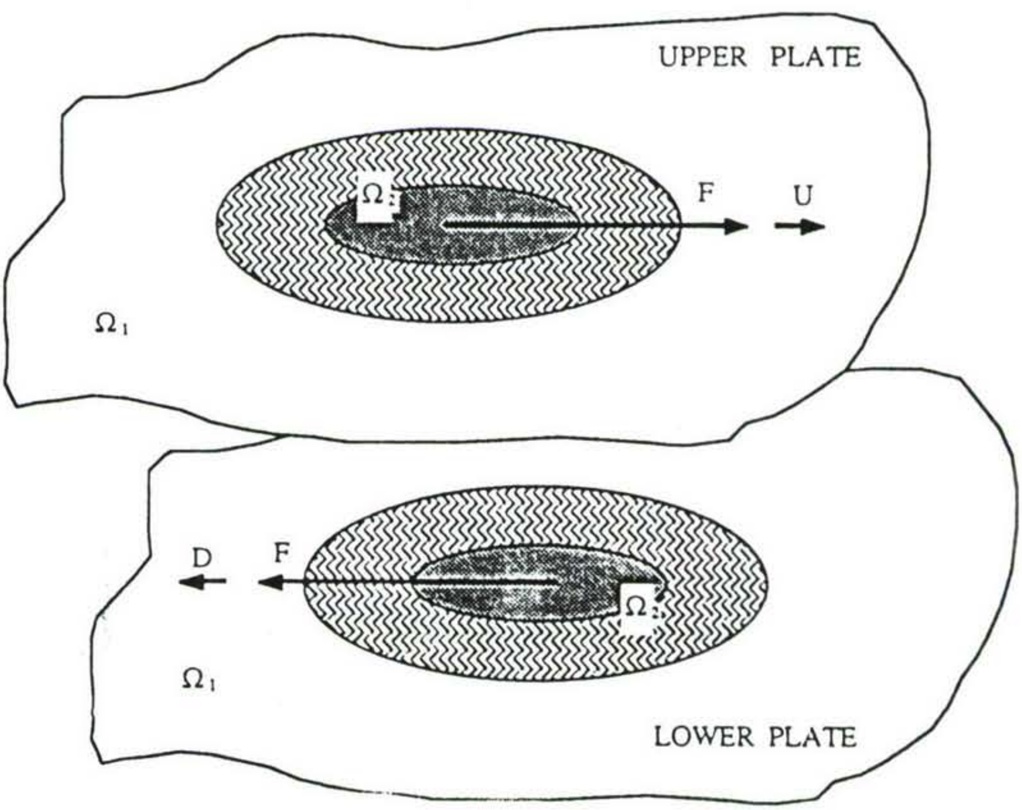


Figure 10 The schematic model.

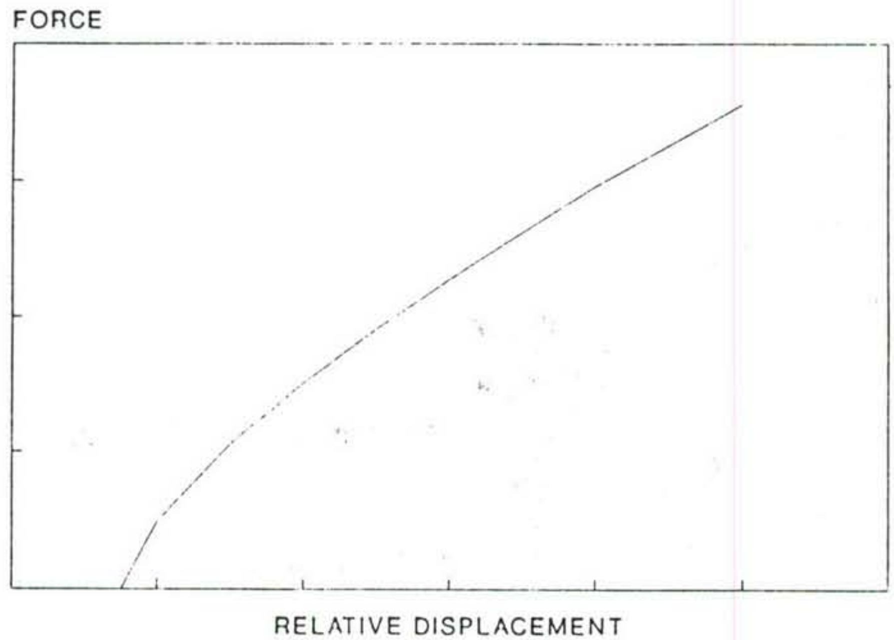


Figure 11a Typical fastener force-displacement relation - Three parameters.

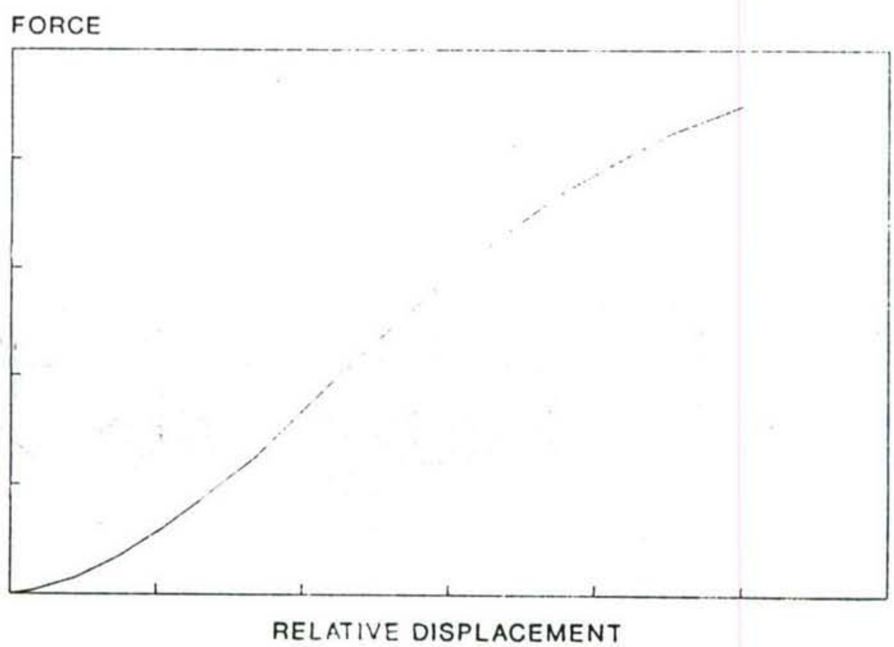


Figure 11b Typical fastener force-displacement relation - Six parameters.

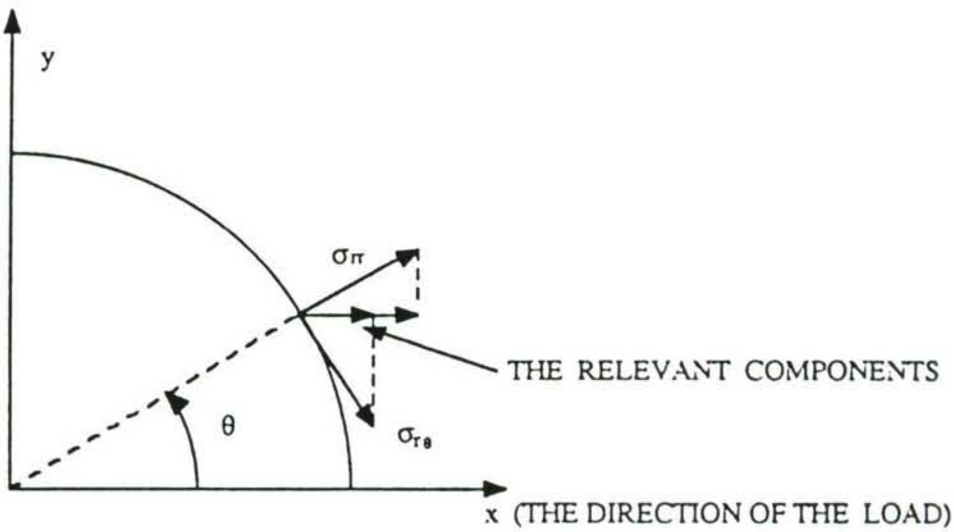


Figure 12 The relevant traction components.

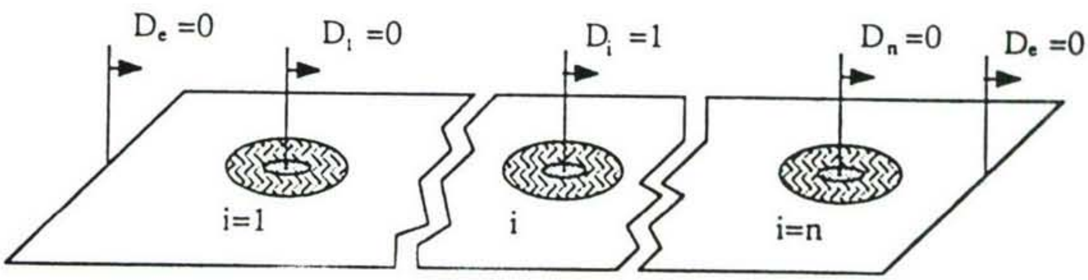


Figure 13 A representative case.

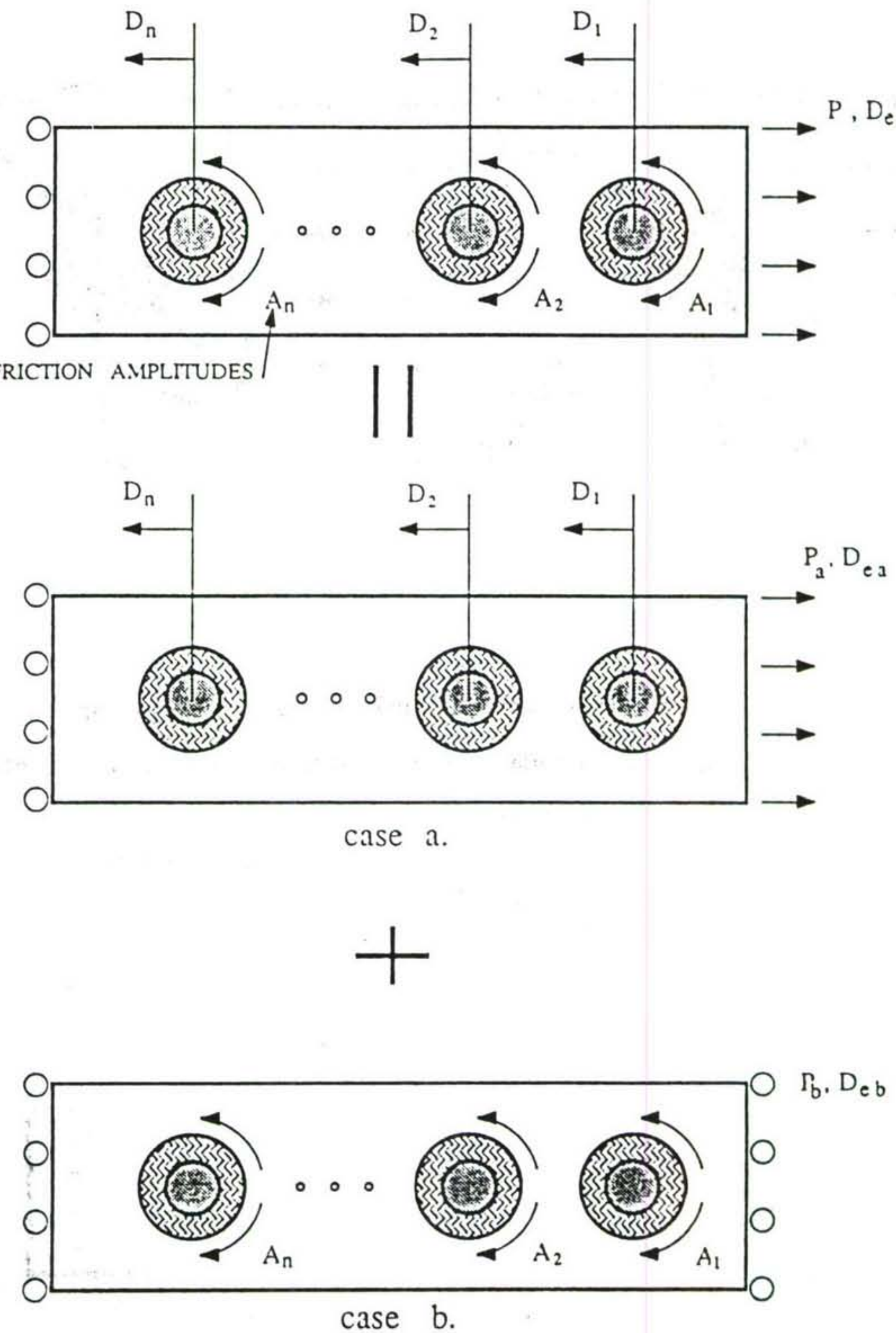
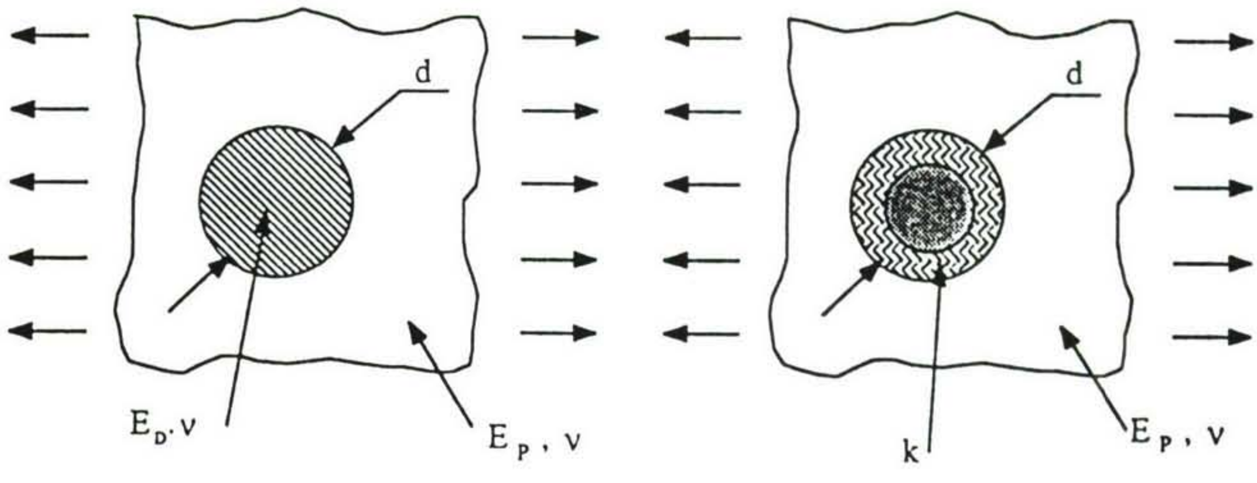


Figure 14 The superposition.



a. Two-dimensional disk. b. Distributed springs.

Figure 15 Circular disk inserted into a two-dimensional plate

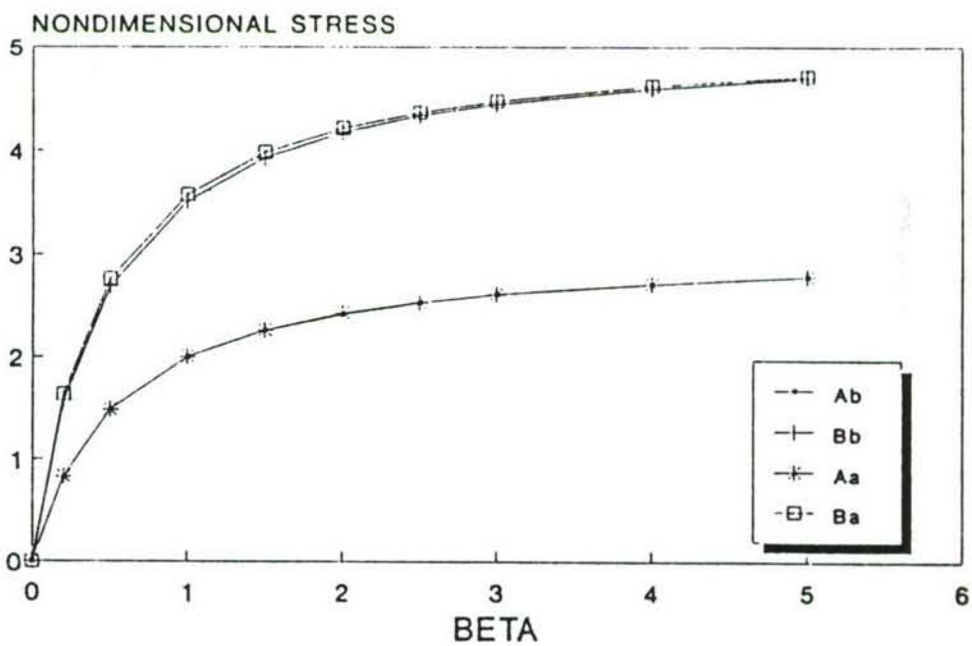


Figure 16 A circular plate inserted into an infinite plate -

The two-dimensional solution versus the distributed springs.

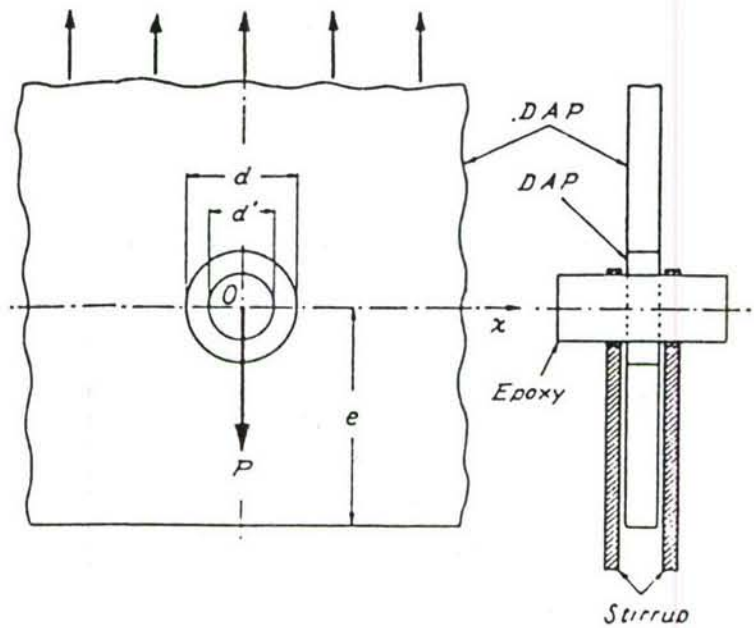


Figure 17 Illustration of model and loading apparatus [7].

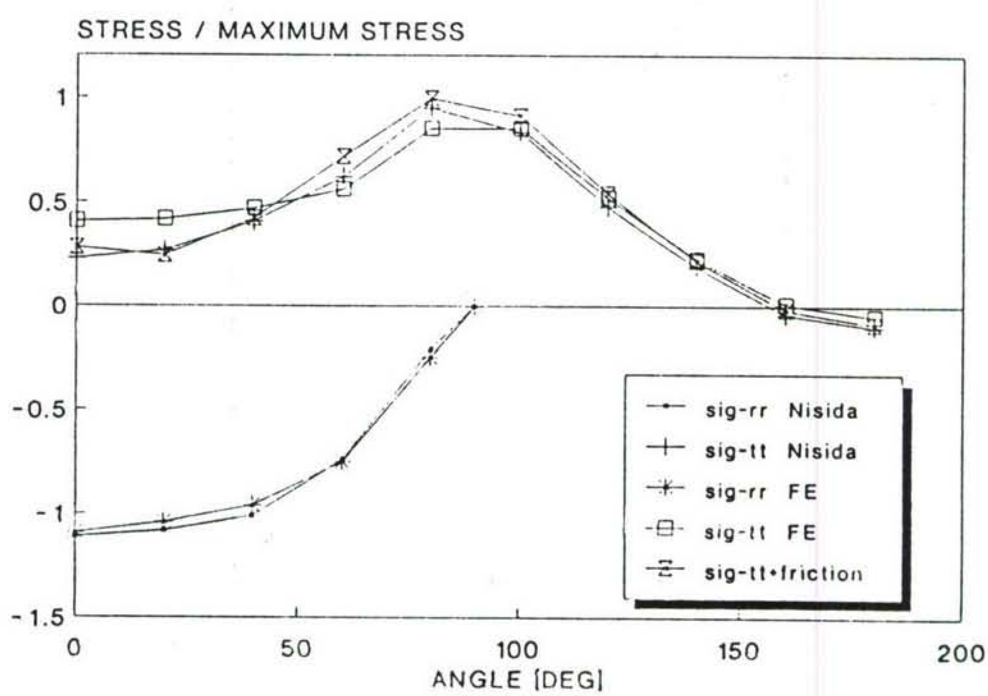


Figure 18 Photoelastic results versus the suggested model.

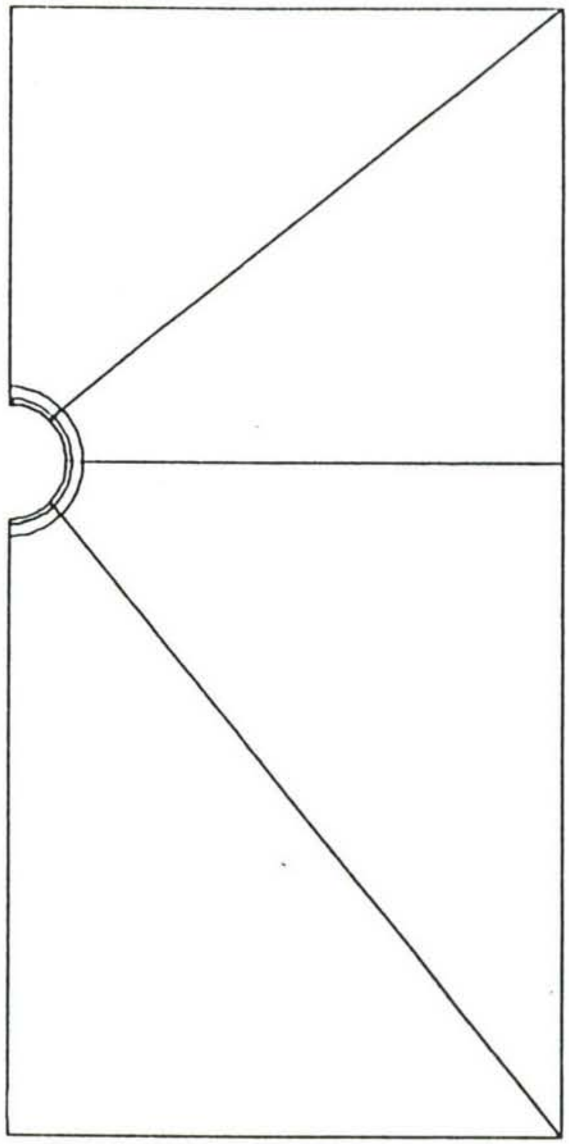


Figure 19 The p-version finite element mesh.

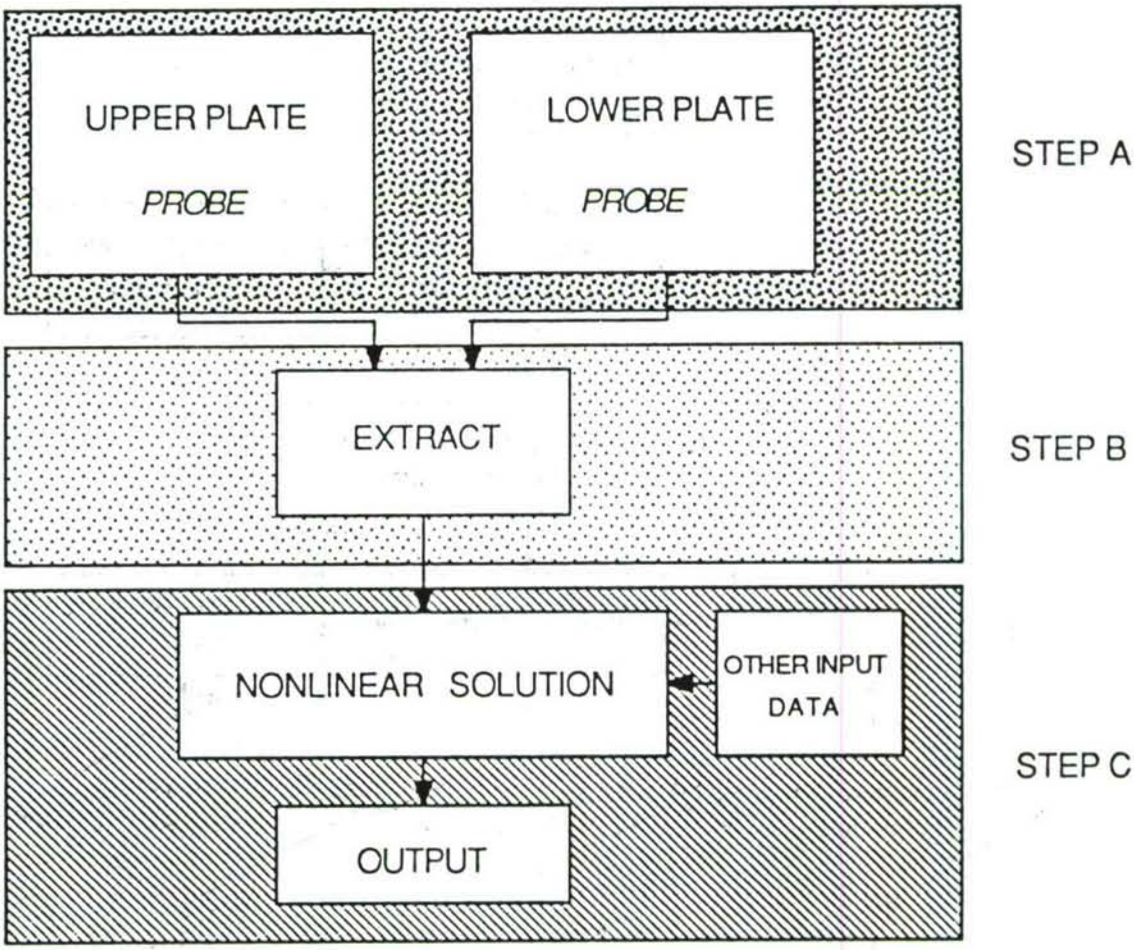


Figure 20 The three basic steps.

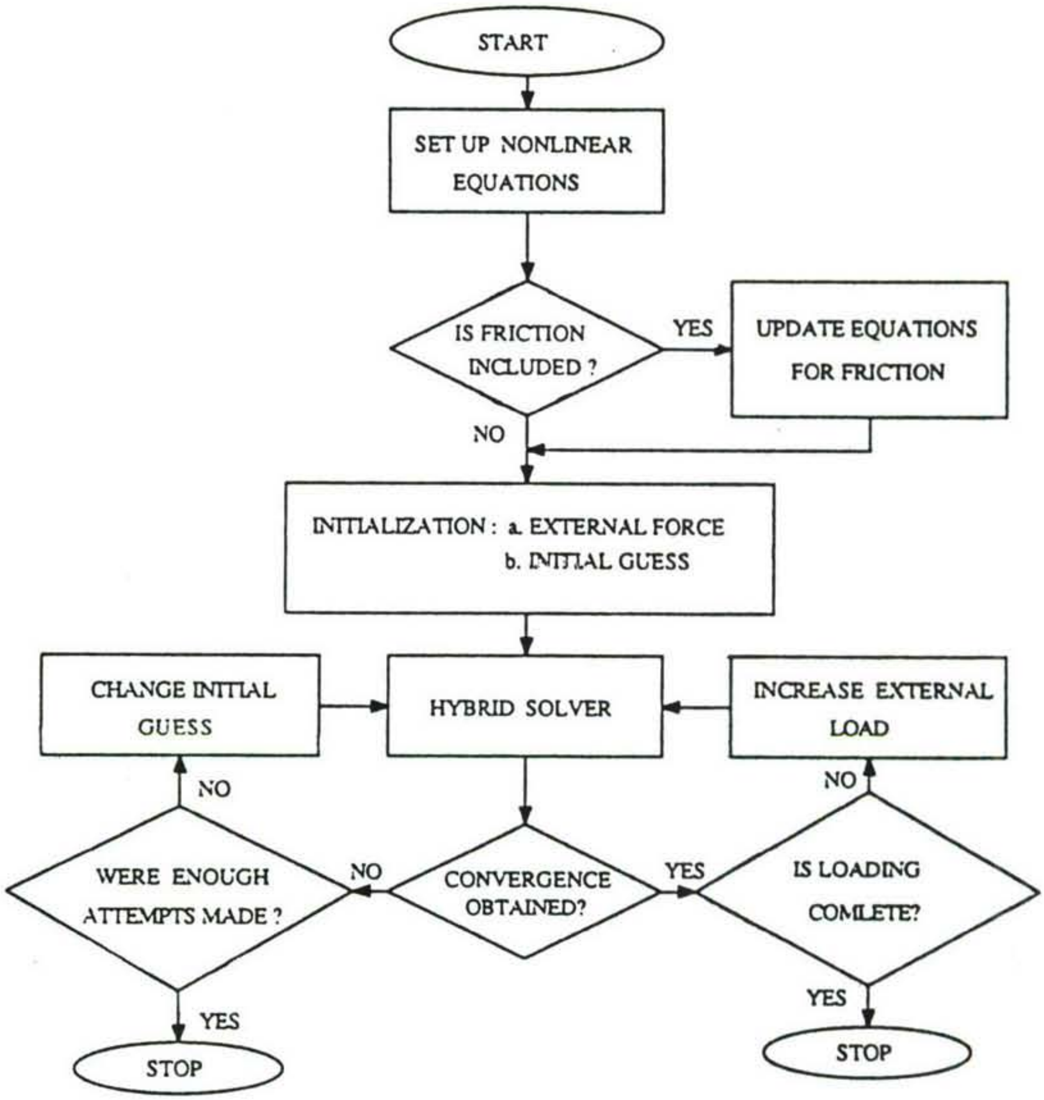
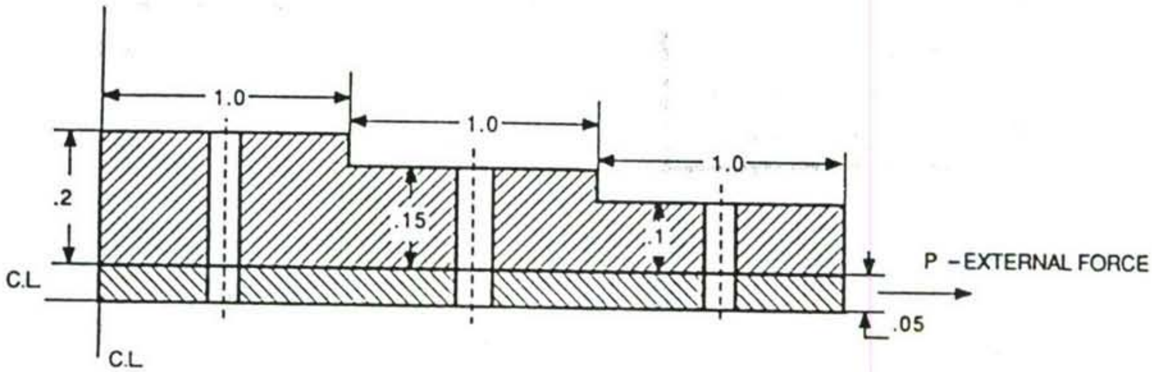


Figure 21 Overall view of the system.



note: not to scale.
all dimensions are in inches.
only a quarter of the model is presented

Figure 22 The test case - Three-fastener joint.

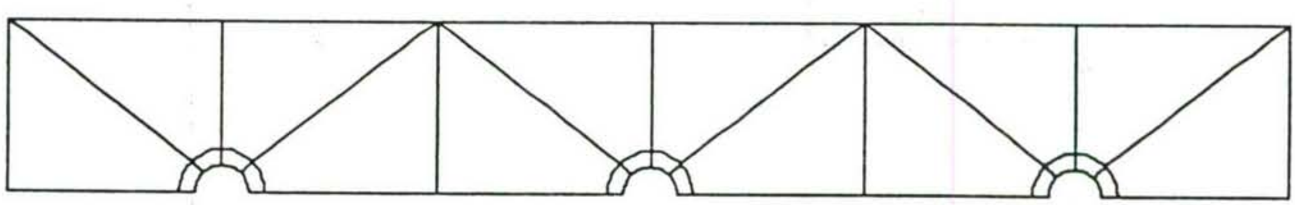


Figure 23 The finite element meah.

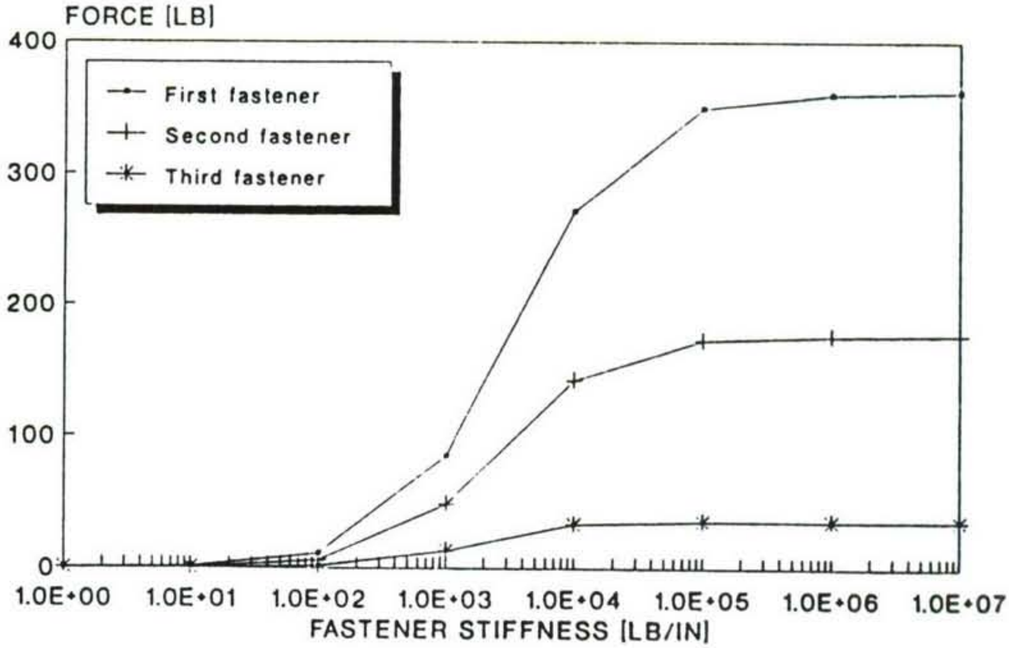


Figure 24 The load distribution between fasteners.

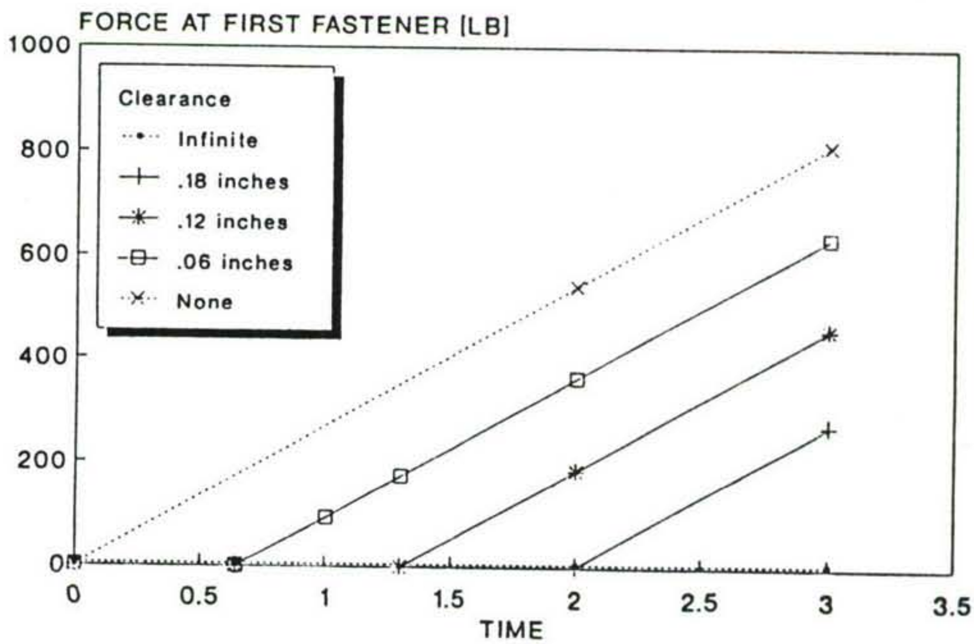


Figure 25 The effect of clearance at the first fastener.

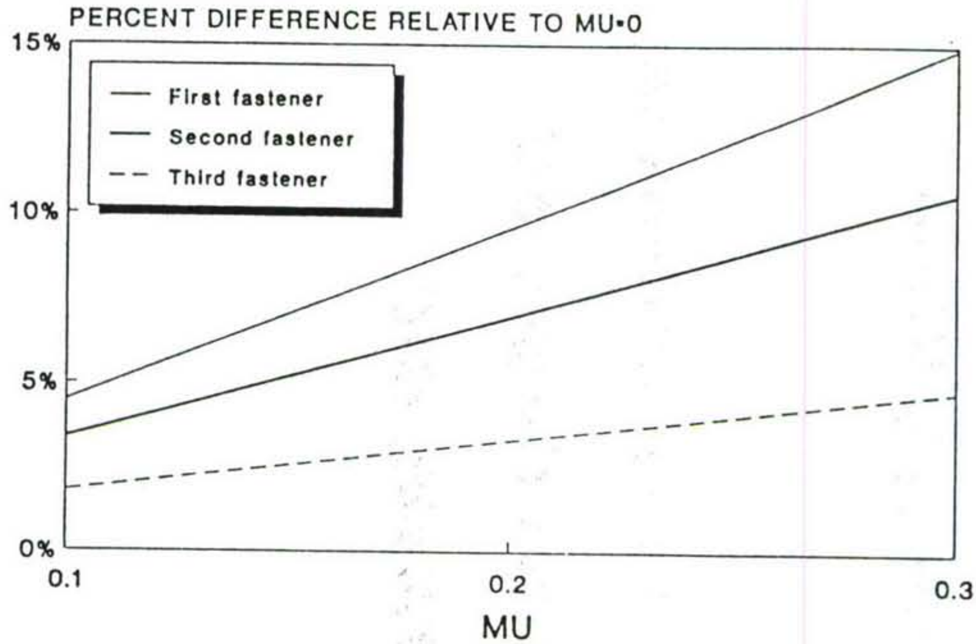


Figure 26 Friction effect on the transferred load.

Damage Tolerance Analysis of the Gulfstream GIV Wing

Faramarz (Faz) Keyvanfar, SIE[1]

Matthew Creager, SIE[1]

Jeff Kreide, GAC[2]

[1] *Structural Integrity Engineering*
[2] *Gulfstream Aerospace Corporation*

Two steps in damage tolerance analysis to comply with the code of Federal Regulations, Title 14, Part 25:

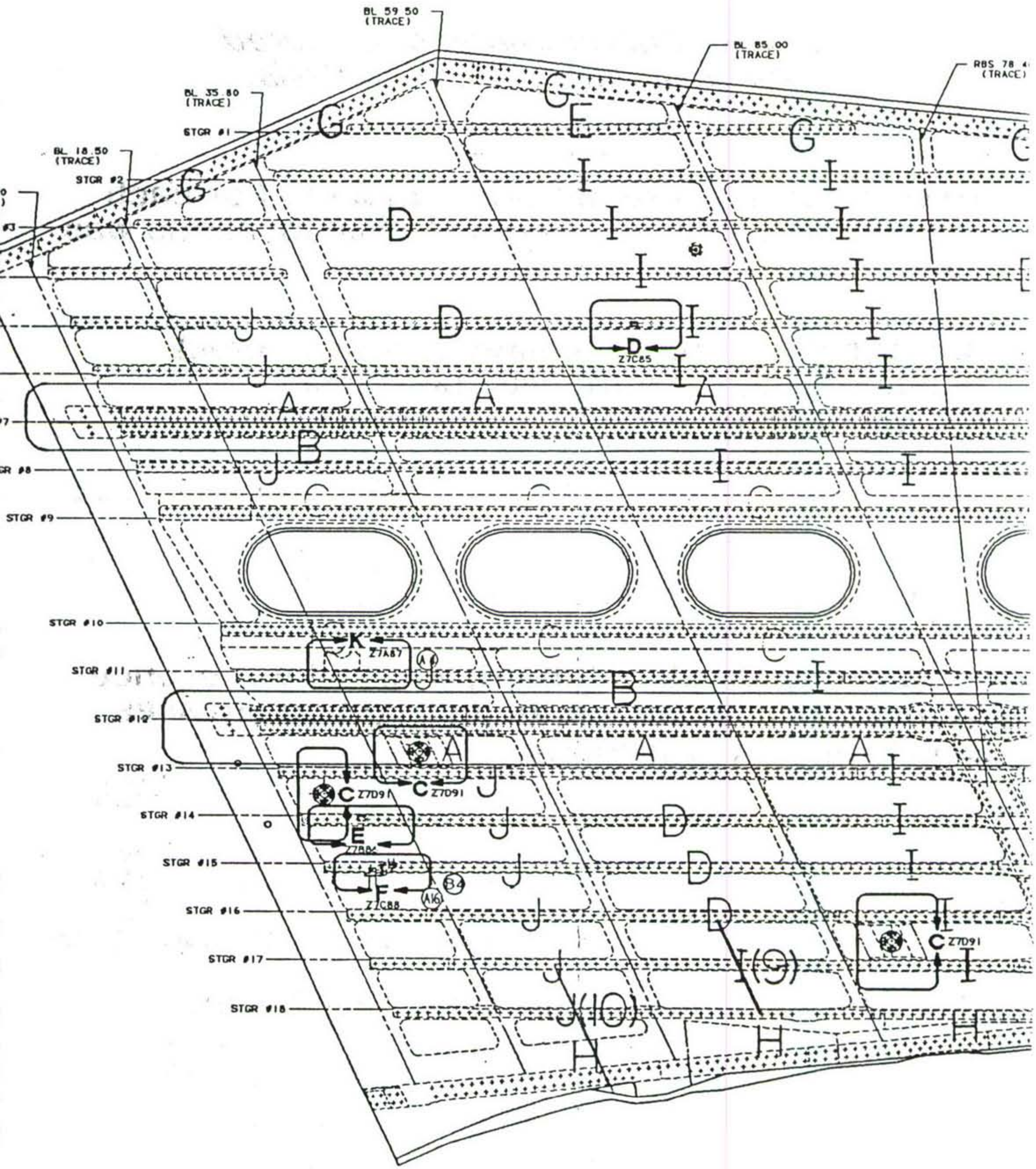
1. Residual Strength Analysis:

   to evaluate capability of the structure containing obvious damage to withstand limit load without catastrophic failure.

2. Crack Growth Analysis:

   to determine adequate inspection intervals to assure crack detection prior to failure.

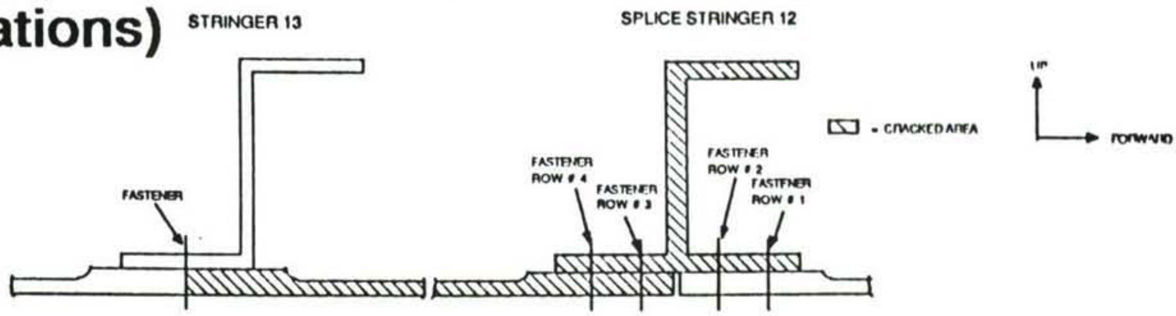
Residual Strength Analysis


BL 59.50
(TRACE)
BL 85.00
(TRACE)
RBS 78.4
(TRACE)
BL 35.80
(TRACE)
BL 18.50
(TRACE)
STGR #1
STGR #2
STGR #8
STGR #9
STGR #10
STGR #11
STGR #12
STGR #13
STGR #14
STGR #15
STGR #16
STGR #17
STGR #18
Z7C85
Z7A87
Z7D91
Z7C88
I(9)
J(10)

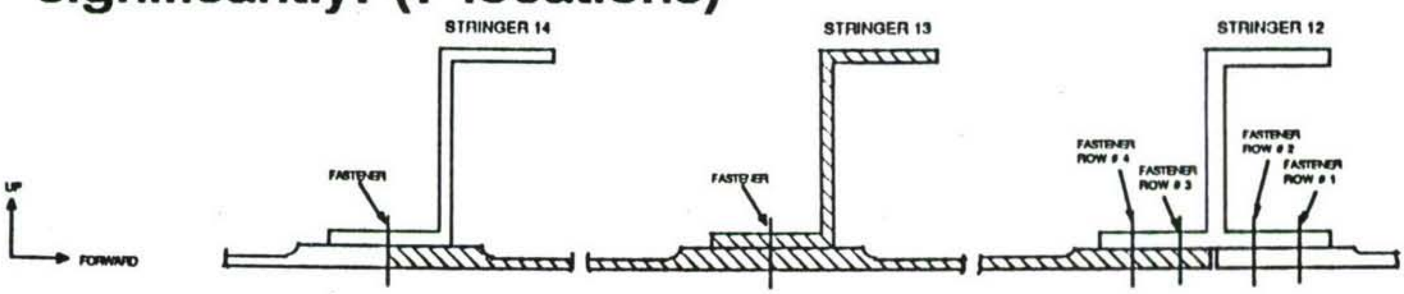
Crack configurations considered for residual strength analysis:

10 crack configurations were studied. For each configuration, the most critical of all locations was analyzed:

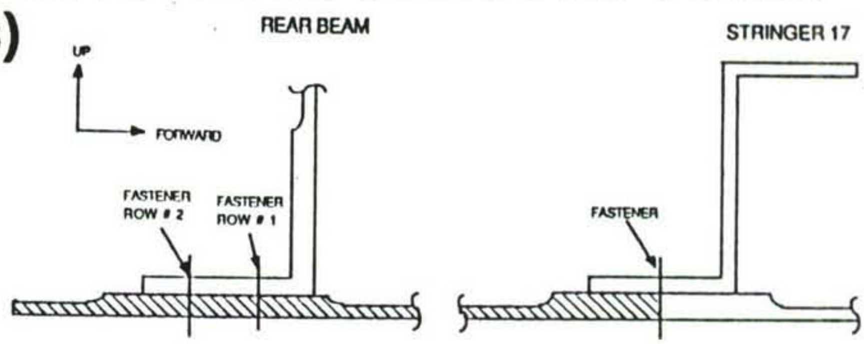
A. Edge crack from a spanwise splice; crack pointing away from the midchord line: (14 locations)



F. Two-Bay Crack outboard, where the presence of a crack may change the net section stresses significantly: (7 locations)

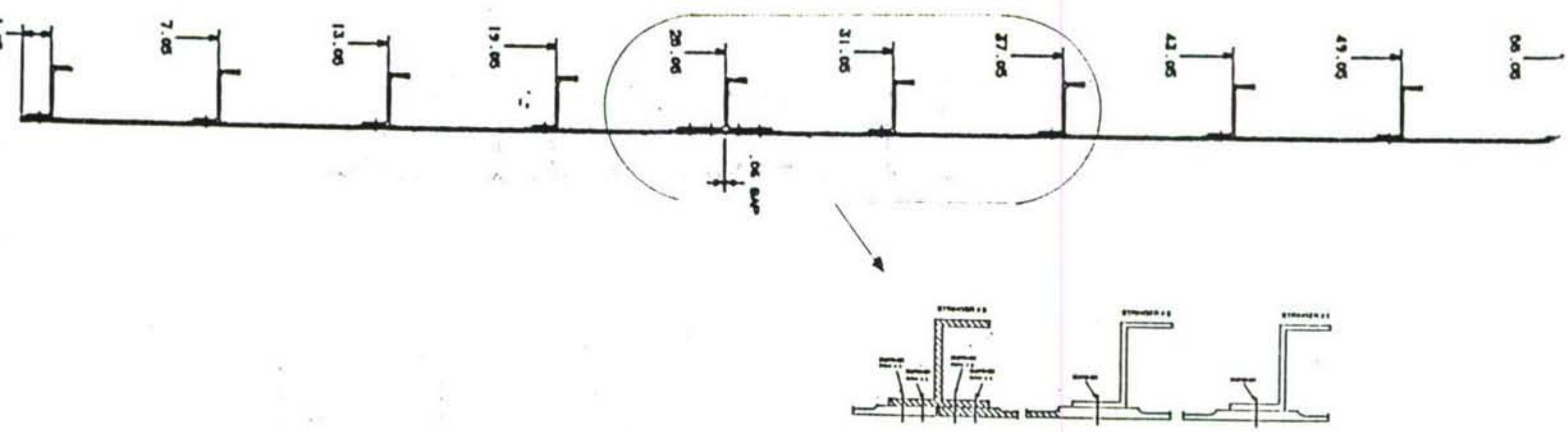


H. Edge crack in the skin at the rear beam: (13 locations)

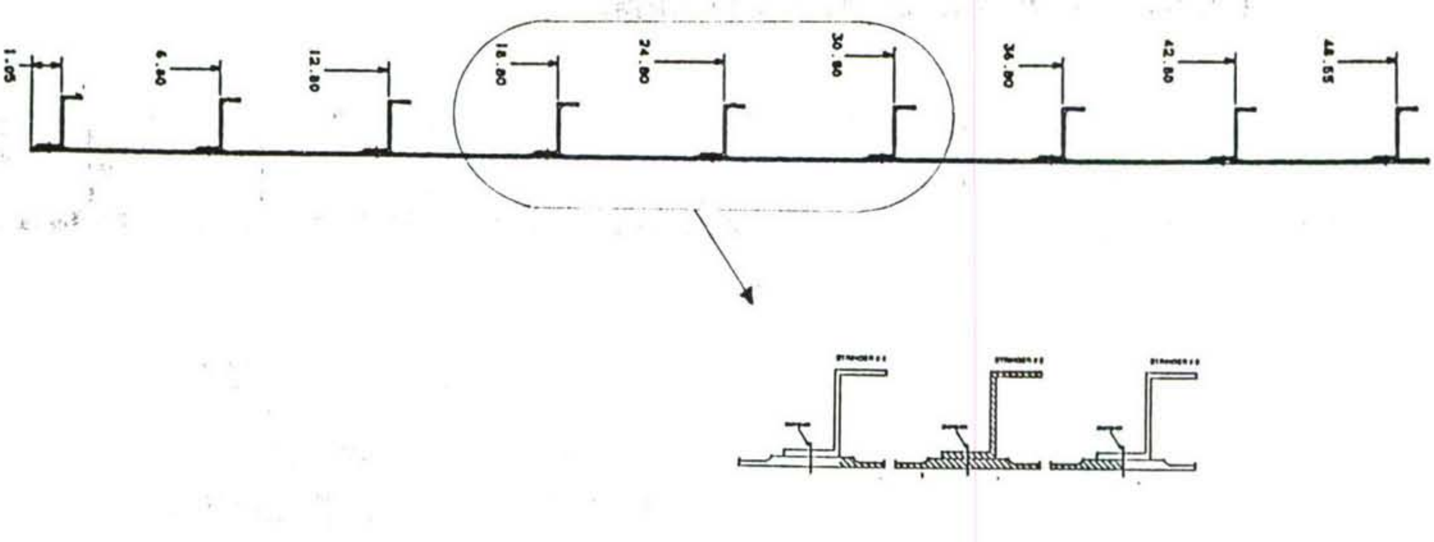


Four residual strength tests were run to validate the methodology and to establish Kc:

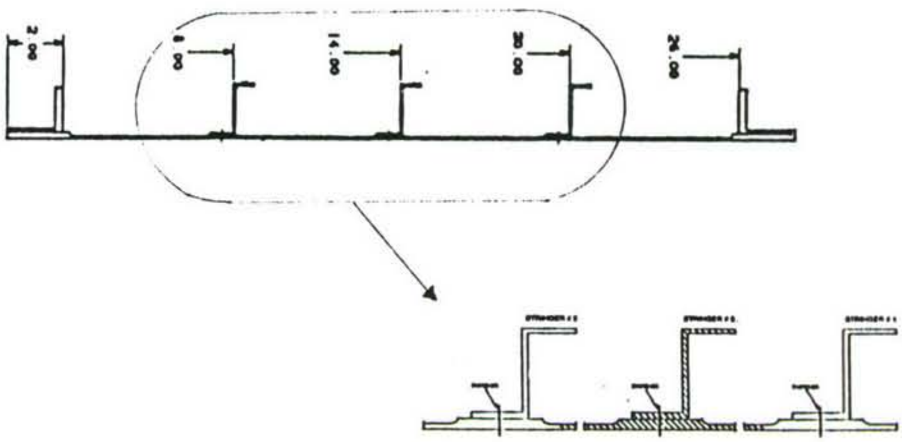
Test 450: a spanwise splice stringer failed plus a one bay skin crack:



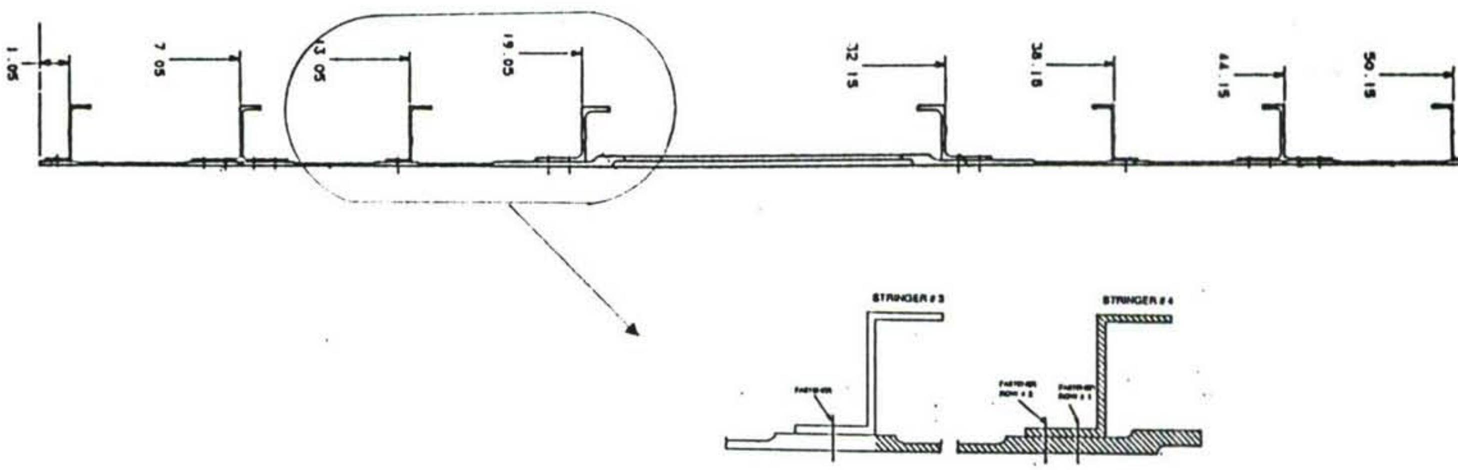
Test 455: a failed stringer plus a two bay skin crack whose tips are far from the skin edges:

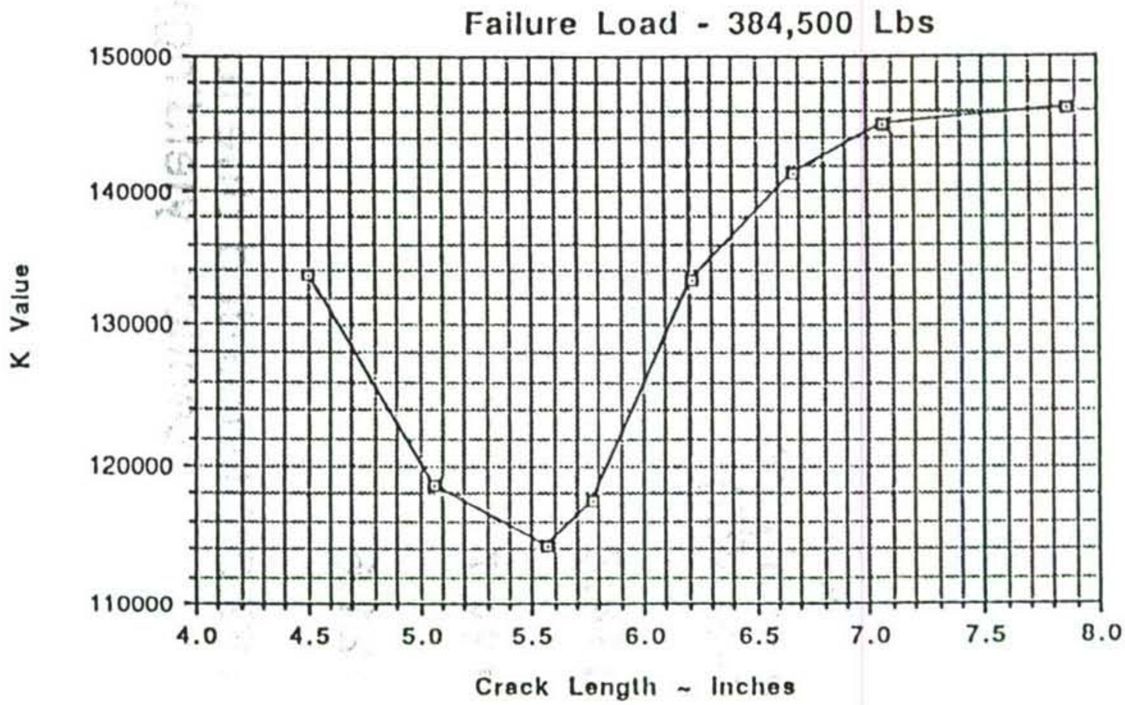


Test 457: a failed stringer with a two bay crack whose tip are near the skin edges:



Test 460: a failed coaming stringer plus a skin crack from an access hole to the adjacent stringer:





File	Crack Length (in)	Coordinate A (in)	Distance to Fastener	K	GENELt (in)
		25.56	0.00		
R450GEN8	4.50	26.63	-1.07	133678	0.110
R450GEN1	5.07	26.06	-0.50	118595	0.142
R450GEN7	5.57	25.56	0.00	114324	0.142
R450GEN2	5.77	25.36	0.20	117584	0.142
R450GEN3	6.22	24.91	0.65	133181	0.126
R450GEN4	6.67	24.46	1.10	141429	0.110
R450GEN5	7.07	24.06	1.50	145062	0.110
R450GEN6	7.87	23.26	2.30	146197	0.110

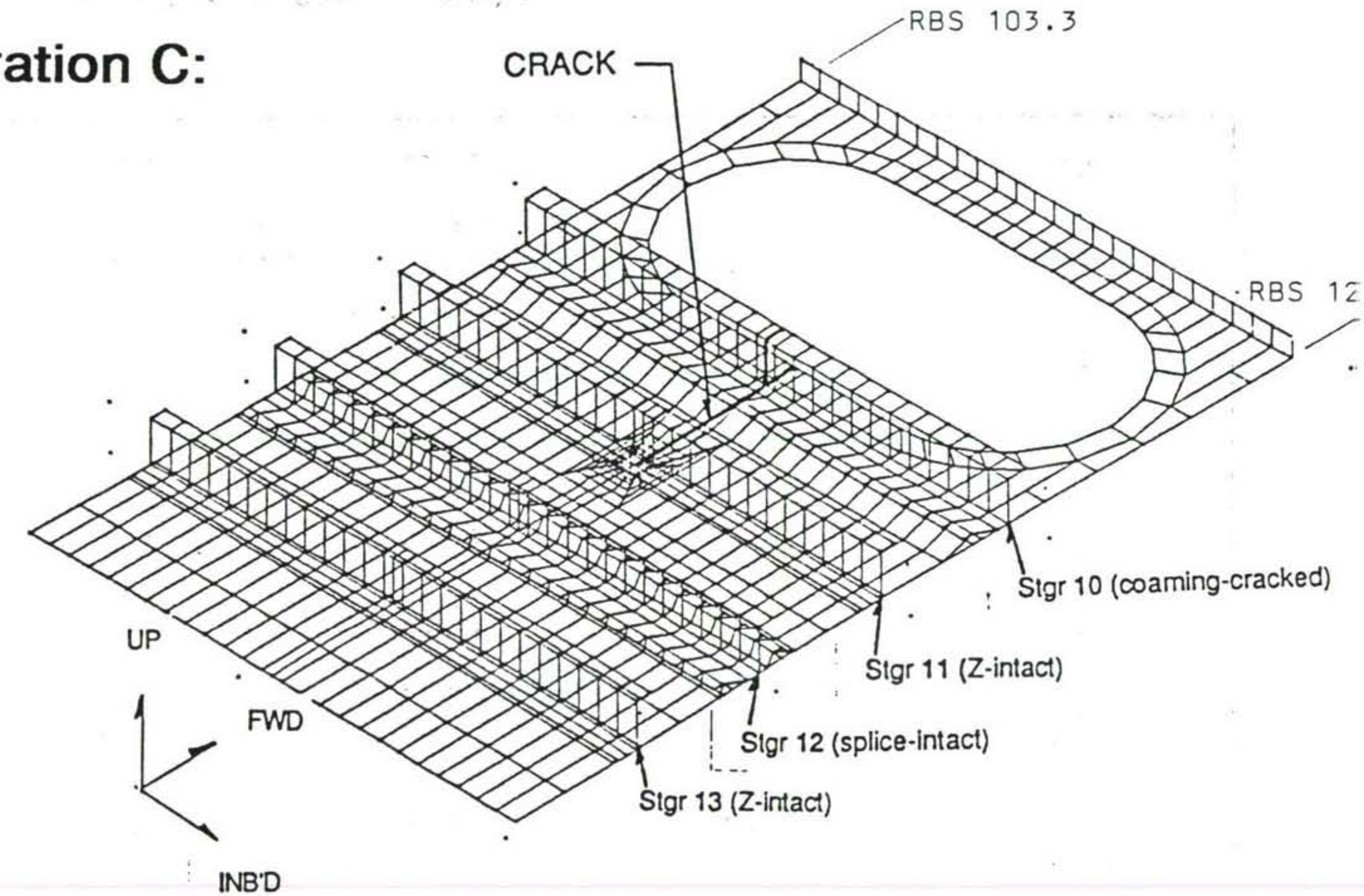
Kc was determined to be the average Kc from various models:

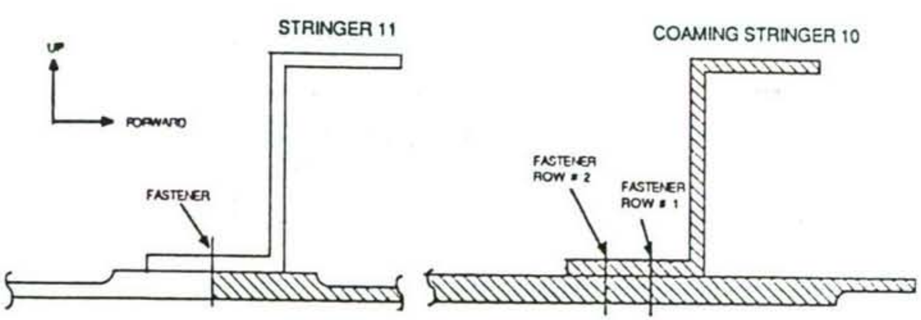
Test number	Failure load from test (lbs)	K from analysis (ksi $\sqrt{in}$)
450	384500	114.3
455	306500	124.7
457	192000	121.6
460	292100	157.2 (discarded due to section yielding)

Average Kc: 120 ksi $\sqrt{in}$

Residual Strength Analysis Example:

Crack configuration C:





Crack Length	K_I (psi $\sqrt{in}$)	K_{II} (psi $\sqrt{in}$)	Fastener Yield	Stringer Yield
7.94	93694	470	Yes	No

CRACK CONFIGURATION	CRACK LENGTH (inch)	K (psi.sqrt(in))	FASTENER YIELD	STRINGER YIELD
A	6.44	85,980	YES	NO
B	5.56	51,150	NO	NO
C	7.94	93,690	YES	NO
D	12.00 -TIP A 12.00 -TIP B	70,070 59,200	NO NO	NO NO
E	12.40 -TIP A 12.40 -TIP B	58,590 28,240	NO NO	NO NO
F	12.44	53,960	YES	NO
G	9.96	84,400	YES	YES
H	7.60	39,590	NO	NO
I	12.40 -TIP A 12.40 -TIP B	53,630 60,300	NO NO	NO NO
J	18.55	99,640	NO	NO

All K values are smaller than Kc (=120,000psi.sqrt(in)): all above locations are damage tolerant.

Crack growth Analysis

Group I: 17 locations:

- 9 on Rear Beam
- 8 on Front Beam

Most critical location at RBS 35.8 on Rear Beam.

Group II: 48 locations:

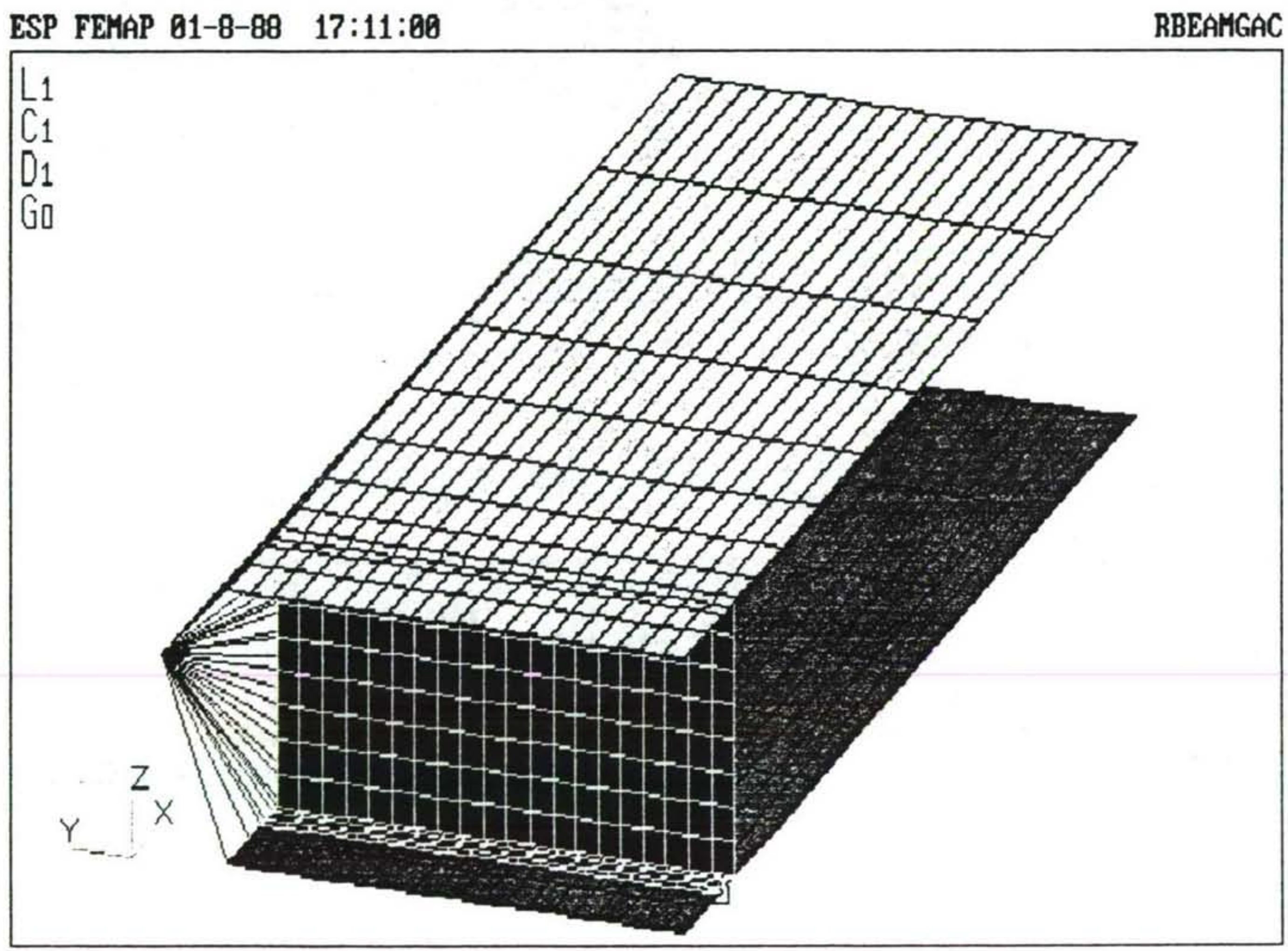
- 7 on Rib Posts
- 12 on Rib Caps
- 1 on Strut
- 6 Fuselage Attach Lugs
- 10 on Lug Backup Fittings
- 12 in Wing-to-Wing Splice Area

Reduced to a total of 10 locations by comparison

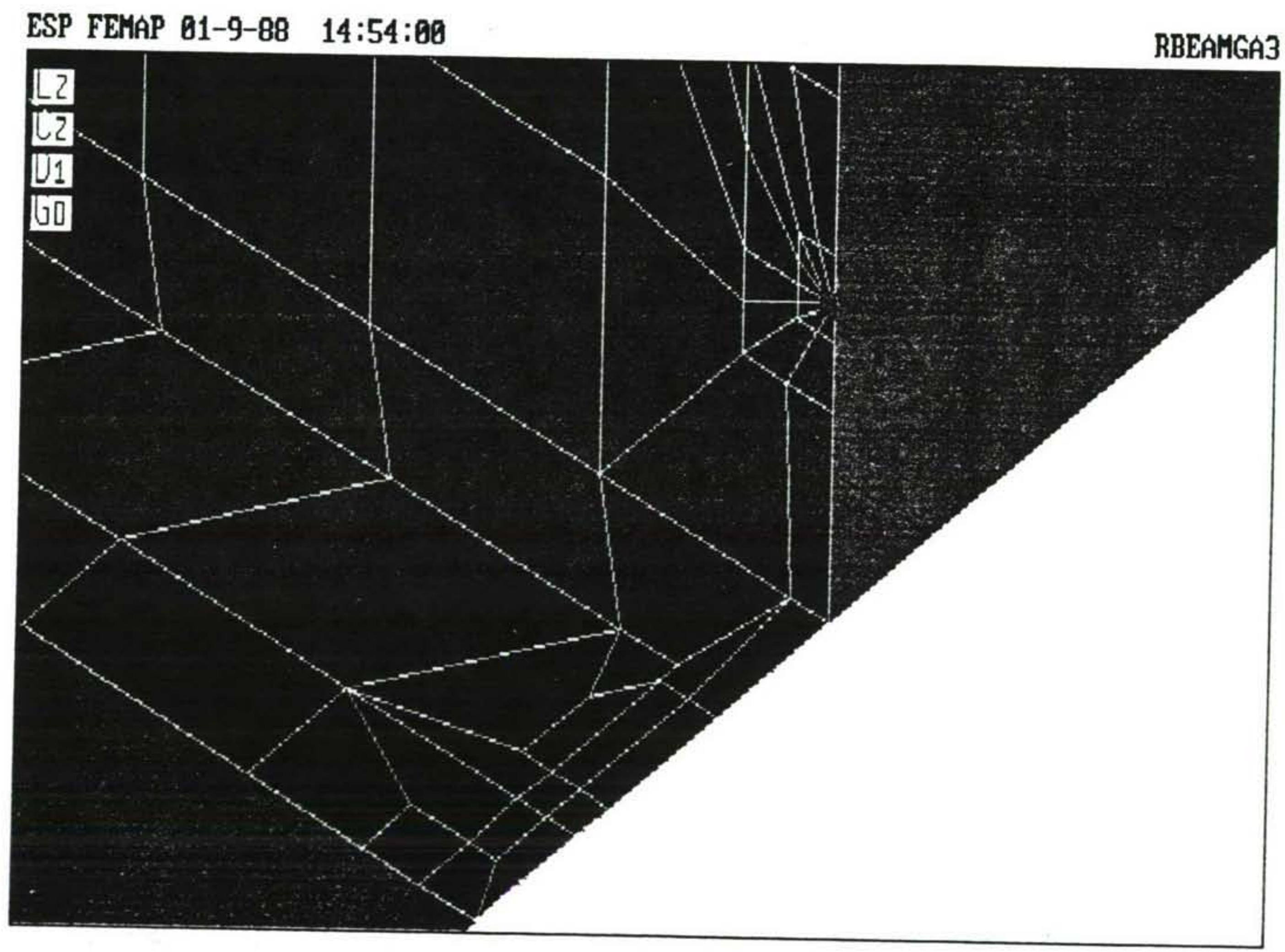
Group III: 8 locations:

- 4 Fuselage Attach Lugs (adjacent lug having failed)
- 4 Lug Backup Fittings (adjacent lug having failed)

Finite Element Model of Rear Beam and Wing Skins:



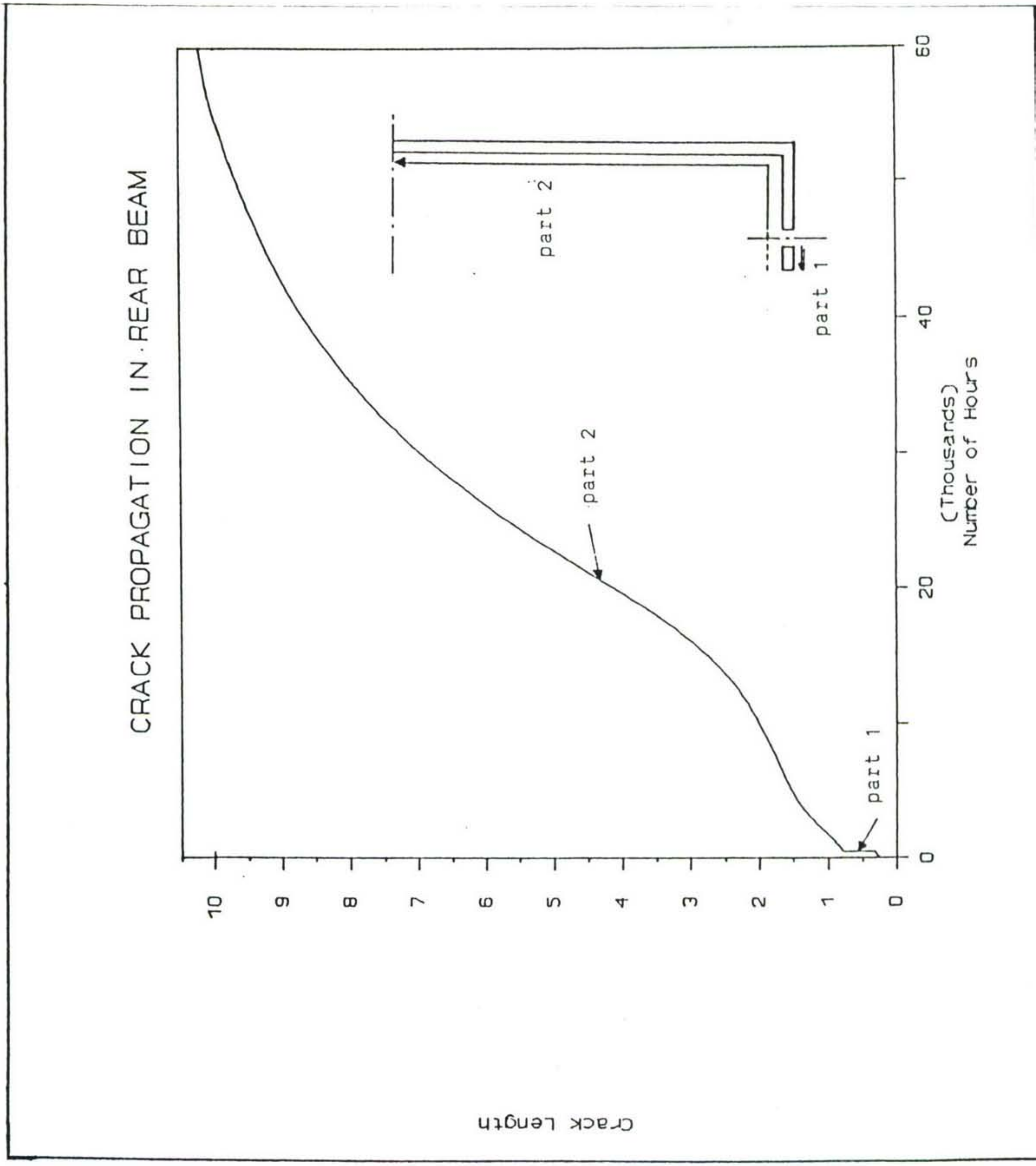
Finite Element Model of Rear Beam Showing Cracked Element:

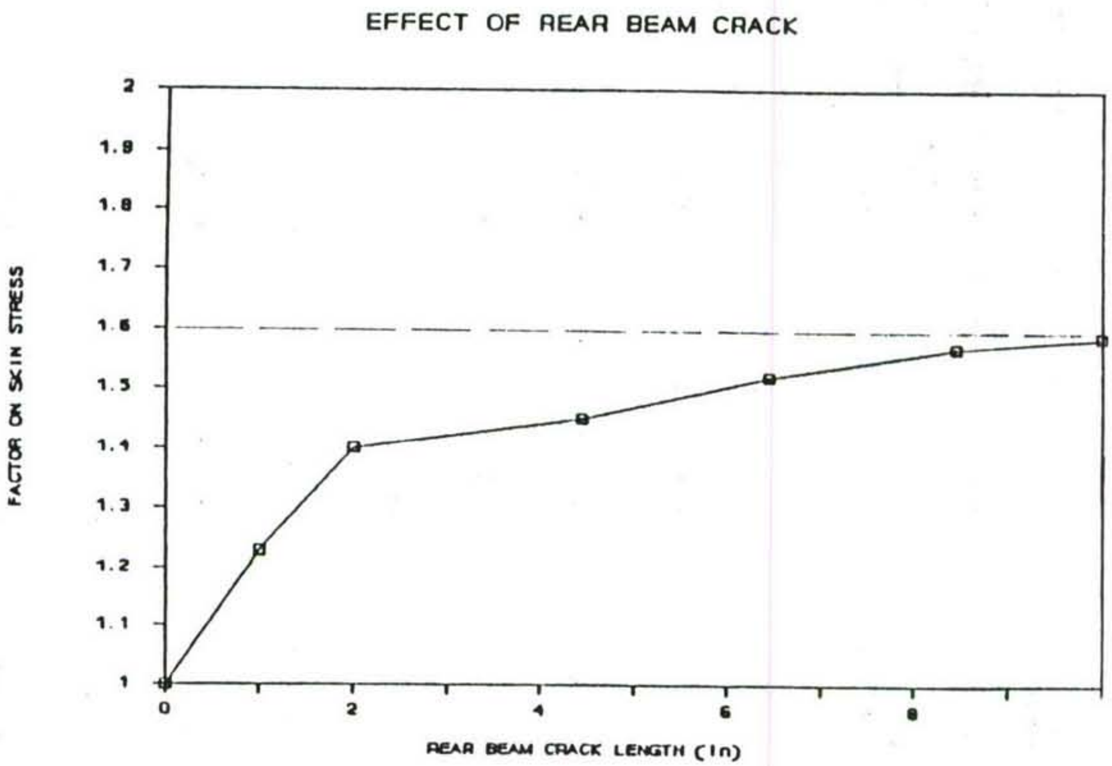


The wing skin can be considered as intact for the duration of crac: growth up to mid-height of rear beam:

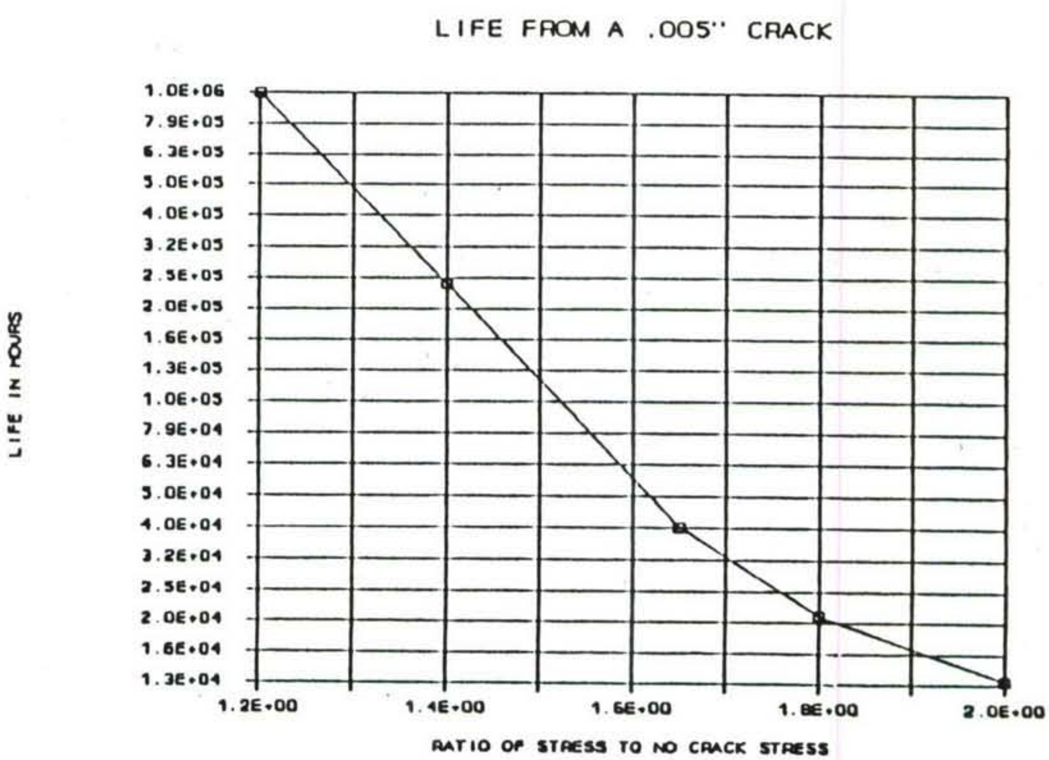
Proof: examine following relations:

Number of flight hours(N)	versus	Rear beam crack length
Rear beam crack length	versus	Factor on skin stress
Factor on skin stress	versus	Number of hours to break skin ligament


CRACK PROPAGATION IN REAR BEAM
Crack Length
Number of Hours
(Thousands)
0
1
2
3
4
5
6
7
8
9
10
0
20
40
60
part 1
part 2
part 1
part 2



Factor on skin stress is less than 1.6.

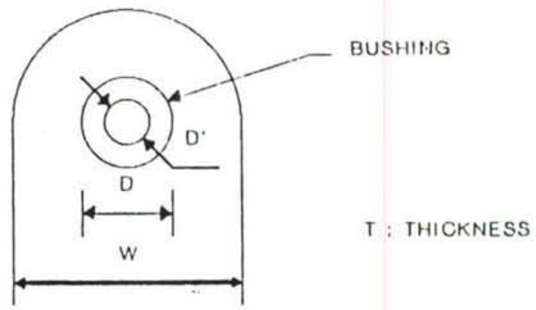


Number of hours to break skin ligament is more than 60,000.

Crack growth lives and inspection intervals for Group II:

LOCATION #	DESCRIPTION	LIFE (Hours) PART I	PART II	TOTAL	INSPECTION INTERVAL (Hours)
1	RIB STRUT	39,000	105,000	144,000	NONE
2	RIB CAP	55,500	9,000	64,500	8,000
3	RIB CAP	21,500	3,000	24,500	8,000
10	RIB POST (vertical crack)	27,000	14,500	41,500	14,000
15	RIB POST (spanwise crack)	22,500	20,000	42,500	14,000
36	WING-TO-FUSELAGE BACKUP STRUCTURE	136,500	-	136,500	NONE
37	DOUBLER ON WING-TO -WING SPLICE PLATE	85,500	20,000	105,500	NONE
39	SPLICE PLATE TO STRINGER CONNECTION	12,500	6,500	19,000	6,500
40	SKIN AT SPLICE PLATE	-	23,500	23,500	8,000
41	SPLICE PLATE	43,000	28,000	71,000	NONE

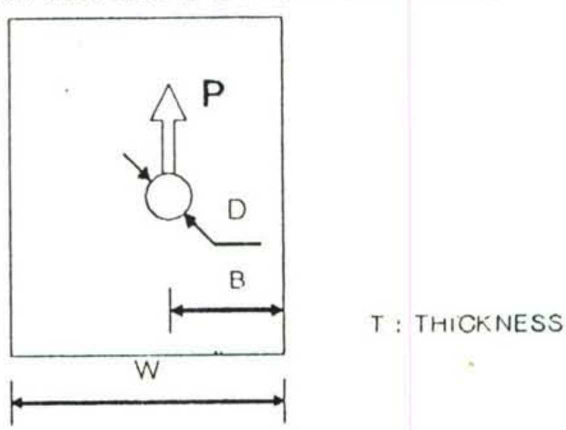
WING TO FUSELAGE LUG GEOMETRY



LOCATION #	POSITION (in.)		D (in.)	D' (in.)	T (in.)	W (in.)
21	FORWARD	BL40.1875	.812	.812	0.62	1.50
22	FORWARD	BL33.1875	1.187	1.187	1.20	1.875
23	FORWARD	BL6	1.25	1.0	0.73	1.875
24	AFT	BL 43.812	1.30	1.30	0.77	2.22
25	AFT	BL33.1875	1.20	1.20	1.20	2.00
26	AFT	BL6	1.125	0.875	0.91	1.80

LUG BACKUP STRUCTURE GEOMETRY

(CRACK FROM FIRST FASTENER HOLE ON FLANGE)



LOCATION #	T (in.)	W (in.)	D (in.)	B (in.)
31	.125	2.92	.375	.712
32	.125	2.03	.4375	.825
33	.144	2.75	.625	1.375
34	.144	3.00	.5	1.5

Crack growth lives for Group III:

LOCATION	LIFE IN HOURS
LUG 22 WITH LUG 21 FAILED	35,000
LUG 21 WITH LUG 22 FAILED	46,500
LUG 25 WITH LUG 24 FAILED	123,000
LUG 24 WITH LUG 25 FAILED	125,500
BACKUP STRUCTURE 32 WITH LUG 21 FAILED	INFINITE
BACKUP STRUCTURE 31 WITH LUG 22 FAILED	78,000
BACKUP STRUCTURE 34 WITH LUG 24 FAILED	OVER 1000000
BACKUP STRUCTURE 33 WITH LUG 25 FAILED	OVER 1000000

All initial crack lengths for above cases are .005 inch.
All lugs except lug 24 are inspectable. The crack growth life for lug 25 with lug 24 failed from an initial crack length of .05 inch is 46,000 hours.

Summary and Conclusions:

- The critical stress intensity factor was determined by test and finite element analysis to be 120 ksi $\sqrt{in.}$
- All ten (10) categories of crack configuration analyzed for residual strength were damage tolerant.
- Crack growth analysis performed on various wing components showed that the shortest crack growth life was well in excess of the inspection interval.

PROBABILISTIC FAILURE ASSESSMENT

RELIABILITY ASSESSMENT BASED ON QUANTITATIVE ENGINEERING ANALYSIS AND OPERATING EXPERIENCE

WORK PERFORMED AT JPL
SPONSORED BY
NASA HEADQUARTERS
OFFICE OF SPACE FLIGHT

MATTHEW CREAGER [1]
NICHOLAS MOORE [2]
DONALD EBBELER [2]

[1] STRUCTURAL INTEGRITY ENGINEERING
[2] JET PROPULSION LABORATORY

DEFINITION OF CERTIFICATION / QUALIFICATION

PROCESS BY WHICH THE EXPECTATION OF RELIABLE MISSION OPERATION IS ESTABLISHED

GOAL

DEVELOPMENT AND IMPLEMENTATION OF A MORE EFFECTIVE APPROACH FOR CERTIFICATION / QUALIFICATION OF RELIABLE MISSION OPERATION FOR FLIGHT SYSTEMS

APPLICABILITY

COSTLY SYSTEMS FOR WHICH EXPECTATION OF RELIABLE MISSION OPERATION MUST BE ESTABLISHED FROM LIMITED INFORMATION

AN EXAMPLE OF THE PROBLEM THAT PROBABILISTIC FAILURE ASSESSMENT IS DESIGNED TO ADDRESS

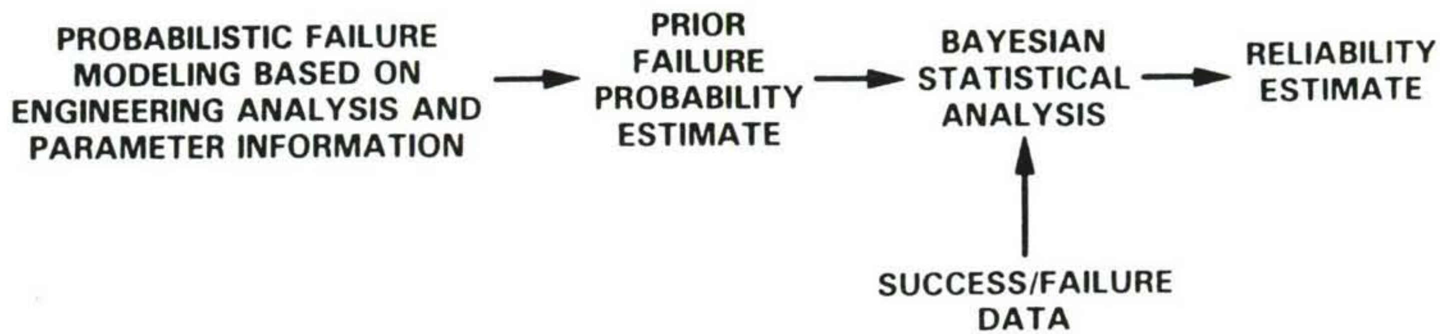
- CERTIFICATION OF THE NSTS PROPULSION SYSTEM BY TESTING ALONE IS NOT FEASIBLE BECAUSE EXTENSIVE ZERO FAILURE TESTING WOULD BE REQUIRED TO DEMONSTRATE ACCEPTABLE MISSION RELIABILITY
  - SUCH TESTING WOULD BE PROHIBITIVELY EXPENSIVE
  - DESIGN CHANGES INVALIDATE PREVIOUS TESTING
- CONVENTIONAL DETERMINISTIC APPROACHES AND METHODOLOGIES FOR FAILURE AND PREDICTION DO NOT:
  - EFFECTIVELY ACCOUNT FOR PROBABILISTIC FAILURE PHENOMENA WHEN REQUIREMENTS FORCE DEPARTURE FROM SAFETY FACTORS AND MARGINS BASED ON EXTENSIVE DIRECT EXPERIENCE
  - PROVIDE ANY CONSISTENT, QUANTITATIVE MEASURE OF RISK

THE OVERALL APPROACH CONSISTS OF:

1. IDENTIFYING EACH RELEVANT FAILURE MODE
2. GENERATING AN APPROPRIATE ENGINEERING MODEL TO DESCRIBE THAT FAILURE MODE
3. EVALUATING THE UNCERTAINTIES IN THE FAILURE MODEL DUE TO :
   - INTRINSIC VARIATION IN ELEMENTS OF THE FAILURE MODEL
   - LACK OF KNOWLEDGE ABOUT ELEMENTS OF THE FAILURE MODEL
4. A PROBABILISTIC ANALYSIS USING THE ENGINEERING MODEL AND THE UNCERTAINTIES
5. UPDATING RELIABILITY ESTIMATES USING COMPONENT AND SYSTEM LEVEL TEST RESULTS

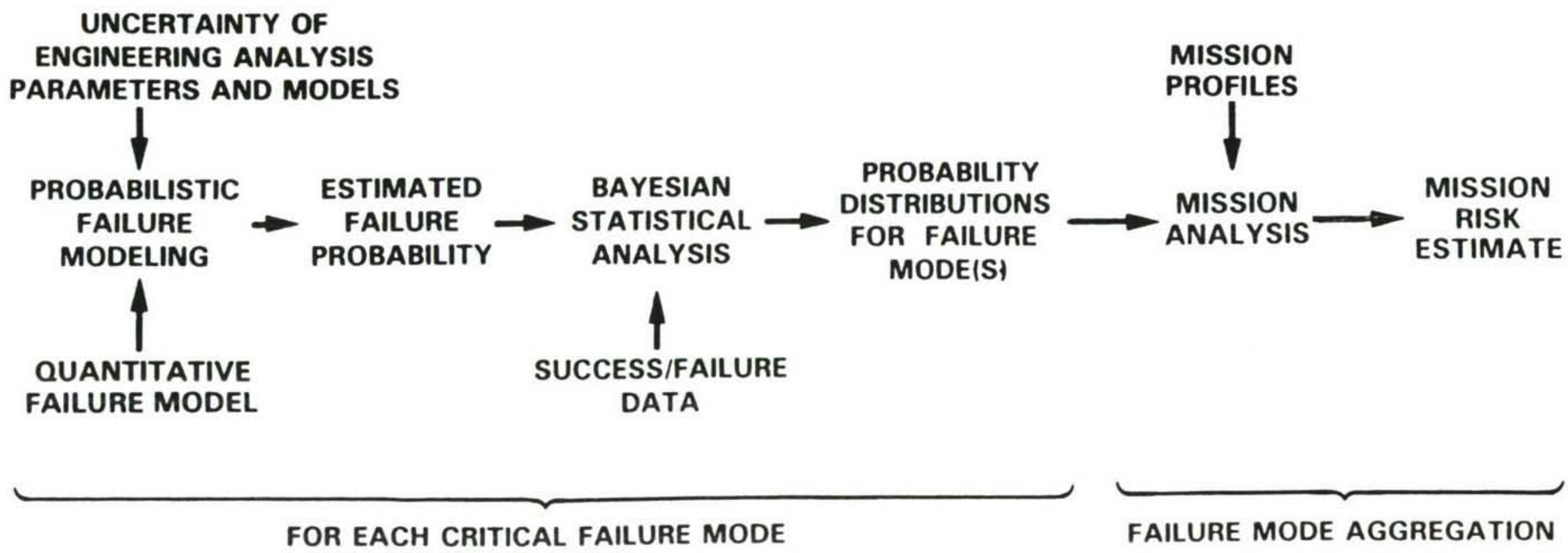
A DISTINGUISHING FEATURE OF THE PROBABILISTIC FAILURE ASSESSMENT APPROACH IS THE CAPABILITY TO INCORPORATE ALL OF THE RELEVANT ANALYTIC AND TEST INFORMATION

QUANTITATIVE RELIABILITY ESTIMATION



- SUCCESS/FAILURE DATA IS TYPICALLY A WEAK RELIABILITY INFORMATION SOURCE FOR NEW, COMPLEX, COSTLY SYSTEMS
- ENGINEERING ANALYSIS AND PARAMETER INFORMATION IS USUALLY THE MAJOR SOURCE OF RELIABILITY INFORMATION
- THE RELIABILITY ESTIMATE IS THAT WHICH IS WARRANTED BY THE AVAILABLE INFORMATION AND CAN BE UPDATED AS MORE INFORMATIN BECOMES AVAILABLE

METHODOLOGY



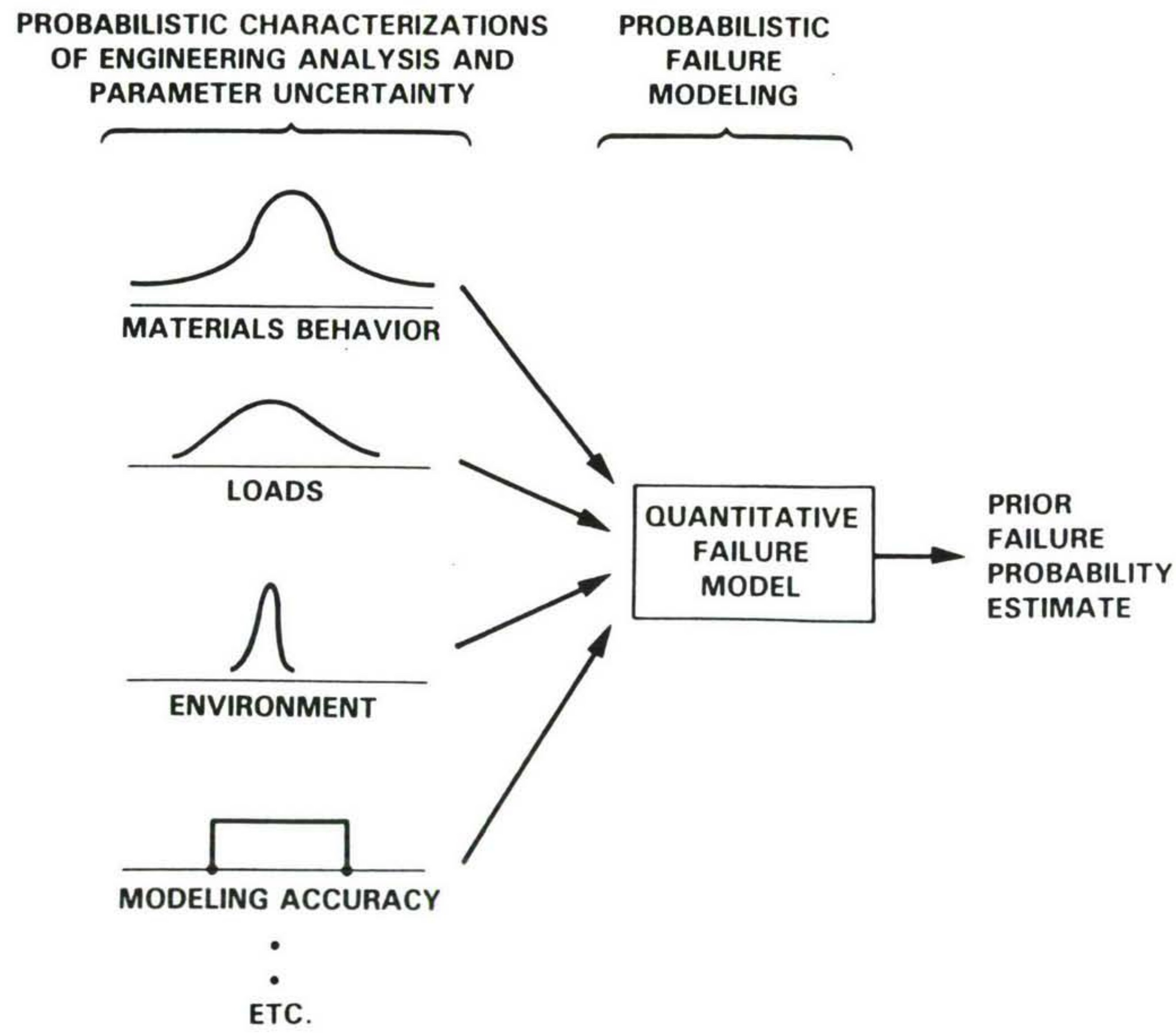
- QUANTITATIVE FAILURE MODELS ARE BASED ON THE PHYSICS/MECHANICS OF FAILURE PHENOMENA

- PROBABILISTIC ANALYSIS USING SUCH FAILURE MODELS MUST ACCOUNT FOR:

  - STOCHASTIC NATURE OF FAILURE PHENOMENA

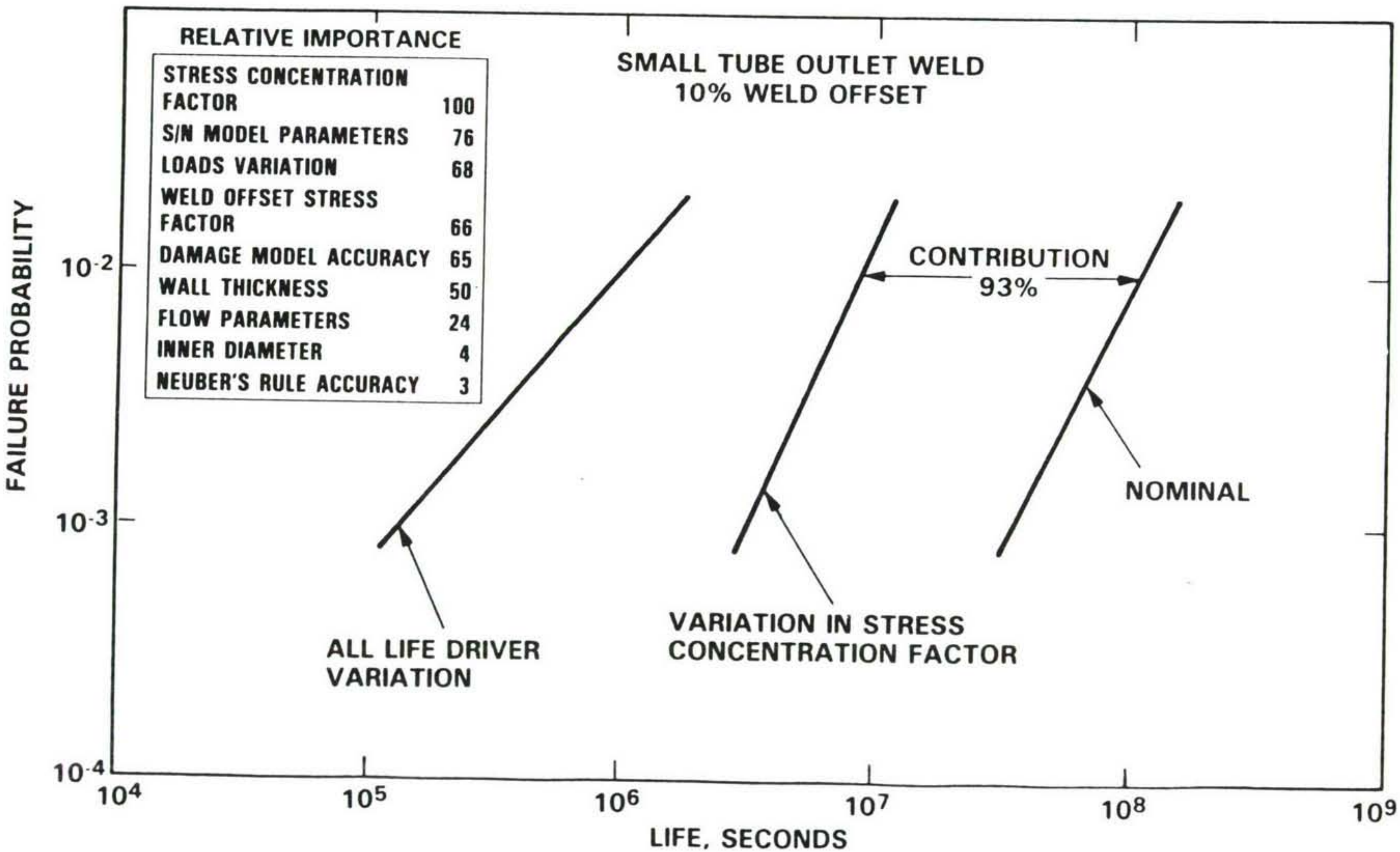
  - UNCERTAIN KNOWLEDGE OF ANALYSIS PARAMETERS

METHODOLOGY



EXAMPLES OF APPLICATION OF PROBABILISTIC FAILURE ASSESSMENT

SSME HEAT EXCHANGER COIL HCF LIFE
LIFE DRIVER SENSITIVITIES

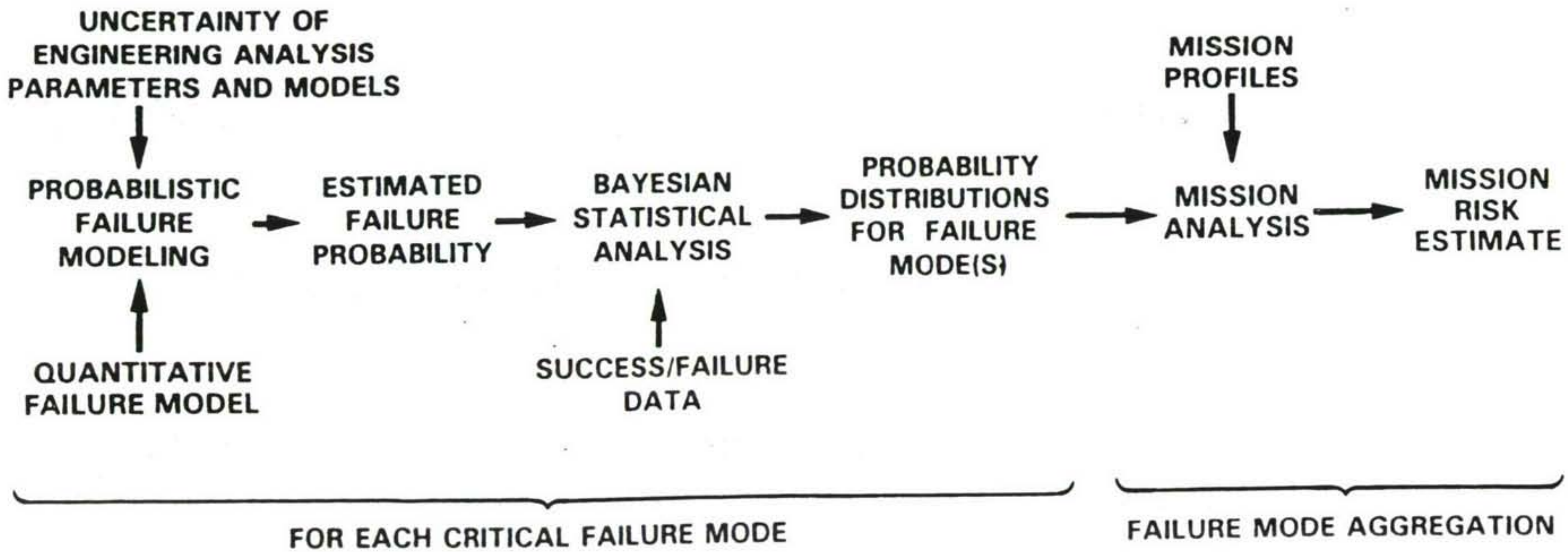


CASE STUDY EXAMPLES

COMPONENT	FAILURE MODE	ISSUES/CONCLUSIONS
HPOTP MAIN DISCHARGE DUCT	HIGH CYCLE FATIGUE (WELD NO. 6)	UNCERTAINTY OF WELD STRESS CONCENTRATION FACTOR, LOADS, AND MATERIALS PROPERTIES. ESTIMATED B.01 LIFE >90 MISSIONS FOR WORST CASE WELD OFFSET. NO ADDITIONAL PARAMETER INFORMATION REQUIRED FOR 60 MISSION SERVICE LIFE.
ATD-HPFTP TURBINE DISK (DESIGN PHASE)	LOW CYCLE FATIGUE (BLADE ATTACHMENT)	UNCERTAINTY IN COOLANT TEMPERATURE AND ENGINEERING ANALYSIS. ESTIMATED B.01 LIFE IS BELOW DESIGN GOAL AND IS NOT INFORMATION SENSITIVE. REDESIGN TO REDUCE STRESS IS REQUIRED TO INCREASE LIFE.

EXAMPLE SHOWING THE INTEGRATION OF OPERATING EXPERIENCE

METHODOLOGY



- QUANTITATIVE FAILURE MODELS ARE BASED ON THE PHYSICS/MECHANICS OF FAILURE PHENOMENA
- PROBABILISTIC ANALYSIS USING SUCH FAILURE MODELS MUST ACCOUNT FOR:
  - STOCHASTIC NATURE OF FAILURE PHENOMENA
  - UNCERTAIN KNOWLEDGE OF ANALYSIS PARAMETERS

A DETERMINISTIC DESIGN ANALYSIS WAS CONSIDERED TO DETERMINE AN ALLOWABLE VALUE FOR A DESIGN STRESS

1. A TYPICAL TRANSPORT TYPE LOADING SPECTRUM WAS USED

2. A DESIGN LIFE OF 20,000 HOURS WAS ASSUMED

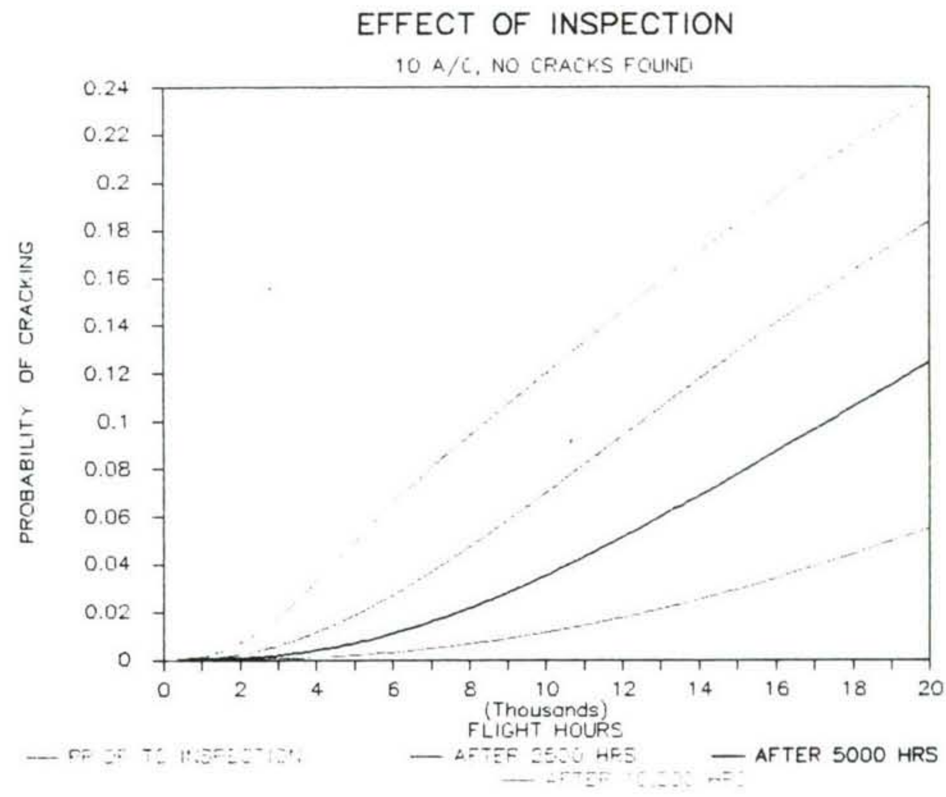
3. A VALUE OF Kt*DLS WAS CHOSEN WHICH GAVE AN ANALYTIC LIFE OF 80,000 HOURS

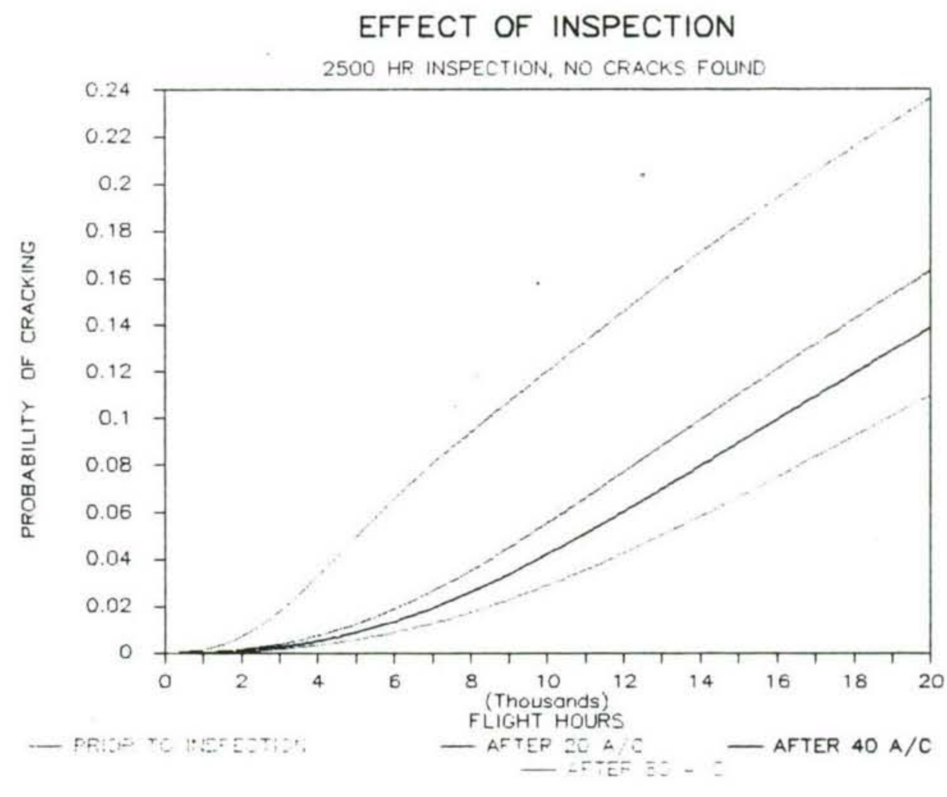
4. AVERAGE FATIGUE DATA WAS USED

THIS RESULTED IN A MODERATELY BAD DESIGN

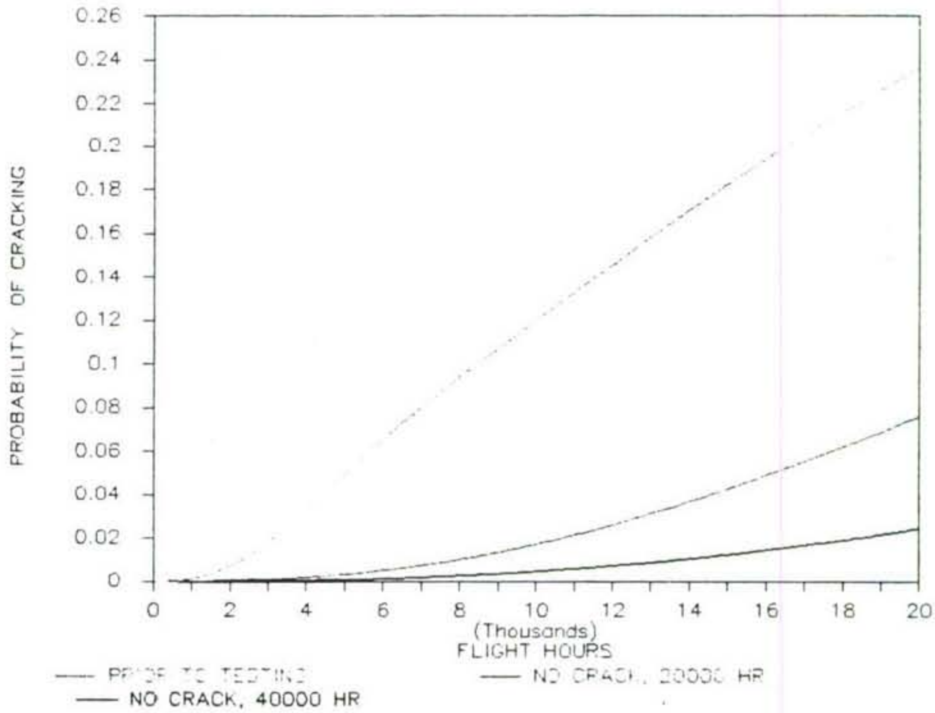
A SIMPLIFIED EXAMPLE PROBALISTIC FAILURE ANALYSIS WAS CONSIDERED:

- A FAILURE PROBABILITY WAS PREDICTED FOR THE DESIGN BY CONSIDERING:
  - THE INTRINSIC MATERIAL PROPERTY VARIATION
  - THE UNCERTAINTY DUE TO ERRORS IN Kt DETERMINATION
  - THE UNCERTAINTY DUE TO ERRORS IN DESIGN LIMIT STRESS DETERMINATION
- THE PREDICTED FAILURE PROBABILITY WAS UPDATED BY MEANS OF BAYES RULE TO SHOW:
  - THE EFFECT OF RUNNING A FULL SCALE FATIGUE TEST
  - THE EFFECT OF IN SERVICE INSPECTIONS

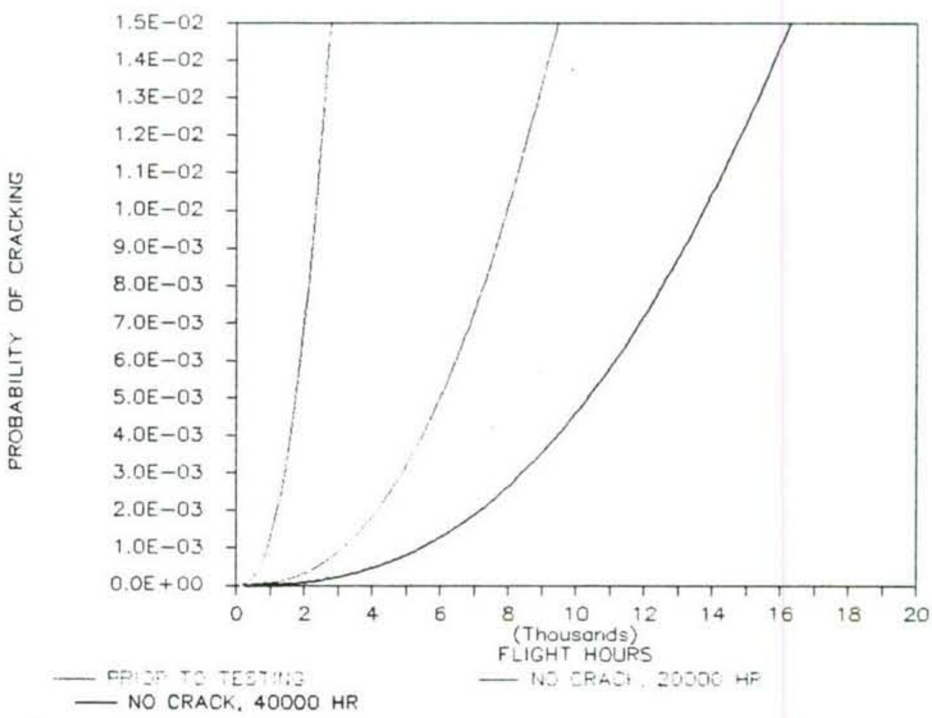

EFFECT OF INSPECTION
10 A/C, NO CRACKS FOUND
PROBABILITY OF CRACKING
0.24
0.22
0.2
0.18
0.16
0.14
0.12
0.1
0.08
0.06
0.04
0.02
0
0
2
4
6
8
10
12
14
16
18
20
(Thousands)
FLIGHT HOURS
AFTER 5000 HRS


EFFECT OF INSPECTION
2500 HR INSPECTION, NO CRACKS FOUND
PROBABILITY OF CRACKING
0.24
0.22
0.2
0.18
0.16
0.14
0.12
0.1
0.08
0.06
0.04
0.02
0
0
2
4
6
8
10
12
14
16
18
20
(Thousands)
FLIGHT HOURS
AFTER 20 A/C
AFTER 40 A/C

EFFECT OF FULL SCALE TEST



EFFECT OF FULL SCALE TEST



THIS EXAMPLE HAS SHOWN:

- THAT TEST AND OPERATING EXPERIENCE CAN BE QUANTITIVELY COMBINED WITH AN ENGINEERING BASED RELIABILITY ANALYSIS

- THAT ADDITIONAL INFORMATION CAN BE INCORPORATED AS IT BECOMES AVAILABLE

- THAT OPERATIONAL DECISIONS CAN BE BASED ON A QUANTITATIVE EVALUATION OF ALL RELEVANT INFORMATION - EVEN WHEN THE TYPES AND SOURCES (INFORMATION ARE HIGHLY DIVERSE

CONCLUSIONS

PROBABILISTIC FAILURE ASSESSMENT:

- PROVIDES FAILURE PROBABILITY ESTIMATE WHICH IS WARRANTED BY ALL THE AVAILABLE INFORMATION
- ENABLES RELIABILITY TO BE ANALYZED QUANTATIVELY IN THE DESIGN PROCESS
- ENABLES THE QUANTITATIVE EVALUATION OF OPTIONS FOR CORRECTIVE ACTION TO CONTROL UNACCEPTABLE RISK, e.g., DESIGN CHANGES VIS-A-VIS LIFE LIMITS
- PROVIDES FOR MORE INFORMED DECISIONS REGARDING ALLOCATION OF PROGRAM RESOURCES ON A COST/ BENEF BASIS
- PROVIDES A STRUCTURE FOR INFORMATION FROM POST OPERATION INSPECTIONS AND INSTRUMENTED OPERATION TO BE FED BACK INTO THE ENGINEERING ANALYSIS ON WHICH RISK ASSESSMENT IS BASED

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE

SAN ANTONIO, TEXAS

5-7 DECEMBER 1989

DR. F. D. EICHENBAUM, LASC-GA

J. W. CHAPMAN, LASC-GA

D. O. HAMMOND, WR-ALC

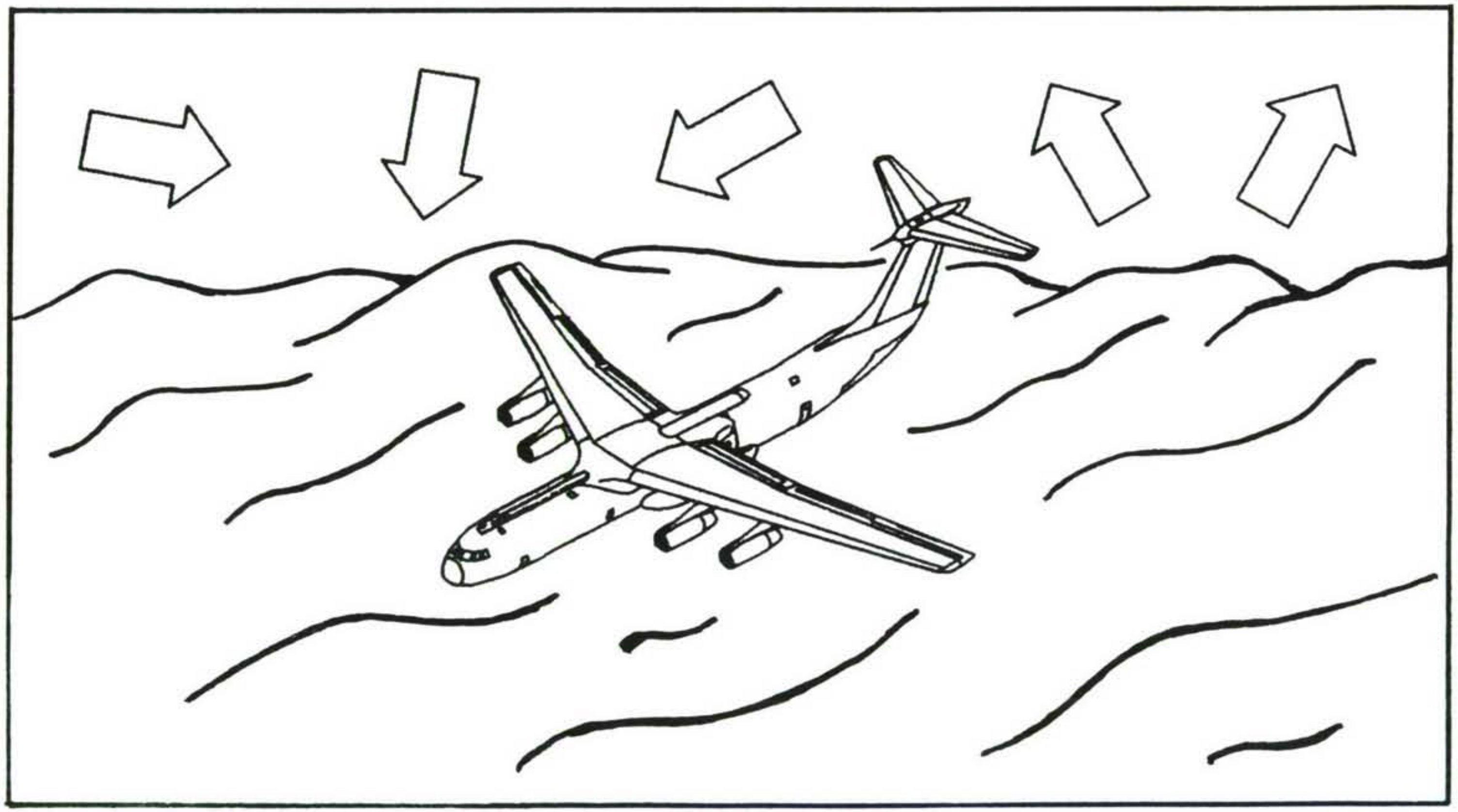
HIGH SPEED LOW LEVEL FLIGHT REGIME

Conventional repeated loads models separate gust and maneuver effects in recorded time histories based on the duration of vertical load factor excursions. Those excursions of vertical load factor which exceed a given duration are attributed to maneuver and the remaining excursions are attributed to gust. The associated maneuver loads are derived from a quasistatic analysis, whereas the gust loads are based on a dynamic model in which the gust excitation is a Gaussian random process having a specified power spectrum. Because the two load sources are assumed to be independent, corresponding predicted load peak distributions are combined by superposition.

Frequency domain analyses of flight measured data from the C-141 Severe Missions Loads Recording Program (SMLRP) show that this approach does not work well in the high speed low level flight regime.

A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

HIGH SPEED LOW LEVEL FLIGHT REGIME



OUTLINE

A procedure has been developed to remove the contribution of pilot control inputs from time histories of measured aircraft response. This procedure utilizes frequency response functions derived from a theoretical model of the aircraft corresponding to the prevailing flight conditions. Techniques are included to (1) correct discrepancies between the theoretical model and the actual aircraft by analysis of gust-free flight data, (2) validate the theoretical model for gust response, and (3) develop pilot models for flight in gust and assess their effects on loads.

The more representative model of the pilot recognizes that one of his functions consists of responding to aircraft motions, and the other consists of providing independent control inputs unrelated to aircraft motion. The first function is most appropriately incorporated into the dynamic model, while the second is treated as maneuvering in the normal sense. The new procedure for separating gust and maneuver loads allows such a pilot model to be constructed.

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

OUTLINE

o CONVENTIONAL SEPARATION PROCEDURES

o NEW SEPARATION PROCEDURE

-- THEORETICAL MODEL CALIBRATION
-- GUST VALIDATION
-- PILOT MODEL DEVELOPMENT

o DEMONSTRATION

o ADVANTAGES AND CONCLUSIONS

TYPICAL SEPARATION PROCEDURE

The conventional method of estimating aircraft loads during flight involves recording and peak counting the load factor measured by a vertical accelerometer located near the center of gravity of the aircraft. Peaks that are of less than two seconds duration are assumed to result from dynamic gust response, and the remainder are attributed to maneuver activity. Both sets of peaks are further segregated and accumulated by flight parameter block, which consists of fuel, cargo, Mach number and altitude.

A theoretical dynamic model of the aircraft is then applied to the gust load factor peaks within each flight parameter block to infer the corresponding vertical gust velocity peak spectrum. These spectra are accumulated within altitude bands to provide the environmental criteria for gust. Maneuver peaks are usually classified by type of activity.

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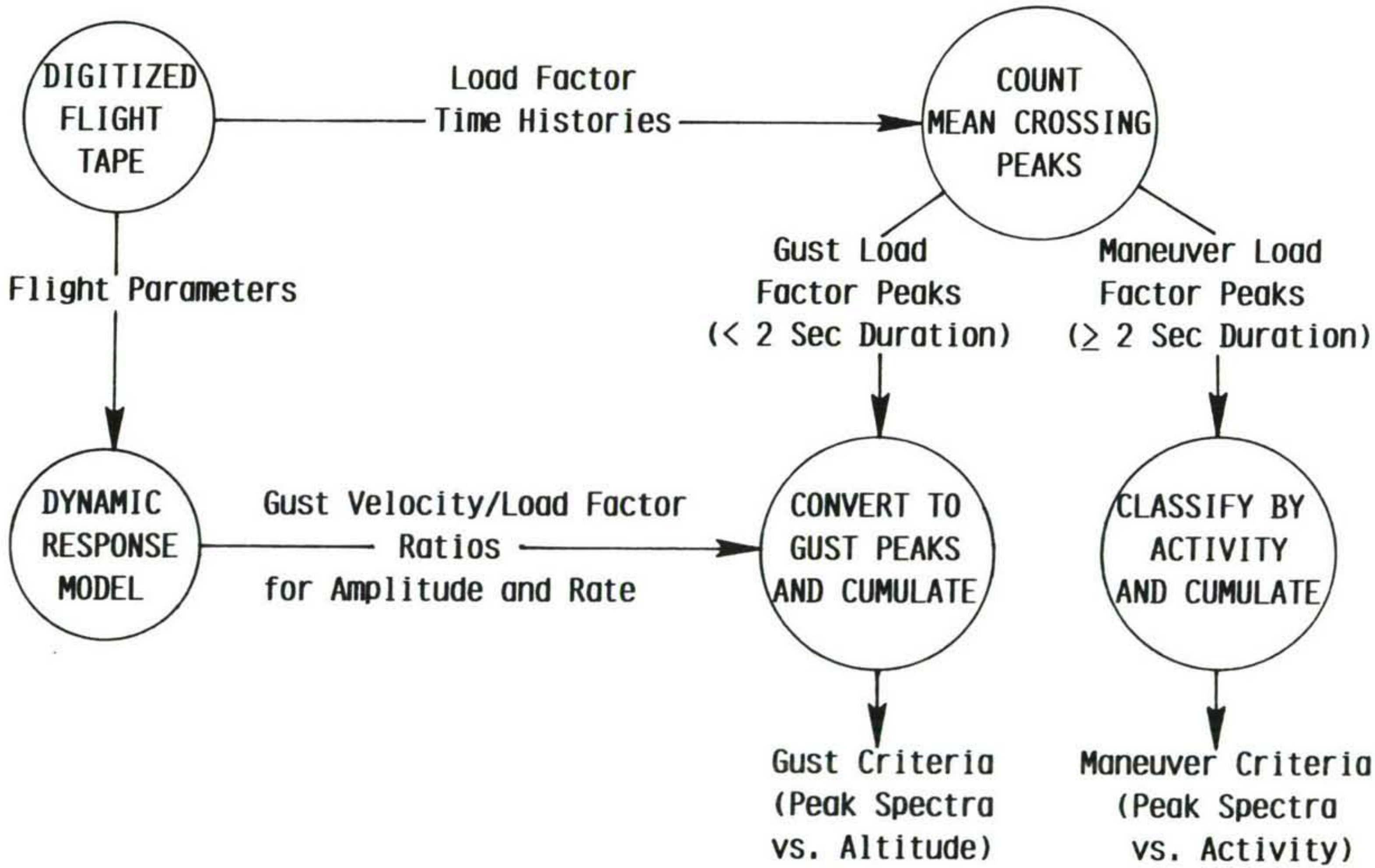
A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

TYPICAL SEPARATION PROCEDURE



STRESS PREDICTION BASED ON USAGE

Aircraft usage may be expressed in terms of time spent within each flight parameter block. A stress peak spectrum due to gust may be derived from the theoretical dynamic response model, once the gust criteria and the time in each flight parameter block have been established. A stress peak spectrum due to maneuver may be derived from a theoretical quasistatic model, once the load factor peaks due to maneuver are available. The total peak spectrum of the stress is then obtained by superimposing the peak spectra from the two sources.

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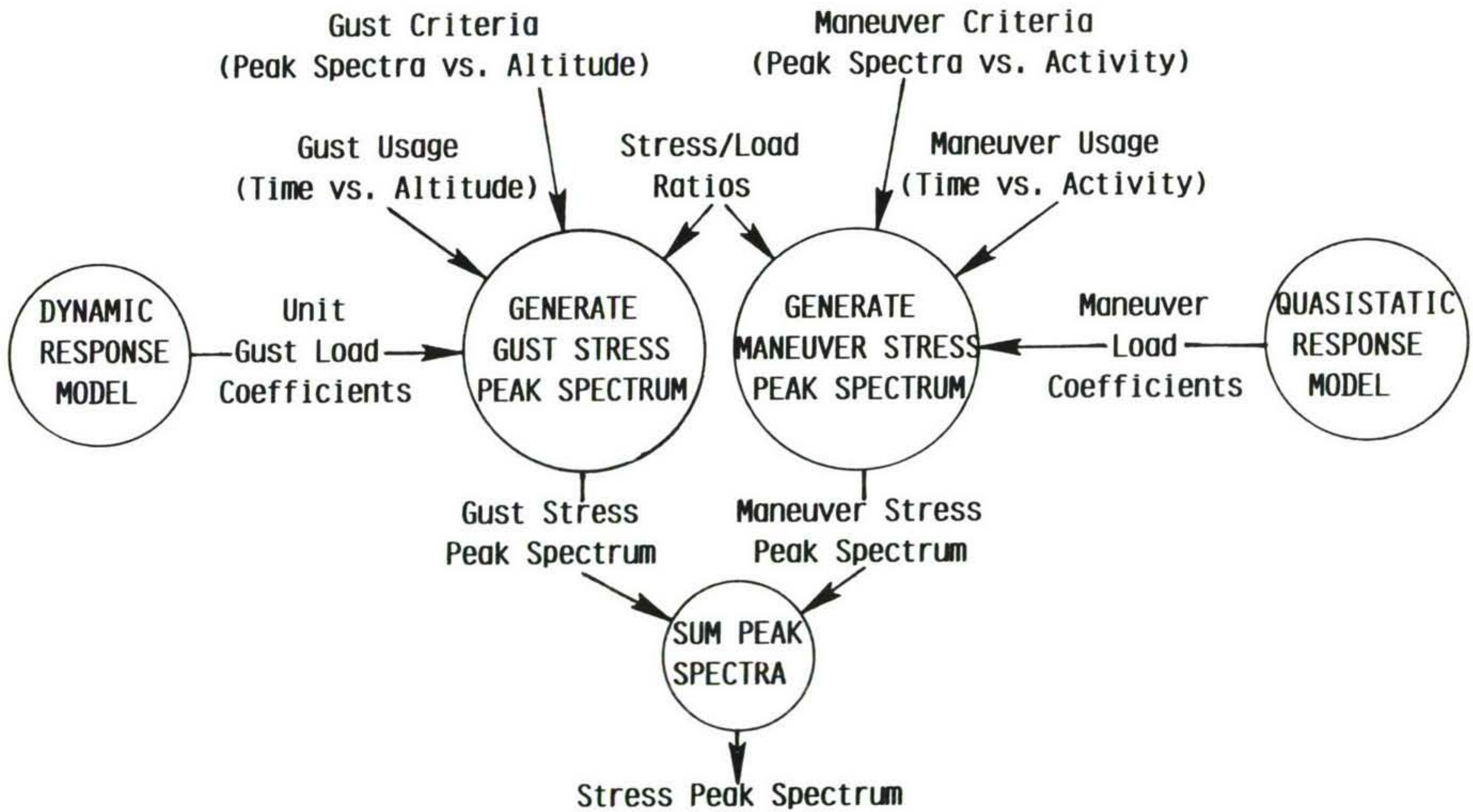
A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

STRESS PREDICTION BASED ON USAGE



CHARACTERISTICS OF TYPICAL FLIGHT LOADS

The conventional gust-maneuver separation was designed for aircraft usage in which maneuvering occurs only occasionally. Consequently, the peaks attributable to gust have very little contamination due to the pilot, and only a small proportion of the total gust peaks are masked by the longer maneuver peaks.

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

CHARACTERISTICS OF TYPICAL FLIGHT LOADS

- TYPICAL GUST LOADS TEND TO COVER A WIDE RANGE OF AMPLITUDES AND TO BE RELATIVELY UNCONTAMINATED BY PILOT INDUCED LOADS.
- TYPICAL PILOT INDUCED LOADS TEND TO BE MODERATE AND INFREQUENT, SO THAT MASKING OF GUST PEAKS IS NEGLIGIBLE.
- CONSEQUENTLY, CONVENTIONAL PROCEDURES FOR SEPARATING GUST AND MANEUVER LOADS ARE ADEQUATE FOR ORDINARY FLIGHT.

CHARACTERISTICS OF SEVERE MISSION LOADS

Conversely, in certain severe usage such as terrain following, loads due to gust and loads induced by pilot control tend to be large and to occur simultaneously. Because the current method of separation distinguishes between the two load sources on the basis of peak duration alone, it does not allow for simultaneous application of two disparate load distributions which may be in or out of phase.

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

CHARACTERISTICS OF SEVERE MISSION LOADS

- GUST AND PILOT INDUCED LOADS DURING SEVERE MISSIONS TEND TO BE LARGE, TO OCCUR SIMULTANEOUSLY, TO HAVE DISPARATE DISTRIBUTIONS WHICH MAY BE IN OR OUT OF PHASE WITH EACH OTHER, AND TO HAVE PEAKS WHICH MASK ONE ANOTHER.
- PILOT INDUCED LOADS WHICH ARE STATISTICALLY COHERENT WITH GUST LOADS MAY BE SIGNIFICANT.
- CONSEQUENTLY, CONVENTIONAL PROCEDURES FOR SEPARATING GUST AND MANEUVER ARE INAPPROPRIATE FOR APPLICATION TO SEVERE MISSION FLIGHT LOADS.

SOLUTION OF THE SEVERE MISSIONS PROBLEM

If it were possible to separate the initial load factor time history into its gust and maneuver constituents, then this capability might be applied to solve the severe missions problem. It might also allow the introduction of frequency domain analysis to generate a linear control system description of the coherent pilot for inclusion in the dynamic response model.

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

SOLUTION OF THE SEVERE MISSIONS PROBLEM

- o IF THE LOAD FACTOR TIME HISTORY COULD BE SEPARATED INTO ITS GUST AND MANEUVER COMPONENTS PRIOR TO PEAK COUNTING, THEN THE SIMULTANEOUS STRESS CONTRIBUTIONS FROM BOTH SOURCES COULD BE COMPUTED AND SUMMED.

- o THIS MIGHT ALSO ALLOW THE USE OF FREQUENCY DOMAIN METHODS TO EXTRACT A CONTROL SYSTEM DESCRIPTION OF THE COHERENT PILOT WHICH COULD BE INCORPORATED INTO THE THEORETICAL DYNAMIC RESPONSE MODEL.

NEW SEPARATION METHOD

A new method has been developed to accomplish this separation. It requires time history recording of the load factor along with the control surface deflection by which the pilot alters the load factor. The frequency response function connecting load factor to control surface deflection is computed from the dynamic response model and applied to the recorded control surface deflection to obtain the load factor induced by the pilot. Subtraction of this result from the recorded load factor yields the load factor due to gust only.

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

NEW SEPARATION METHOD

- RECORD ELEVATOR DEFLECTION TIME HISTORY ALONG WITH LOAD FACTOR TIME HISTORY AND FLIGHT PARAMETERS (FUEL, CARGO, MACH, ALTITUDE).
- EMPLOY ANALYTICAL DYNAMIC RESPONSE MODEL TO GENERATE FREQUENCY RESPONSE FUNCTION FOR LOAD FACTOR DUE TO ELEVATOR DEFLECTION, USING RECORDED FLIGHT PARAMETERS.
- COMPUTE TIME HISTORY OF PILOT INDUCED LOAD FACTOR BY FILTERING THE RECORDED ELEVATOR DEFLECTION TIME HISTORY WITH THE ABOVE FREQUENCY RESPONSE FUNCTION.
- COMPUTE TIME HISTORY OF GUST INDUCED LOAD FACTOR BY SUBTRACTING THE COMPUTED TIME HISTORY OF PILOT INDUCED LOAD FACTOR FROM THE RECORDED LOAD FACTOR TIME HISTORY.

NEW SEPARATION PROCEDURE

The load factor separation procedure is illustrated here. The mathematical operations are performed interactively by means of the ISIS* software package which was developed to analyze flight measured data from the C-141 Severe Mission Loads Recording Program (SMLRP).

*Interactive System Identification and Simulation

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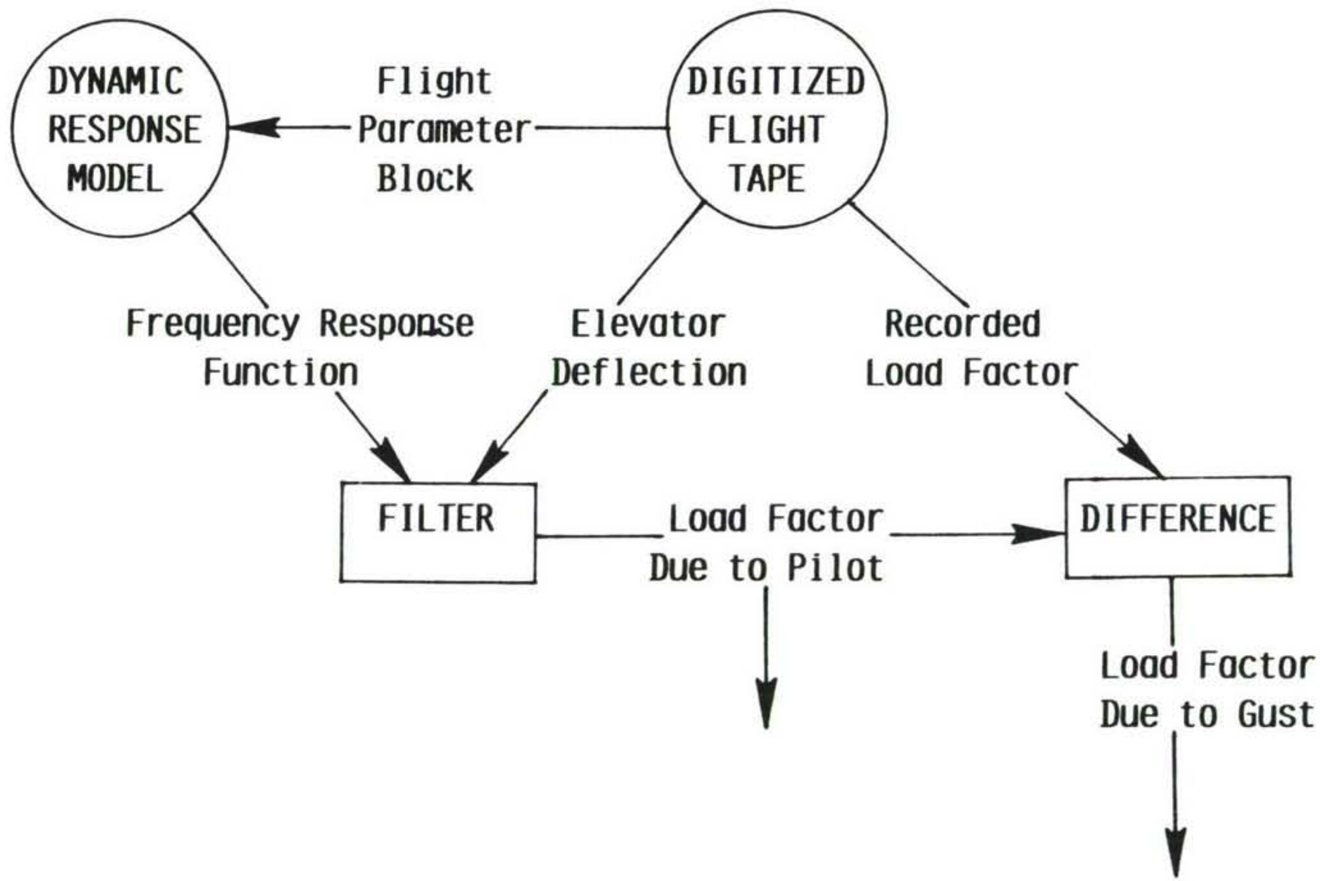
A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

NEW SEPARATION PROCEDURE



INTERACTIVE CAPABILITIES PROVIDED BY MAIN SUBROUTINES OF ISIS PROGRAM

The ISIS program was designed to extract from time history records of aircraft mounted sensors, a maximum amount of useful information by the most efficient means. The program is connected to dynamic response software to furnish independent theoretical models of the actual system undergoing analysis. All time domain operations whose speed and precision would benefit by executing them in the frequency domain are so treated. Transformation of data in and out of frequency domain for this purpose is accomplished by a continuous overlap Fast Fourier Transform (FFT) process. Interactive operation is thereby facilitated by virtually eliminating delays encountered during conventional processing.

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

INTERACTIVE CAPABILITIES PROVIDED BY MAIN SUBROUTINES OF ISIS PROGRAM

- READ TAPE. Read, store, and display data from header record of digitized flight tape. Download and demultiplex requested time histories of various sampling rates and place them in direct access files of uniform record length.

- ANALYZE TIME HISTORIES. Compute cross spectra and power spectra of time histories. Derive root mean square amplitudes and zero crossing rates from power spectra.

- FILTER TIME HISTORIES. Employ theoretical frequency response functions to filter time histories. Introduce high and/or low pass cutoff frequencies. Increase or decrease sampling rate. Differentiate or integrate. Generate Gaussian random time histories having desired frequency properties.

- COMBINE TIME HISTORIES. Combine time histories by factored addition, multiplication or division. Compute the root mean square amplitudes and correlation coefficient matrix between corresponding segments of selected time histories, and utilize these data to perform a multivariate least squares fit.

- PLOT. Plot time histories, spectra and filters. Complex data may be plotted either in amplitude-phase or in real-imaginary form.

- LIST FILES. Show information about data files generated by the program. These consist of time histories, spectra, and filters.

LEAST SQUARES FITTING OF TIME HISTORIES

The least squares fitting process by which time histories chosen as dependent are expressed as linear combinations of independent time histories is governed by a simple matrix equation relating the fitting coefficients to the correlations between the independent variables and the correlations between independent and dependent variables. The correlations may be reduced to their corresponding root mean squares (RMS) amplitudes and correlation coefficients for further convenience.

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

LEAST SQUARES FITTING OF TIME HISTORIES

A SET OF CONSTANT COEFFICIENTS WHICH EXPRESS K DEPENDENT TIME HISTORIES IN TERMS OF J INDEPENDENT TIME HISTORIES IS GIVEN BY THE MATRIX EQUATION

$$\underset{N\times J}{[X]}\ \underset{J\times K}{[C]} = \underset{N\times K}{[Y]}$$

WHERE [C] IS THE MATRIX OF COEFFICIENTS AND THE COLUMNS OF [X] AND [Y] REPRESENT, RESPECTIVELY, INDEPENDENT AND DEPENDENT DISCRETE N POINT TIME HISTORIES. IF $N \geq J$, THEN THE LEAST SQUARES SOLUTION FOR THE COEFFICIENTS IS GIVEN BY

$$\underset{J\times K}{[C]} = (\underset{J\times N}{[X]^T}\ \underset{N\times J}{[X]})^{-1}\ \underset{J\times N}{[X]^T}\ \underset{N\times K}{[Y]}$$

$$= \underset{J\times J}{[R_{xx}]^{-1}}\ \underset{J\times K}{[R_{xy}]}$$

$$= \underset{J\times J}{\lfloor \sigma_x \rfloor^{-1}}\ \underset{J\times J}{[\rho_{xx}]^{-1}}\ \underset{J\times K}{[\rho_{xy}]}\ \underset{K\times K}{\lfloor \sigma_y \rfloor}$$

WHERE $[R_{xy}]$ AND $[R_{xy}]$ ARE TIME HISTORY CORRELATION MATRICES, $[\rho_{xx}]$ AND $[\rho_{xy}]$ ARE CORRESPONDING CORRELATION COEFFICIENT MATRICES, AND $\lfloor \sigma_x \rfloor$ AND $\lfloor \sigma_y \rfloor$ ARE DIAGONAL MATRICES CONTAINING THE ROOT MEAN SQUARE AMPLITUDES.

FLIGHT CALIBRATION OF DYNAMIC RESPONSE MODEL

Feasible application of a theoretical dynamic response model to the analysis of flight recorded time histories requires extremely close correspondence between the theoretical model and the aircraft. Such close agreement may be achieved by utilizing time histories from still air maneuver flights to calibrate the theoretical model.

The first calibration step establishes a separate correction factor for the deflection of each control surface by reconciling the most closely associated computed responses such as c.g. vertical or roll accelerations with their measured counterparts. These factors are retained in a second calibration step to determine correction factors which reconcile other computed responses such as loads or stresses with their measured counterparts. The correction factors computed for our current dynamic response model turn out to be particularly insensitive to prevailing flight conditions.

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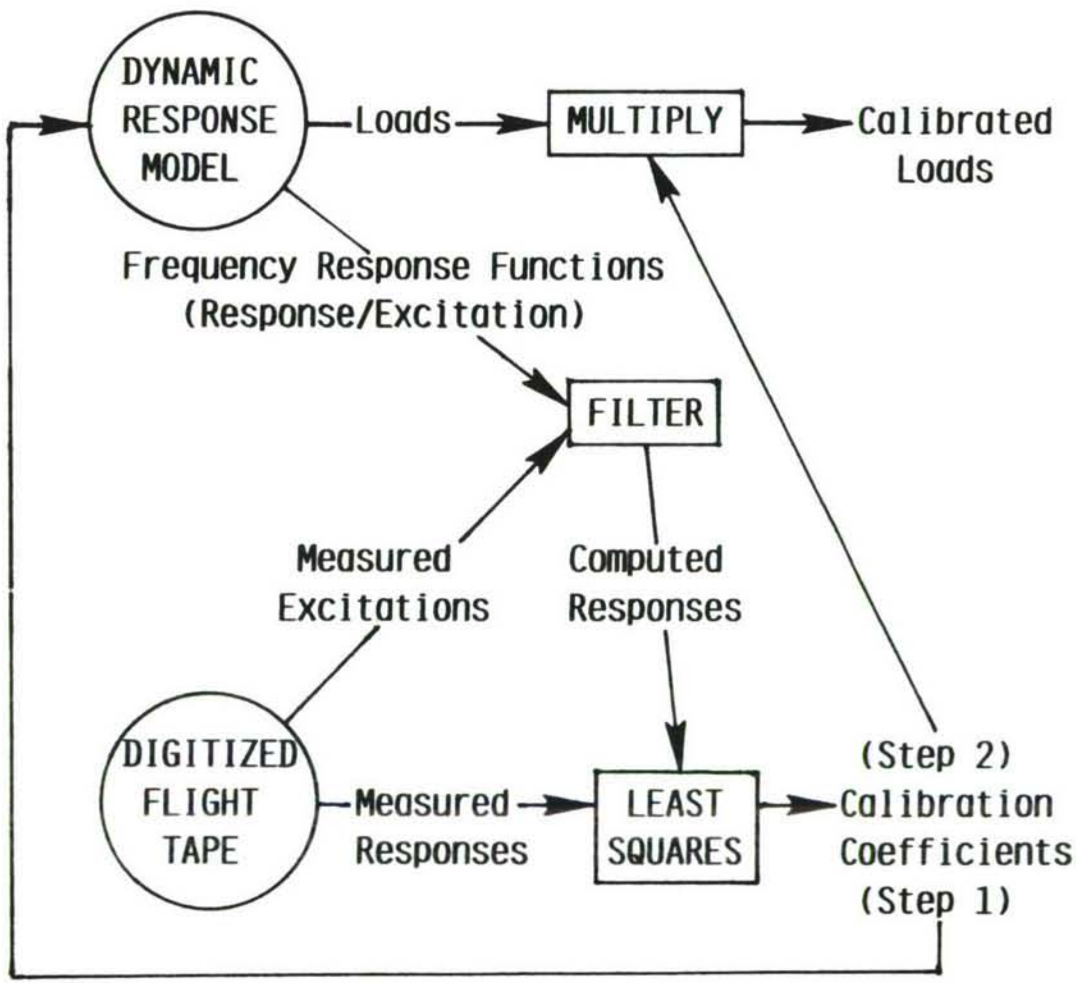
A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

FLIGHT CALIBRATION OF DYNAMIC RESPONSE MODEL

GENERAL PROCEDURE



CALIBRATION SEQUENCE

EXCITATIONS

Control Surface Deflections:
- Elevator
- Aileron

RESPONSES

Step 1:
- Vertical Acceleration
- Roll Acceleration

Step 2:
- Loads and/or Stresses

GUST VALIDATION OF DYNAMIC RESPONSE MODEL

Once the theoretical dynamic response model has been calibrated through still air maneuver analysis, it may be validated for gust response by analyzing data collected during flight in turbulence. This is accomplished by employing the separation procedure to extract the gust induced portions of time histories measured by sensors mounted on the aircraft. These gust response time histories are then frequency analyzed to obtain power spectra, RMS amplitudes and zero crossing rates. Equivalent theoretical data is generated by the theoretical model using a von Karman unit gust spectrum. Finally, a comparison between the measured and theoretical data is performed using vertical acceleration at the center of gravity (load factor) as the normalizing parameter.

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A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

GUST VALIDATION OF DYNAMIC RESPONSE MODEL

- o EXTRACT GUST RESPONSE TIME HISTORIES FROM MEASURED DATA, INCLUDING LOAD FACTOR
- o SPECTRALLY ANALYZE THESE TIME HISTORIES TO OBTAIN:
  - -- POWER SPECTRA
  - -- RMS AMPLITUDES
  - -- ZERO CROSSING RATES
- o GENERATE THE CORRESPONDING DATA USING THE THEORETICAL GUST RESPONSE MODEL WITH A VON KARMAN UNIT GUST INPUT SPECTRUM
- o NORMALIZE MEASURED SPECTRAL DATA TO UNIT GUST VELOCITY BASED ON UNIT THEORETICAL TO MEASURED RMS LOAD FACTOR RATIO, AND COMPARE RESULTS WITH THEORETICAL SPECTRAL DATA TO VALIDATE THEORETICAL GUST RESPONSE MODEL

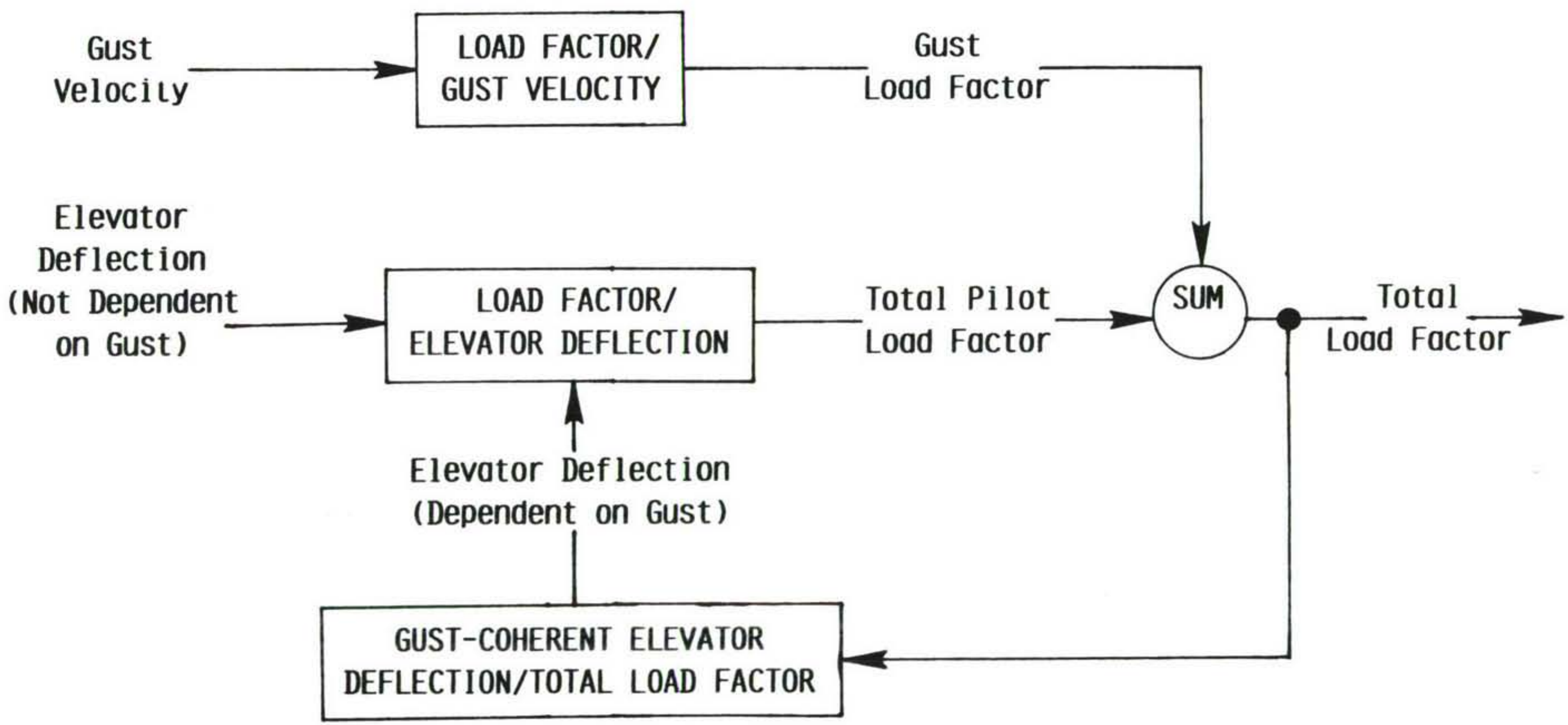
PILOT FEEDBACK MODEL

As mentioned previously, the ability to separate the gust and pilot induced load factors enables us by an additional process to incorporate the gust-coherent portion of the pilot input as a feedback function within the control system of the aircraft response model. The diagram shows the entire pilot model associated with pilot response to vertical acceleration, whereas the theoretical gust response model with coherent pilot feedback does not include the incoherent pilot input.

A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

PILOT FEEDBACK MODEL

FREQUENCY RESPONSE FUNCTIONS:



FREQUENCY DOMAIN ANALYSIS OF PILOT FEEDBACK

Separation with feedback ignored has been described and is expressed by two algebraic equations in the frequency domain, which are shown in the upper right hand corner of the figure. The gust-coherent pilot feedback function is then derived as the ratio between complex product pairs, which may be replaced by their corresponding cross spectra for implementation. Finally, the gust load factor with feedback ignored must undergo an additional filtering to include feedback.

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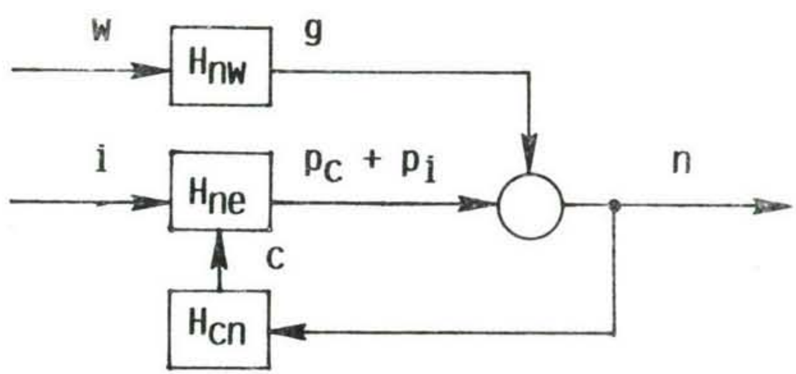
A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

FREQUENCY DOMAIN ANALYSIS OF PILOT FEEDBACK



H_{xy} = frequency response function of x due to y

COMPLEX AMPLITUDES

c, i = coherent & incoherent elevator deflections $(cg^* = 0)$, $(ig^* = 0)$

e = c+i = total elevator deflection

g = gust load factor

n = total load factor

p_c, p_i = coherent and incoherent pilot load factors

w = vertical gust velocity

SEPARATION WITHOUT FEEDBACK

$$g = n - (p_c + p_i)$$

$$p_c + p_i = H_{ne}\, e$$

DERIVATION OF FEEDBACK FUNCTION

$$eg^* = (c + i)g^* = cg^*$$

$$= H_{cn}\,(g + p_c)g^*$$

$$ng^* = (g + p_c + p_i)g^*$$

$$= (g + p_c + H_{ne}\, i)g^*$$

$$= (g + p_c)g^*$$

$$H_{cn} = eg^*/ng^*$$

SEPARATION WITH FEEDBACK

$$g + p_c = (1 - H_{cn} H_{ne})^{-1}\, g$$

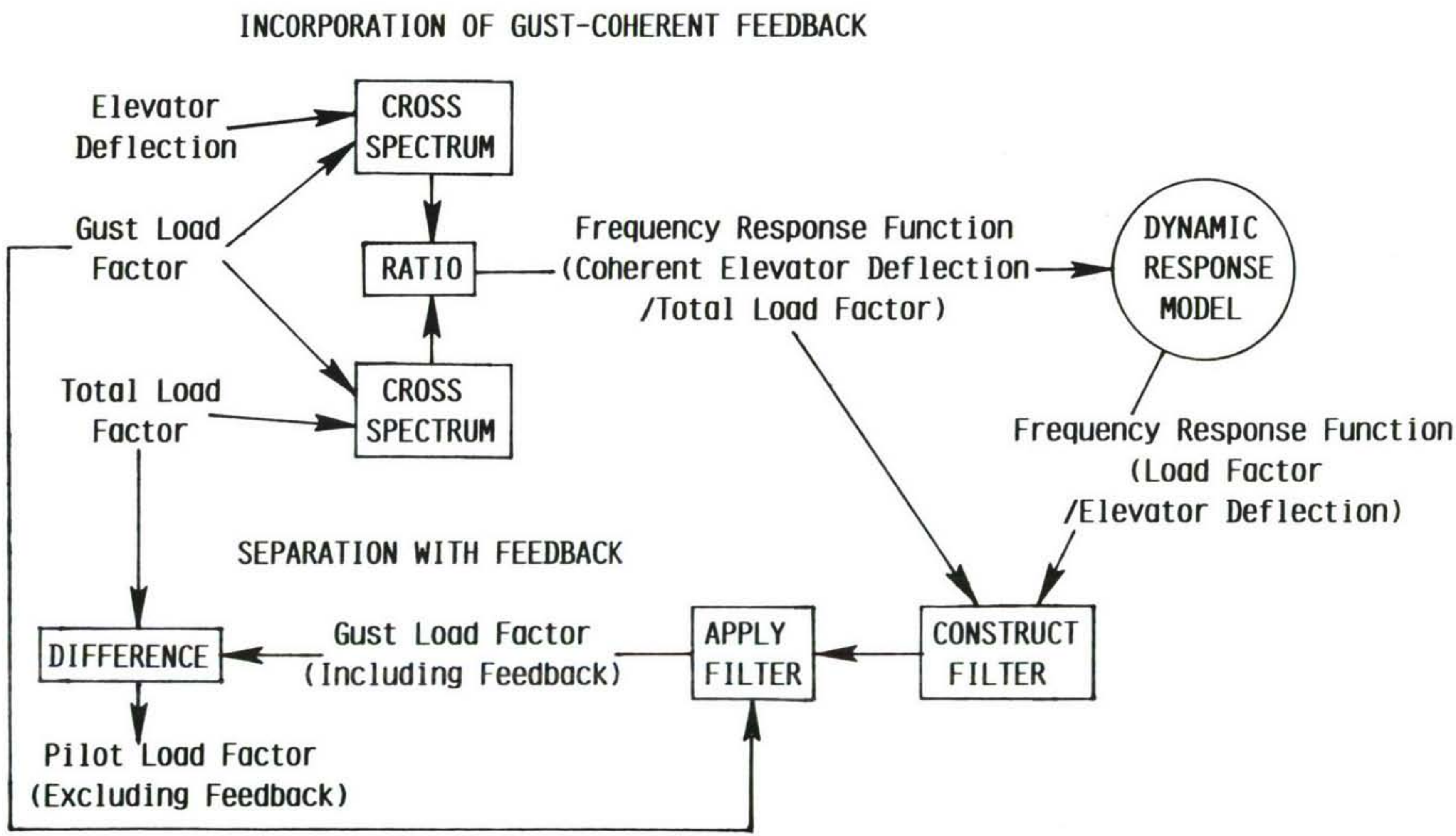
$$p_i = n - (g + p_c)$$

IMPLEMENTATION OF PILOT FEEDBACK

The upper portion of this figure illustrates the implementation procedure used to determine gust-coherent pilot feedback from flight recorded data and express it as a frequency response function for direct insertion into the control system of the theoretical dynamic response model. The lower portion of the figure shows the corresponding processing of the separated load factor components to obtain a modified gust component that includes feedback, and a modified pilot component that excludes it. The two steps are performed consecutively using the ISIS software package.

A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

IMPLEMENTATION OF PILOT FEEDBACK

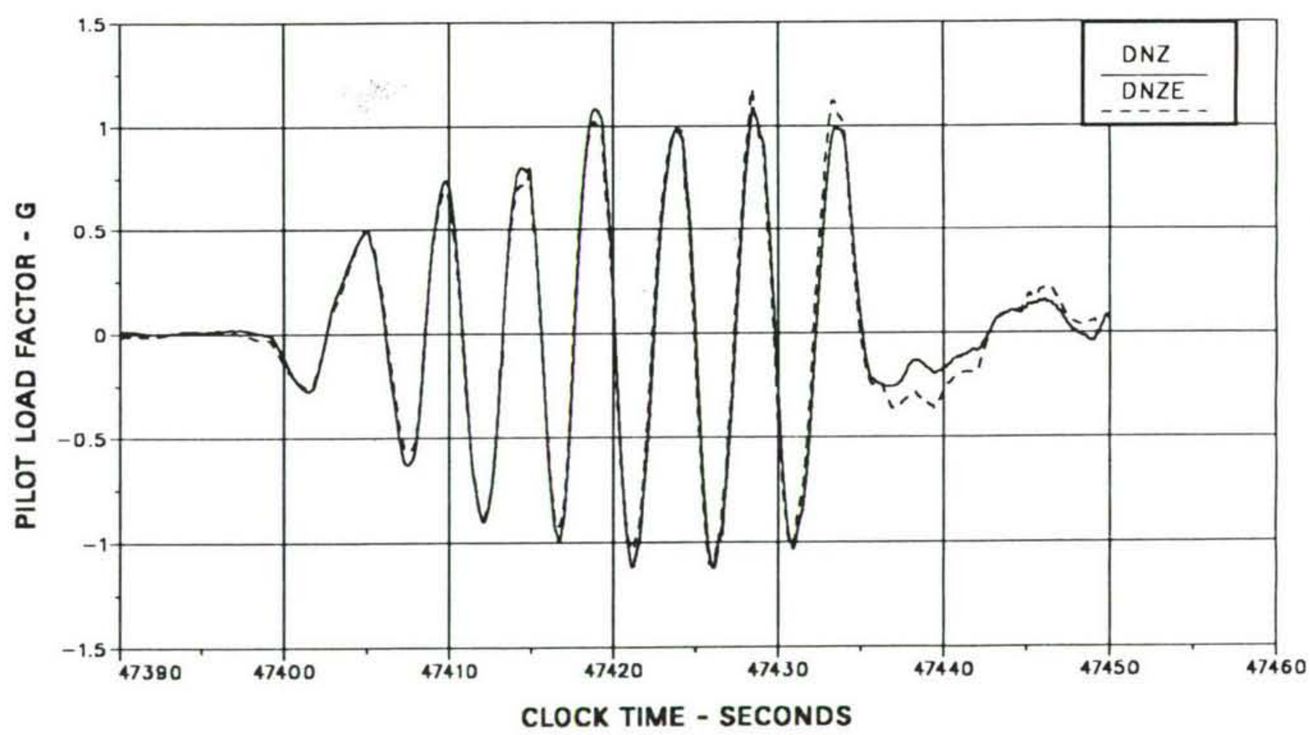


CALIBRATION OF THEORETICAL ELEVATOR EFFECTIVENESS

These results are from a roller coaster maneuver performed in still air. The figure shows a comparison between measured and computed load factor due to elevator deflection when a calibration coefficient derived by a least squares fit of the corresponding time histories is applied to the computed load factor. The correlation coefficient between the time histories is 0.99, which indicates that a simple correction in magnitude is justified.

A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

CALIBRATION OF THEORETICAL ELEVATOR EFFECTIVENESS
C-141B SMLRP FLIGHT 1 (STILL AIR MANEUVER)
COEFFICIENTS: CORRELATION=0.99, CALIBRATION=1.267



DEMONSTRATION OF SEPARATION PROCEDURE

The next three figures illustrate application of the separation procedure to extract the contributions of gust and maneuver from the total load factor. The first figure shows the amplitude and phase of a frequency response function calculated from a theoretical dynamic response model based upon the prevailing flight parameters. It will be applied as a filter to convert the incremental portion of the recorded elevator deflection to the resulting estimated load factor.

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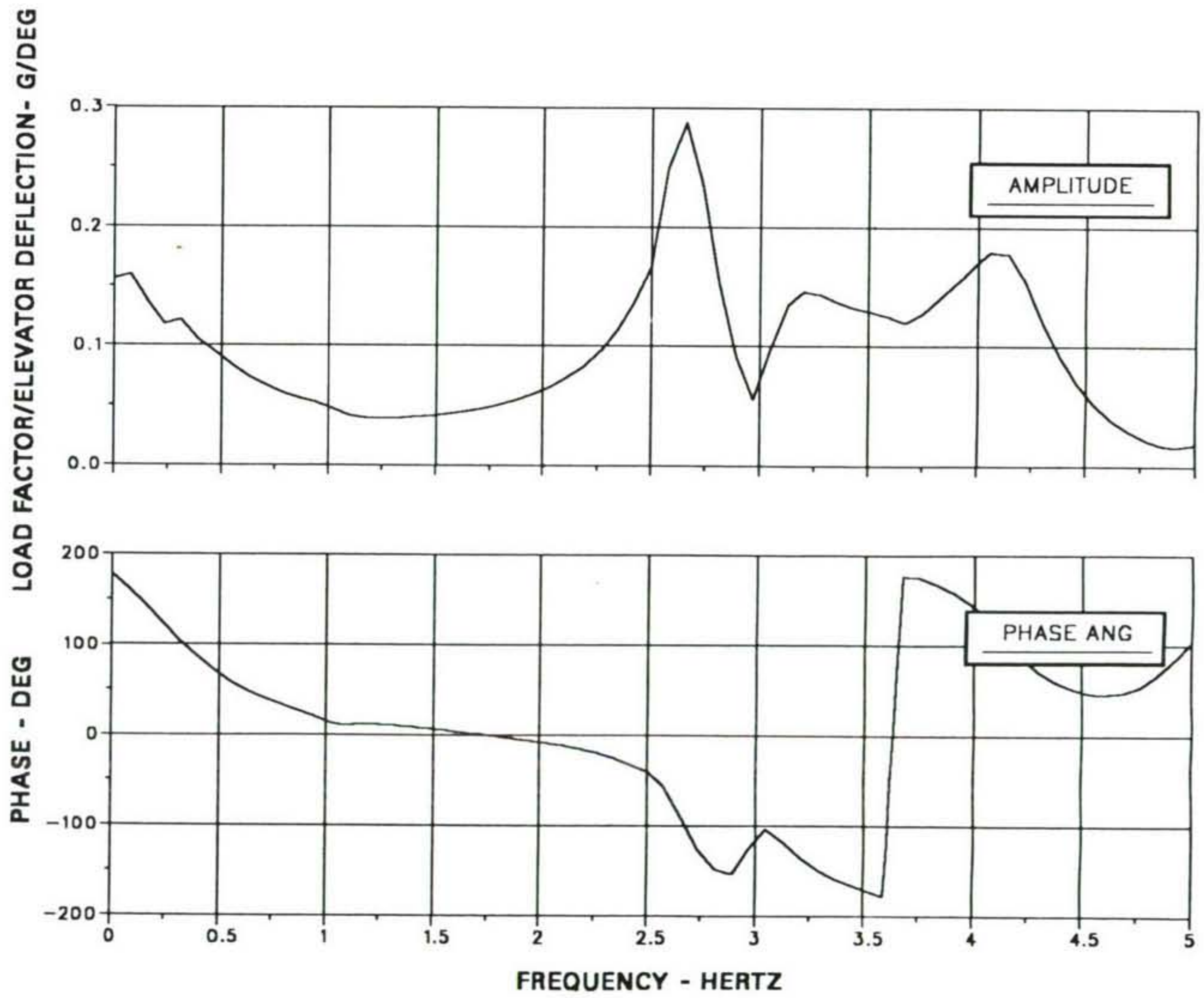
A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

DEMONSTRATION OF SEPARATION PROCEDURE
C-141B SMLRP FLIGHT 100 (LOW LEVEL)
FREQUENCY RESPONSE FUNCTION



DEMONSTRATON OF SEPARATION PROCEDURE

The upper plot shows a time history segment of recorded elevator deflection from the digitized flight tape. Its mean has been removed by high pass filtering to extract the incremental portion. The lower plot represents the corresponding predicted pilot load factor, and is obtained by filtering the incremental elevator deflection with the frequency response function shown previously.

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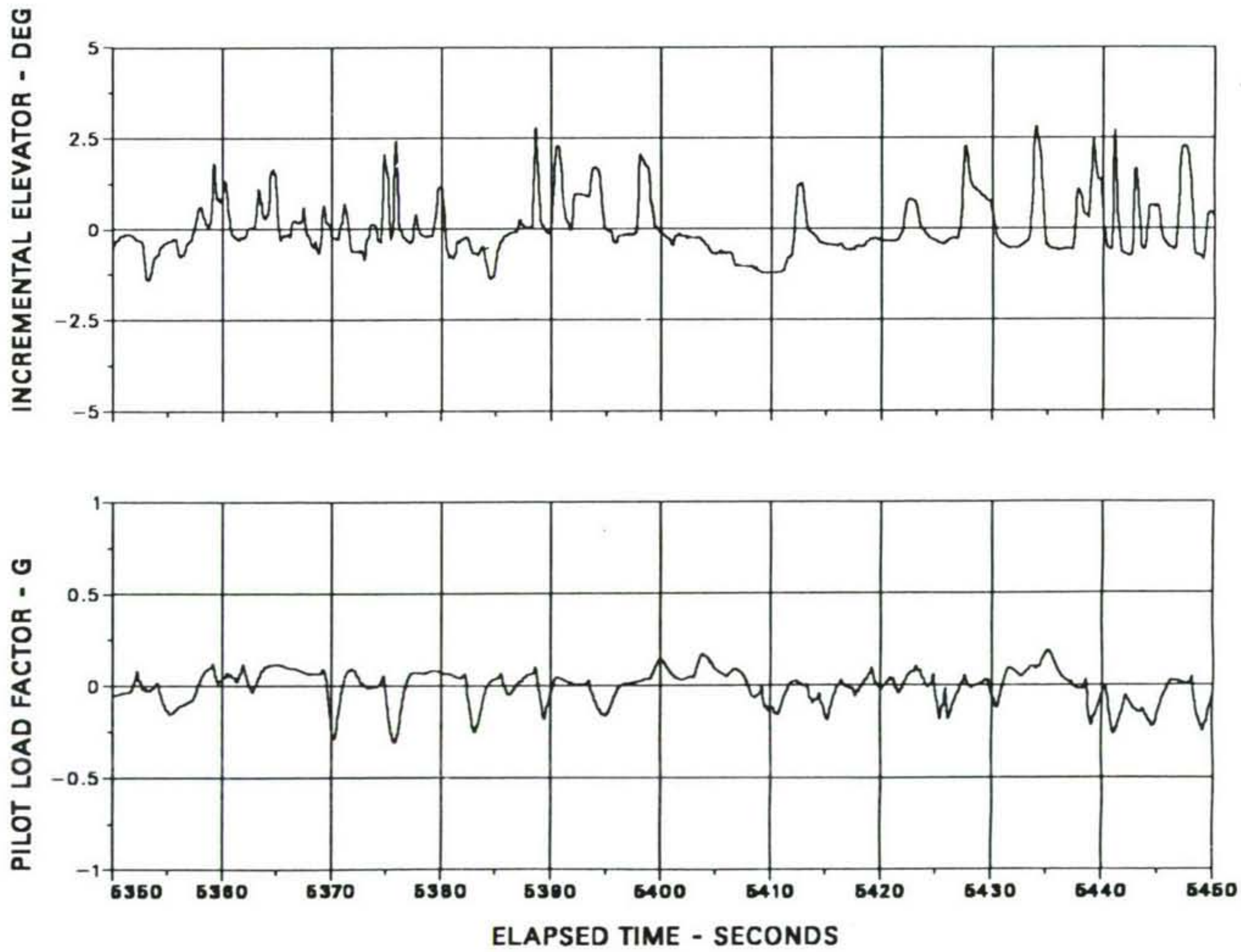
A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

DEMONSTRATION OF SEPARATION PROCEDURE
C-141B SMLRP FLIGHT 100 (LOW LEVEL)
TIME HISTORIES

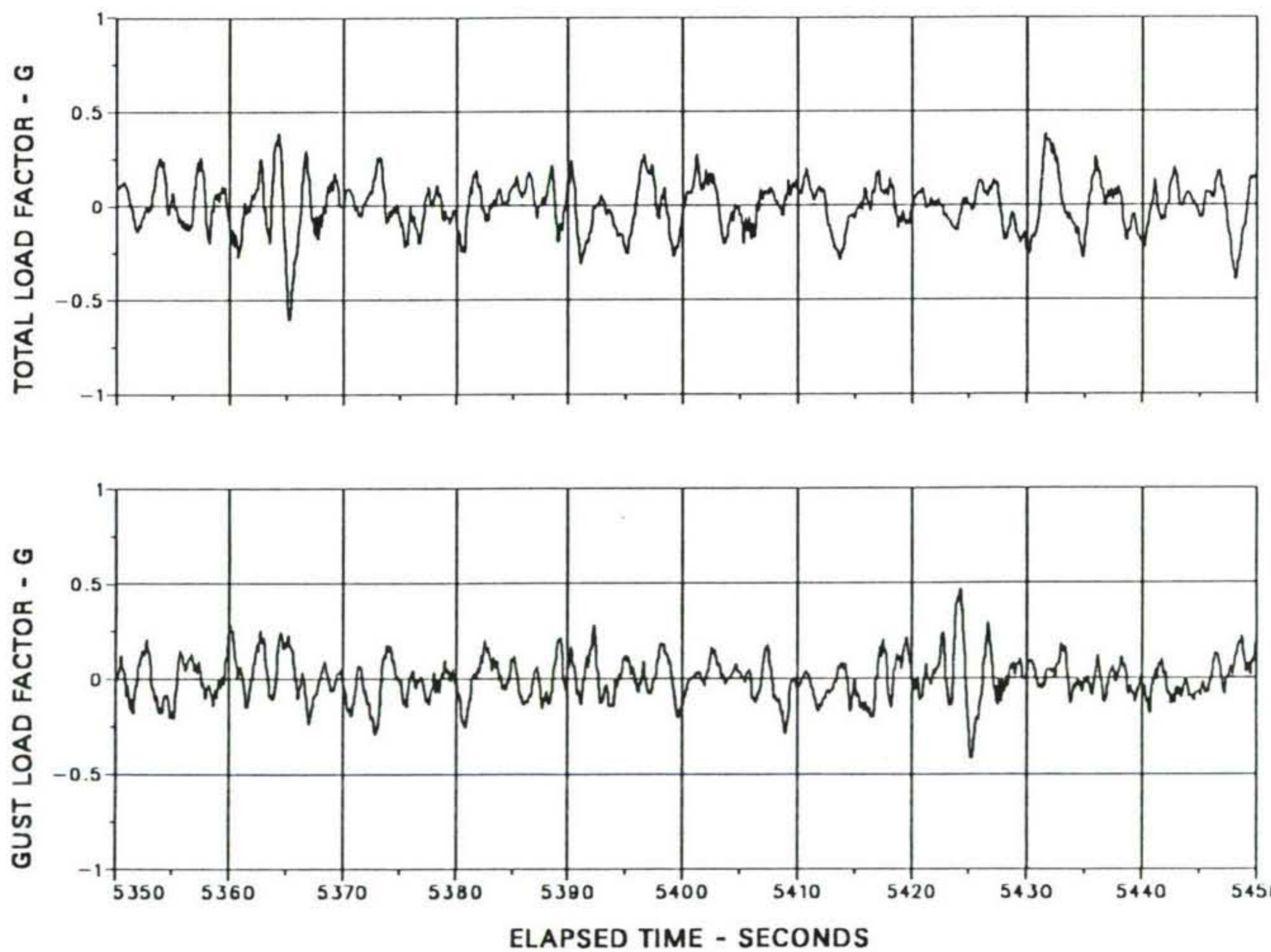


DEMONSTRATION OF SEPARATION PROCEDURE

The upper plot in this figure shows the recorded load factor from the digitized flight tape. Finally, the lower plot represents the load factor due to gust, and is obtained by subtracting the pilot load factor from the total load factor.

A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

DEMONSTRATION OF SEPARATION PROCEDURE
C-141B SMLRP FLIGHT 100 (LOW LEVEL)
TIME HISTORIES



CONCLUSIONS

A procedure for separating gust and maneuver components has been developed. The procedure includes flight calibation of the theoretical model, validation of the theoretical model for flight in gust, and a procedure for modeling the pilot response to gust induced motion. The separated gust and maneuver responses are available for establishing criteria for combining loads resulting from the two sources, gust and maneuver.

A PROCEDURE FOR SEPARATION OF GUST AND MANEUVER LOADS

CONCLUSIONS

- o A PROCEDURE FOR SEPARATING GUST AND MANEUVER IS DESCRIBED
- o THIS NEW PROCEDURE INCLUDES THE ENHANCEMENTS
  - -- FLIGHT CALIBRATION
  - -- GUST VALIDATION
  - -- PILOT MODELING
- o SEPARATED GUST AND MANEUVER RESPONSES ARE AVAILABLE FOR ESTABLISHMENT OF CRITERIA FOR LOADS COMBINATIONS

Correlation of the F-16 Block 30 Finite Element Models (FEM)

Kathleen D. Wilson, ASD/YPEF

Abstract

Finite element modelling methods are used extensively throughout the aerospace industry for the development and evaluation of aircraft design. A correlation study was recently completed for the F-16C/D Block 30 aircraft. The Block 30 FEM linear analysis resulted in several structural areas with negative margins of safety. Nonlinear FEM methods were employed, resulting in positive margins. A full-scale static test of the Block 30 F-16 was conducted for approximately thirty flight conditions. The resulting strain gage data was then compared with the FEM analysis results, showing good correlation. This paper will address the FEM (linear and nonlinear), the placement of gages to obtain the best data for correlation, results of the correlation study, and lessons learned.

Introduction

The evolution of the F-16 aircraft from full scale development version (Block 1 - 5) to Block 30 version involved very few structural changes. Aside from the change to the Common Engine Bay (CEB), what did occur was a significant increase in the Basic Flight Design Gross Weight (BFDGW) of the aircraft, approximately 20% increase. The structural 'goals' for the F-16 remained as a 9.0 g/8000 hour capability. The ability of the finite element models to predict the overall static strength of the F-16 was questioned. As part of the full scale static test to be conducted on the F-16 Block 30 aircraft, a correlation study would be accomplished to verify the math models and the stress analysis.

The correlation study provides a comparison of the stresses/strains predicted by the finite element models at ultimate loads to those measured during the static test. Eight major areas of the F-16 were used in this comparison:

1. Wing
2. Vertical Tail
3. Cockpit
4. Forward Fuselage
5. Center Fuselage
6. Aft Fuselage
7. Engine Access Cover
8. Ventral Fin

The horizontal tail was not included in this study as it was not a part of the static test.

Linear and Nonlinear FEM Analysis

The math models used in the finite element analysis of the F-16 Block 30 involved two types. Linear and nonlinear techniques were utilized. The global finite element model of the total F-16 Block 30 aircraft, a derivation of the F-16 Block 25 model, was developed using MSC/NASTRAN (see Figure 1). The model incorporated the common engine bay (CEB) as the only significant

F-16C Global NASTRAN Finite Element Model

Figure 1

structural change. This linear model is comprised of constant stress elements, shear elements, beams, bars and rods, and some elastic elements for connecting points. A total of 21295 grid points and 57017 elements exist in the Block 30 linear model.

Approximately fifty flight and ground conditions were applied to the model. These included both the General Electric (F110-GE-100) and the Pratt and Whitney (F100-PW-220) engine loadings. The global model contained several areas of which results showed high strains. Through the stress analysis of those high strain areas, negative margins of safety were shown to exist in the wing bolts, wing root rib, the leading edge flap, and the fuselage station 357 bulkhead. These areas were then modelled using the finer grid, ABAQUS nonlinear techniques.

The fine grid models were loaded with the internal loads and/or displacements which resulted from the global model. More accurate and detailed information about these areas was obtained through the use of the fine grid, nonlinear models. The stress analysis results for these areas showed positive margins of safety when based on the internal loads from the fine grid models. A correlation of the test to predicted strains was important in that it would validate the models, and show where further improvements need to be accomplished.

In general, conservatism is built into the models. Nominal drawing/material thicknesses, contours, and assemblies are used in the FEM. This approach is considered slightly conservative since General Dynamics tries to manufacture parts with dimensional tolerances on the plus side to reduce the incidences of scrappage. Manufacturing tolerances are usually -0.01/0.01 and -0.01/0.03. Dimensional and contour tolerances can result in strain values which differ greatly from the predicted values. Depending on part thickness and tolerance band, resultant differences in nominal membrane strain can vary as much as 11.1% to -10.7%, and 23.5% to -20.3% for nominal bending strain. (See Table 1).

Table 1

Predicted Strain Variations Resulting from Manufacturing Tolerances

Nominal Part Thickness	Tolerance	% From Nominal Membrane Strain	% From Nominal Bending Strain
0.10	0.01	-9.09	-17.35
0.10	-0.01	11.11	23.45
0.25	0.01	-3.84	-7.54
0.25	-0.01	4.16	8.50
0.25	0.03	-10.71	-20.28
0.50	0.01	-1.96	-3.88
0.50	-0.01	2.04	4.12
0.50	0.03	-5.66	-11.00

In the case of adding the effective skin area to elements representing flanges, the effective area of the skin was down-sized for conservatism to allow the predicted strains to be greater than the measured strains. Web areas which contain holes were modelled using equivalent thicknesses. Partical sizing of the finite elements would under predict the strains except in very simple stress fields. Figure 2 depicts these two concepts. Joint flexibility was not simulated except for the major splices of wing to fuselage attachment bolts, engine bay doors, vertical tail, and doors near the canopy sill longeron. Removing the stiffness from the door attachment joints allows the load path to remain in the surrounding structure rather than going through nonstructural doors. The canopy was not included in the finite element model.

Loads were applied to the global model to match the distribution of loads on the Static Test Article as closely as possible. This was done by way of a General Dynamics developed program which interfaced both the model and a set of external loads. For the finite element models, a combination of point loads and panel pressure loads were applied. On the static test article, loads were applied through pressure and tension pads normal to the loaded surface.

Strain Gage Placement

The majority of gages used in the Block 30 full scale static test were located for purposes other than the correlation study. Optimum gage location in regards to data necessary for the correlation study was not always possible. Desirable gage locations coincide with the "Stress Recovery Point" of the model elements as shown in Figure 3. When this is not possible, averaging of gage data would have to be done to support the correlation study.

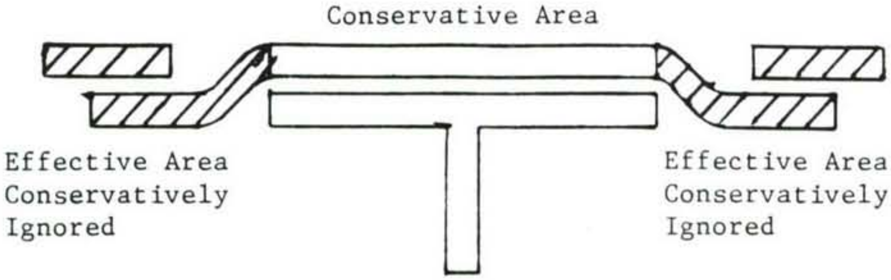
The types of gages used in the static test were axial and shear gages with rosette gages and additional gages as backup. Axial and shear gages provide overall coverage of the strains occurring on the static test article, but provide little information where significant bending may occur in the test article. Rosette gages will provide bending information. Unfortunately the cost of full coverage by rosette gages is prohibitive, and the accessibility of some areas on the aircraft would not allow the installation and function of this type of gage. Back up gages are used wherever possible or necessary. This may include using gages on both sides of a skin panel or in the same location on the opposite side of the aircraft. Due to a tolerance on the strain gage reading of +/- 2%, backup gages provide additional assurance of the internal strain values recorded during testing.

Correlation Approach

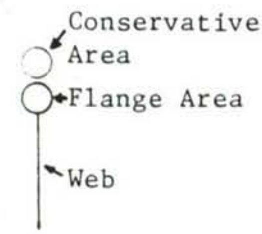
The ideal correlation would have gages placed at the same location on the test article as the stress recovery points of the finite element model. Since this did not occur, the gages used in the correlation study were selected or not selected for a number of reasons.

1. Gages located near stress concentrations like cutouts and fasteners were disregarded. Effects of large stress gradients from these are not reflected by the global model, but could possibly be useful if a fine grid model of the area had been accomplished. The majority of the gages selected were located away from stress concentration areas.

Effective Skin Sizing of Flange Areas

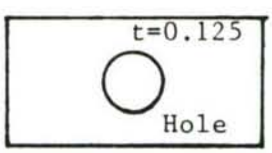


Real Part

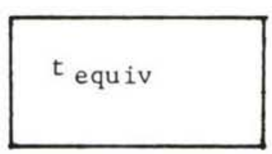


Finite Element Model

Web Areas Containing Holes



Real Part

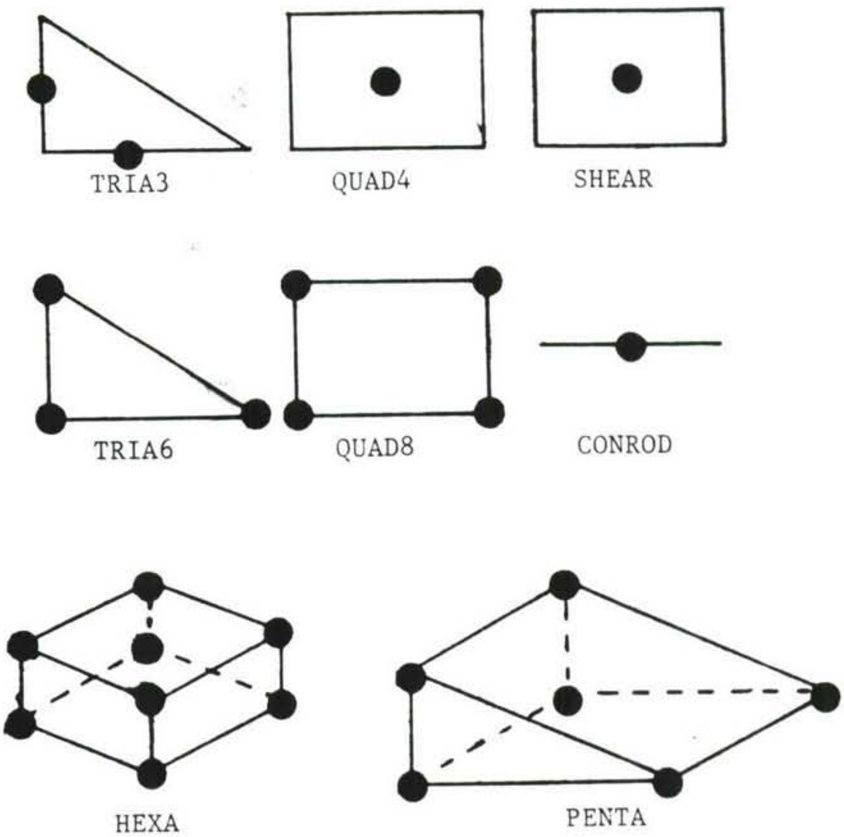


Finite Element

Equivalent Area Thickness is used

Conservative Element Sizing

Figure 2



● Indicate Stress Recovery Points of the NASTRAN Models (Preferred Gage Locations)

Preferred Gage Locations for Finite Elements

Figure 3

2. Gages with measured stress values less than one-third of the allowable stress were generally not used in the initial look at the models. These areas were considered to have stresses too insignificant to be of value to the initial effort. However, in the final correlation study, these gages were included.

3. Gages showing high strain gradients from stresses due to secondary displacements, instabilities and local bending due to hard points were not included unless the area had been modelled using ABAQUS. These effects are not represented in the global model, but would possibly show some correlation if the area had been modelled using fine grid nonlinear methods. The fine grid model of Fuselage Station 357.8 Bulkhead, Figure 4, shows some secondary displacement caused by bulkhead web buckling, an effect which is not included in the linear elastic analysis.

4. Where gages were not located at the stress recovery points of the models, averaging of gages would be necessary to obtain data for the correlation. In some cases, it was necessary to average gage data for a number of gages to correlate with a series of elements.

The occurrence of differences between the predicted strains and the measured strains was anticipated. These differences would be expressed as a percentage of the measured stress:

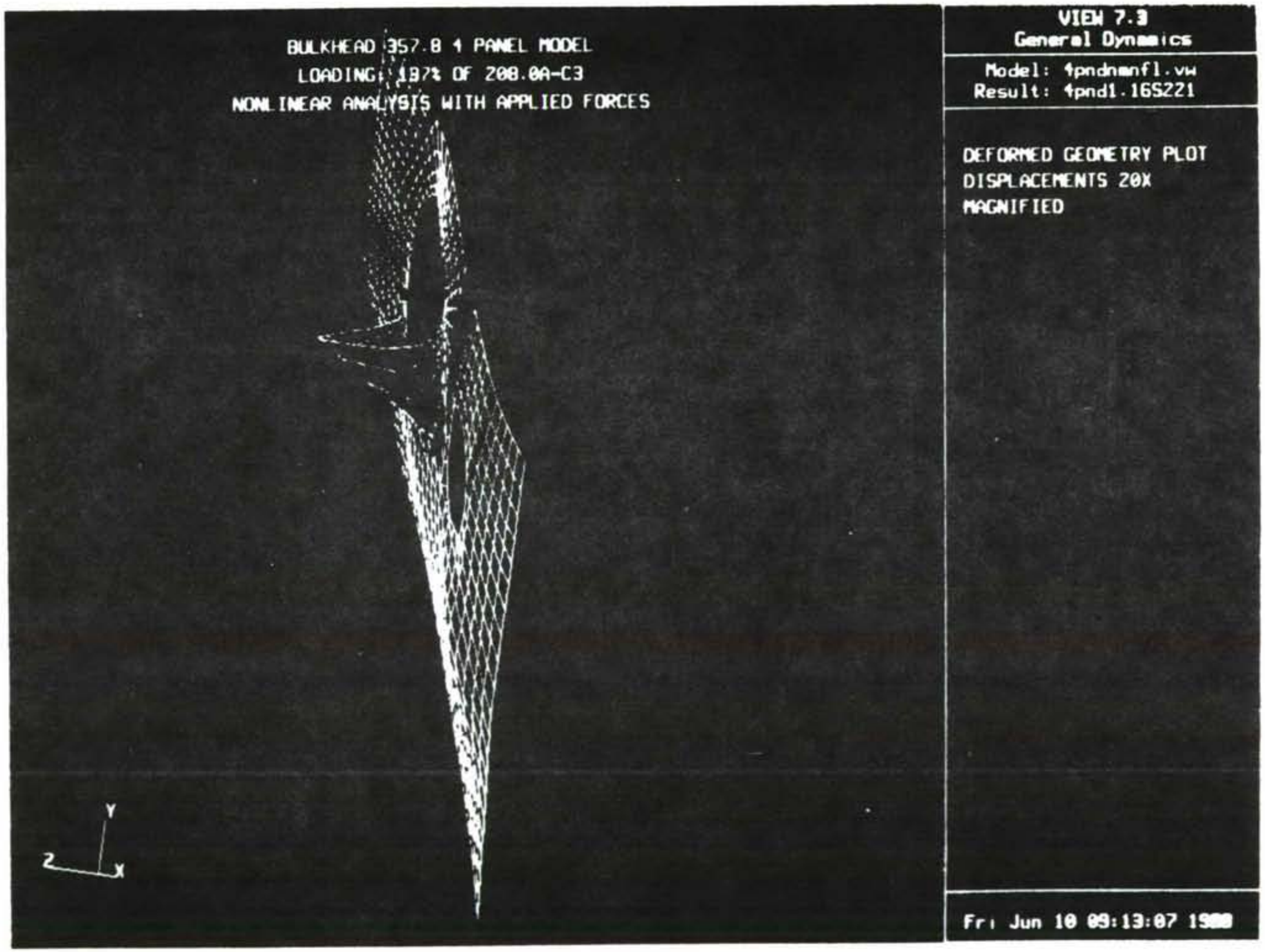
$$\text{Delta } \% = \frac{(\text{Predicted strain - Measured Strain})}{\text{Measured Stain}} * 100$$

This would then show conservatism as a positive percentage and unconservatism as a negative.

Correlation Results

The six ultimate load conditions used in the correlation study are shown in Table 2. Represented are load cases with maximum positive and negative fuselage and wing bending. Conditions include both the F100 and F110 engines, and altitude, Mach, and stores variations.

Overall, the correlation study shows conservatism in the finite element models. A total of 268 elements were included in the comparison with the static test results. The Delta percent between the predicted strains and the measured strains ranged from -50 to 2056.58 (%). From statistical analysis, the 268 elements yield a mean Delta percent of 18.71, with a standard deviation of 129.75, and a median Delta percent of 3.595. A few of the resulting Delta percentages can be considered as out-liers as listed in Table 3. Negligible and trivial strains can produce large Delta percentages which skew the overall results. If the out-liers are removed from the statistical analysis, the remaining 257 elements yield a mean Delta percent of 6.18 with a standard deviation of 14.76, and a median Delta percent of 3.30. The histogram in Figure 5 shows the distribution of the Delta percentages for the 257 remaining elements. If this were a normal curve, 68% of the correlated elements lie between -8.58 and 20.94 Delta percent. The distribution does show that 66.54% of the elements reflect as conservative (Delta percent of zero or greater). It should be noted that of the 33.46% of the correlated elements which fall into the unconservative range, 22.18% are between -10 and 0 Delta percent and 10.51% between -20 and -10 Delta percent.



<u>Fine Grid Model of F.S. 357.8 Bulkhead</u>

Figure 4

Table 2

Conditions Included in Correlation Study

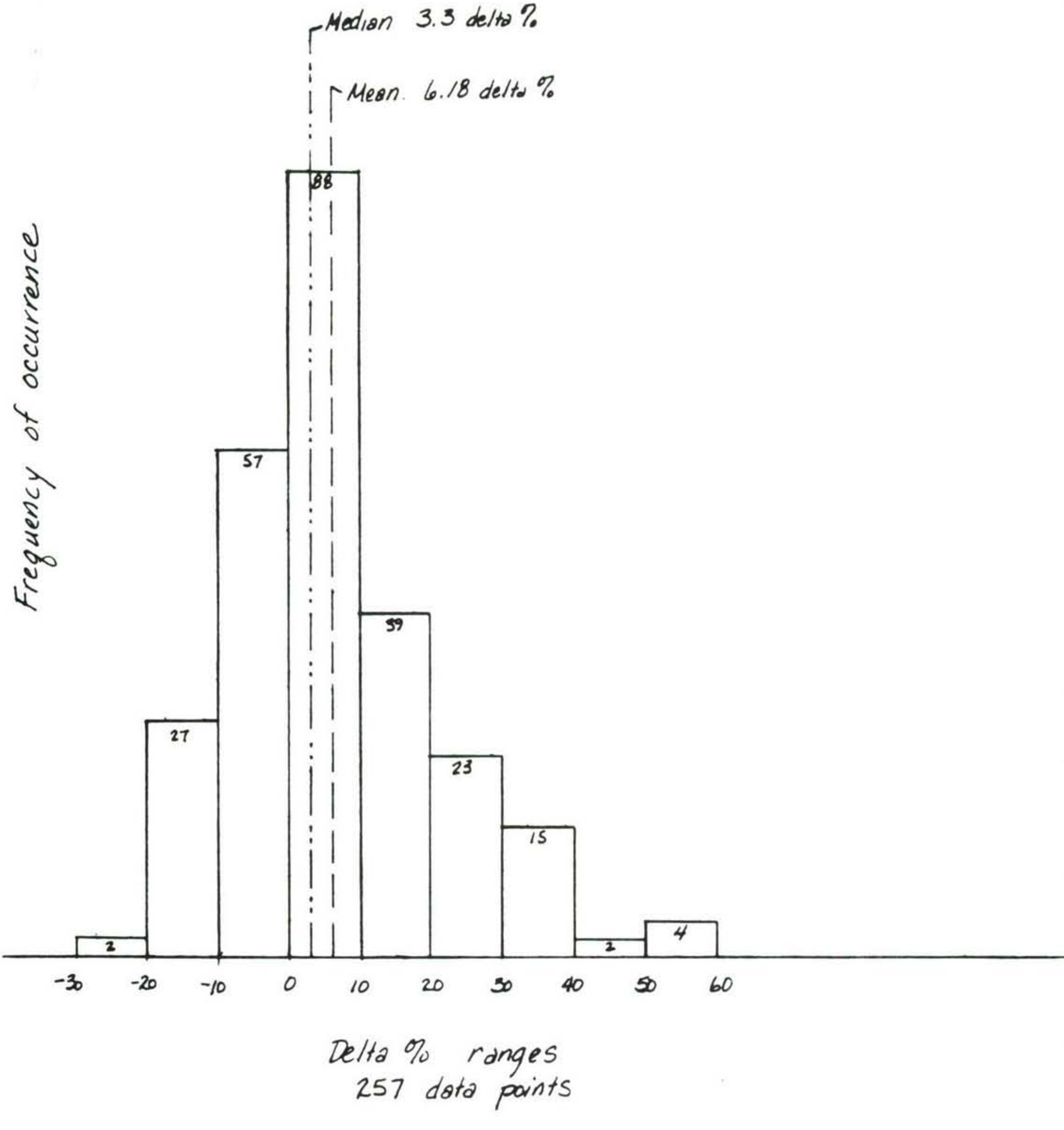
Condition	Flight Maneuver *	Gross Weight *	Critical Structure
STAT-PW-208.0A	9.0 BSPU .95 SL	24979	Wing ETA 0 Aft Fuselage
STAT-PW-307.3	5.86 RPO 1.80 24K	24979	Aft Fuselage
STAT-GE-258.001A	9.0 BSPU 1.20 20K	26708	Leading Edge Flap
STAT-GE-201.0	-3.00 BSPO 1.10 SL	25789	Forward Fuselage
STAT-PW-296.2A	5.86 Roll 1.05 SL	24979	Wing ETA 0, 0.3 Wing Flaperon
STAT-PW-208.0A-C3	9.0 BSPU .95 SL	24979	Aft Fuselage

* NOTE: 1. Maneuvers are given as G loading, Maneuver, Mach and Altitude
2. BSPU = Balanced Symmetric Pull Up
RPO = Rolling Pull Out
BSPO = Balanced Symmetric Push Over
SL = Sea Level
3. Gross Weight is given in pounds

Table 3

Delta Percent Removed as Out-liers

Model	Delta %	Reason for Removal
Cockpit Fwd Fuselage	-50.0	Door effectiveness too high
Vertical Tail Stabilizer	-33.98	Measured strain was trivial
Vertical Tail Stabilizer	62.29	Web Bending, gage not backed-up
Aft Fuselage	72.29	Engine Access Door Bending
Aft Fuselage	85.30	Conservative Flange Sizing
Bulkhead 309.8	91.63	Conservative Sizing of Effective Skin
Vertical Tail Stabilizer	105.20	Measured strain was trivial
Vertical Tail Stabilizer	157.26	Measured strain was trivial
Cockpit Fwd Fuselage	267.0	Measured strain was trivial
Cockpit Fwd Fuselage	370.0	Measured strain was trivial
Vertical Tail Stabilizer	2056.58	Measured strain was trivial



Distribution of Delta Percents

Figure 5

It is necessary to keep in mind that the correlation results are affected by many factors. As discussed earlier, thickness and contour tolerances, eccentricities, conservative sizing of model elements, strain gage type and placement, the type of finite element model utilized, all contribute to the differences between the predicted and measured strains. In addition, undetected cumulative damage and residual displacements from the thirty-two ultimate load applications on the static test article as well as the test load application methods contribute to the differences which result. Although these contributors need to be kept in mind when modelling the aircraft, not every situation can be covered. The global NASTRAN model is not capable of reflecting the nonlinearities of the structure, and although fine grid models have the capability of analyzing some nonlinearities, it is impossible to model everything without either raising the cost, or increasing schedule time.

The F-16, Block 30 wing models were separated into lower wing skin, upper wing skin, wing spar and ribs, and leading edge flap. Both the upper and lower wing skins were modelled with ABAQUS allowing the predicted results to include the skin bending stresses where wing spar discontinuities are caused by changes from full webs to trusses. The predicted skin strains compared favorably with the measured strains and in general showed conservative results. Skin sizing on the upper wing skin contributed to the conservatism. The wing spar, ribs and leading edge flap predicted strains were obtained from the NASTRAN global model. The results showed some conservatism, however fine grid modelling would have improved the results of these two models. Some gages on the wing skins, ribs, spars and leading edge flap were not included in the correlation analysis due to proximity of cutouts and fastener patterns. Also, gages reflecting negligible strains were not included.

The vertical tail predicted strains were obtained from the global model. Correlation results were considered as reasonable. One sizable under prediction of strain occurred in an area on the right hand skin which, in addition to membrane stresses, was experiencing the development of local buckling. A fine grid model would have contributed to the prediction of this condition. The areas around the fitting bolts and near cutouts were not included since a fine grid model would be required to correlate these areas. Some gages with negligible strains were not included.

The predicted strains for the forward fuselage and cockpit combined models were derived from the global NASTRAN finite element model. The comparison of the predicted to measured strains was accomplished using a limit condition due to the nonlinearity of the measured data for ultimate loads around the inlet casting. A fine grid ABAQUS model would have allowed a comparison of the area of nonlinearity. Correlation results were reasonable with the exception of the fuel tank which had trivial strains and the cockpit lower longeron where the seat rail fittings as modelled prevented the longeron from straining. To obtain better correlation in the longeron, the fittings effectiveness would need to be reduced.

The center fuselage predicted strains were derived from critical conditions in the global NASTRAN model. These strains were ratioed to allow comparison to one of the tested conditions. Due to the conservative sizing of the bulkhead skin and flange elements, the comparison showed favorable results. A few gages near fastener patterns would have shown better correlation if fine grid models had been used, however, fine grid models were not required to determine positive margins of safety for static strength.

The global NASTRAN predicted strains were used in the comparison with the aft fuselage and engine access door measured strains. Gages in the areas free of stress concentrations generally showed good correlation; although a few areas showed some secondary bending and buckling effects, such as through the engine access doors. Fine grid models would have aided in determining the effects of secondary bending.

The ventral fins were not originally planned for inclusion in the correlation study, therefore not many gages were planned for installation on the ventral fins. Due to a history of problems, the comparison of the strains was included, especially in the area of the forward attach bolts. Reasonable correlation was obtained.

The F-16 Block 30 stress analysis relies on the results of the finite element models for the internal loads for each flight and ground condition. The stress analysis of those areas which from the correlation resulted in Delta percentages below zero, was reviewed. Although margins of safety for these areas would be lower than the stress analysis originally showed, this did not result in additional areas with marginal or negative margins of safety.

Lessons Learned

Fine grid, nonlinear models produce more accurate detailed information in areas where there are local effects or nonlinear behavior. As indicated, several sections of the Block 30 Aircraft could have used fine grid models to better understand the behavior of the structure for the stress analysis and would have provided better correlation between the predicted and measured data. For Block 40, a more complex global model has been developed which increased the number of grid points and elements by 39% and 20%, respectively. In addition, higher order elements have been added such as TRIA6 and QUAD8 allowing linear strain rather than constant strain at the sides. A correlation study will be conducted on Block 40 using the results of strain gage data from the flight loads survey during the flight test program.

From the results of the correlation study, it was reemphasized that the finite element modelling process is an iterative one. Adjustments can be made to the models as the aircraft is developed.

Fine grid models are being developed in the Block 40 efforts for many of the areas that were identified under the Block 30 correlation study. In particular, attention would be given areas of eccentricities and stress concentrations.

Loading conditions applied to the finite element models should match the application of loads to the test article as closely as possible to aid in the correlation study.

Gage placement for correlation purposes should be as close to the finite element model stress recovery points as possible. Maximize the usage of rosette gages where possible. Back-up gages are added assurance that strain gage anomalies are not causing incorrect results.

Finite element models are not infallible, nor can everything be modelled. Variations between the predicted strains and measured results will occur regardless of the care taken in modelling the structure. It is difficult to account for all the possible factors which influence the test results.

Include the canopy in the finite element model. Although joint flexibility would be required, the canopy would supply a small amount of stiffness to the cockpit area.

Summary

In using finite element models for design of a structure, conservatism in element sizing is key in assuring that the predicted results have not been under predicted for strength. This includes nominal material thicknesses, reduction of effective skin areas around flanges, and equivalent thicknesses for webs containing holes. More extensive use should be made of fine grid nonlinear modelling techniques especially to gain insight into the static strength of areas containing stress concentrations, eccentricities, fastener patterns, and hole patterns during the design and development phase of a new structure or modification to existing structure.

The placement of strain gages for the purposes of gaining insight into the modelling capabilities should be carefully planned to maximize the use of the finite element model stress recovery point locations on the test article. In addition, maximum use of rosette gages and backup gages to increase the confidence levels in the strains measured on the test article.

Negligible and trivial strains can skew the data and cause results which are not meaningful. A post-correlation check should be made of the stress analyses for those elements where negative Delta percentages resulted. This assures that the structure does not contain areas of negative margins of safety.

Finite element models are design and analysis tools that are not infallible. Do not rely solely on the finite element model. Testing is still essential, especially where newly designed aircraft or major modifications to an existing block of aircraft is involved, to verify the models.

References

General Dynamics Report 16PR5037, Vol I, II, and III, "F-16C Static Test Airplane Certification Plan", 1 Dec 88.

General Dynamics Report 16PR5020, Vol I, II, and III, "F-16C/D CEB Internal Loads Report", 30 Aug 89 (Change 1).

General Dynamics Report 16PR8633, "Block 30 Static Test Strain/Stress Correlation to Finite Element Models", 30 May 89 and 13 Sep 89 (Errata 1).

F-16C FULL-SCALE AIRFRAME DURABILITY TEST STATUS/RESULTS

MR CHARLES A. BABISH IV

ASD/YPEF

INTRODUCTION

- RATIONALE FOR TEST
- TEST PROGRAM
- TEST STATUS
- TEST RESULTS TO DATE

RATIONALE FOR TEST

- INCREASE IN BASIC FLIGHT DESIGN GROSS WEIGHT FROM 22500 LBS TO 26910 LBS

- INCREASE IN MAXIMUM TAKEOFF GROSS WEIGHT FROM 33000 LBS TO 37500 LBS

- CHANGE IN OPERATIONAL USAGE FROM

    55% A-A   TO   28% A-A
    20% A-G   57% A-G

- 8000 HOUR SERVICE LIFE BECAME A GOAL FOR THE F-16C/D AIRCRAFT, THUS A GREATER POTENTIAL FOR PROBLEMS EXIST

TEST PROGRAM

- TEST ARTICLE
  - STRUCTURALLY COMPLETE BLOCK 30 AIRFRAME
  - DUMMY LANDING GEARS, HORIZONTAL TAILS, AND ENGINE
- APPLIED LOADS
  - AIR, INERTIA, ENGINE THRUST, GROUND, STORE
  - COCKPIT AND FUEL TANKS PRESSURIZED
  - LEADING EDGE FLAP ACTUATED UNDER LOAD
  - TRAILING EDGE FLAPERON ACTUATED UNDER LOAD
  - RUDDER LOADED BUT NOT ACTUATED

TEST PROGRAM (CONT'D)

- TEST USAGE
  - TWO LIVES OF 8000 HOURS EACH
  - 5840 FLIGHTS PER LIFE
  - TEST SPECTRUM CONSISTS OF A 500 HOUR BLOCK OF RANDOMLY ORDERED FLIGHTS REPEATED AS NECESSARY
  - AVERAGE CYCLING RATE:

    30 LOAD PTS/MIN

    168 LOAD PTS/FLIGHT HOUR

    10 FLIGHT HOURS/CLOCK HOUR

TEST STATUS

- TEST WAS STOPPED AT 7330 HOURS DUE TO A PRESSURE LEAK IN THE CENTER FUSELAGE FUEL TANK (CENTER FUSELAGE BULKHEAD FAILURES)

- THE ONE-LIFE MAJOR INSPECTION WAS CONDUCTED AT THIS TIME

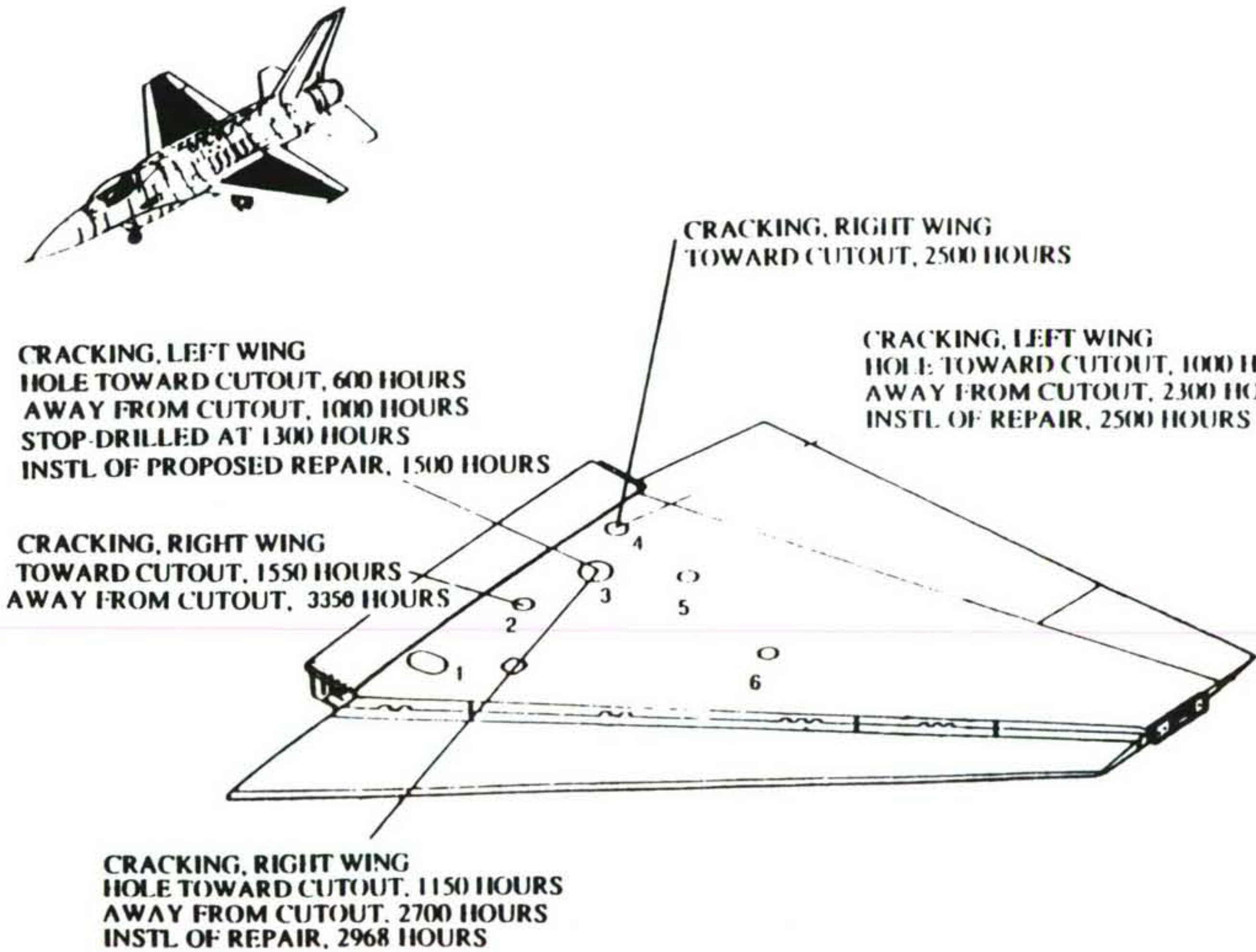
- ADDITIONAL CRACKED AREAS WERE FOUND

- REPAIR OPTIONS UNDER STUDY

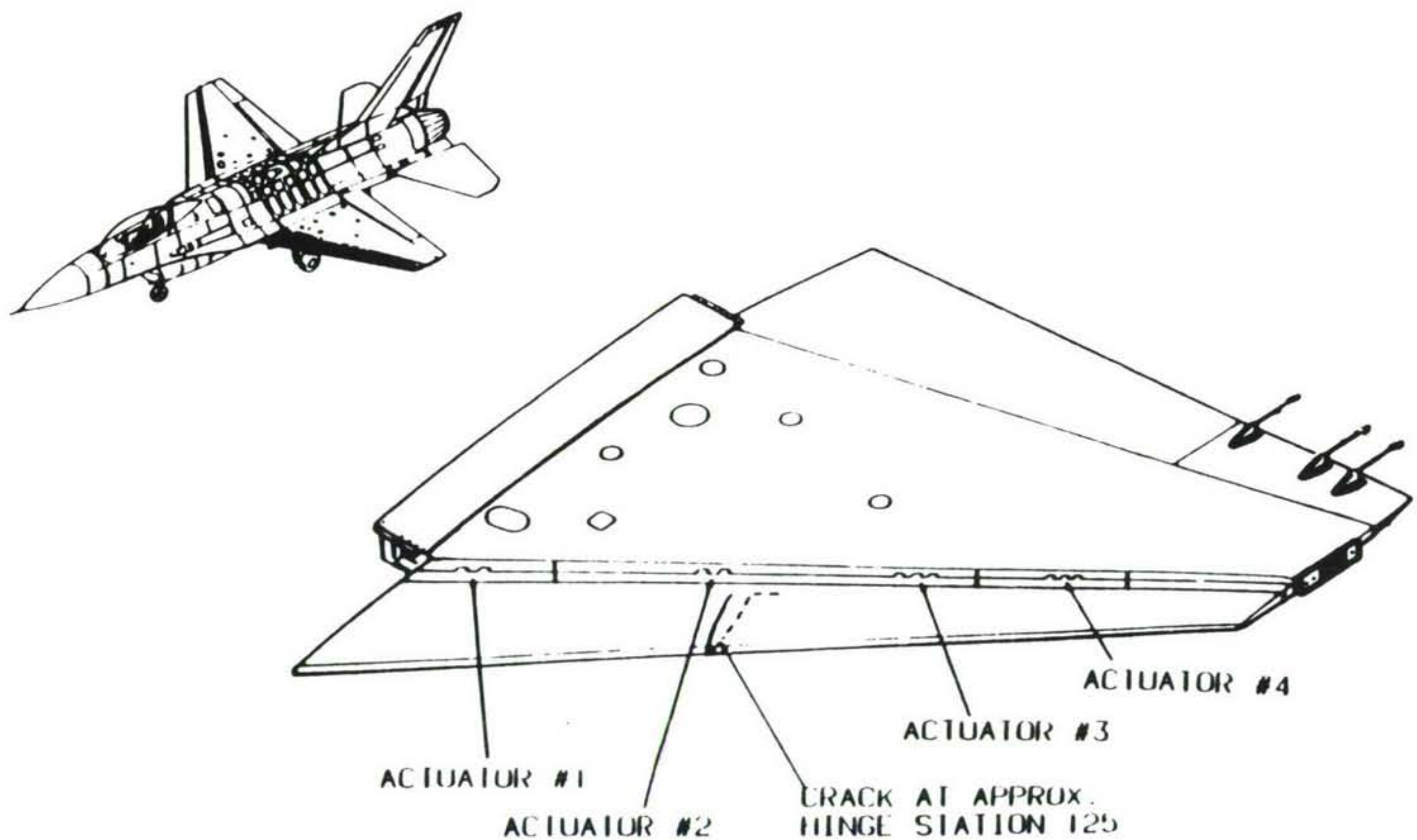
TEST RESULTS TO DATE

- CRACKS IN UPPER WING SKIN ACCESS HOLES (600 HOURS)
- LEADING EDGE FLAP FAILURES (704 HOURS)
- FORWARD FUSELAGE LONGERON/WEB FAILURES (2968 HOURS)
- CRACKS IN BULKHEAD AT F.S. 341.8 (3988 HOURS)
- CRACKS IN BULKHEAD AT F.S. 357.8 (3988 HOURS)
- CRACKS IN BULKHEAD AT F.S. 446.1 (3988 HOURS)
- CENTER FUSELAGE BULKHEAD CRACKS (7330 HOURS)

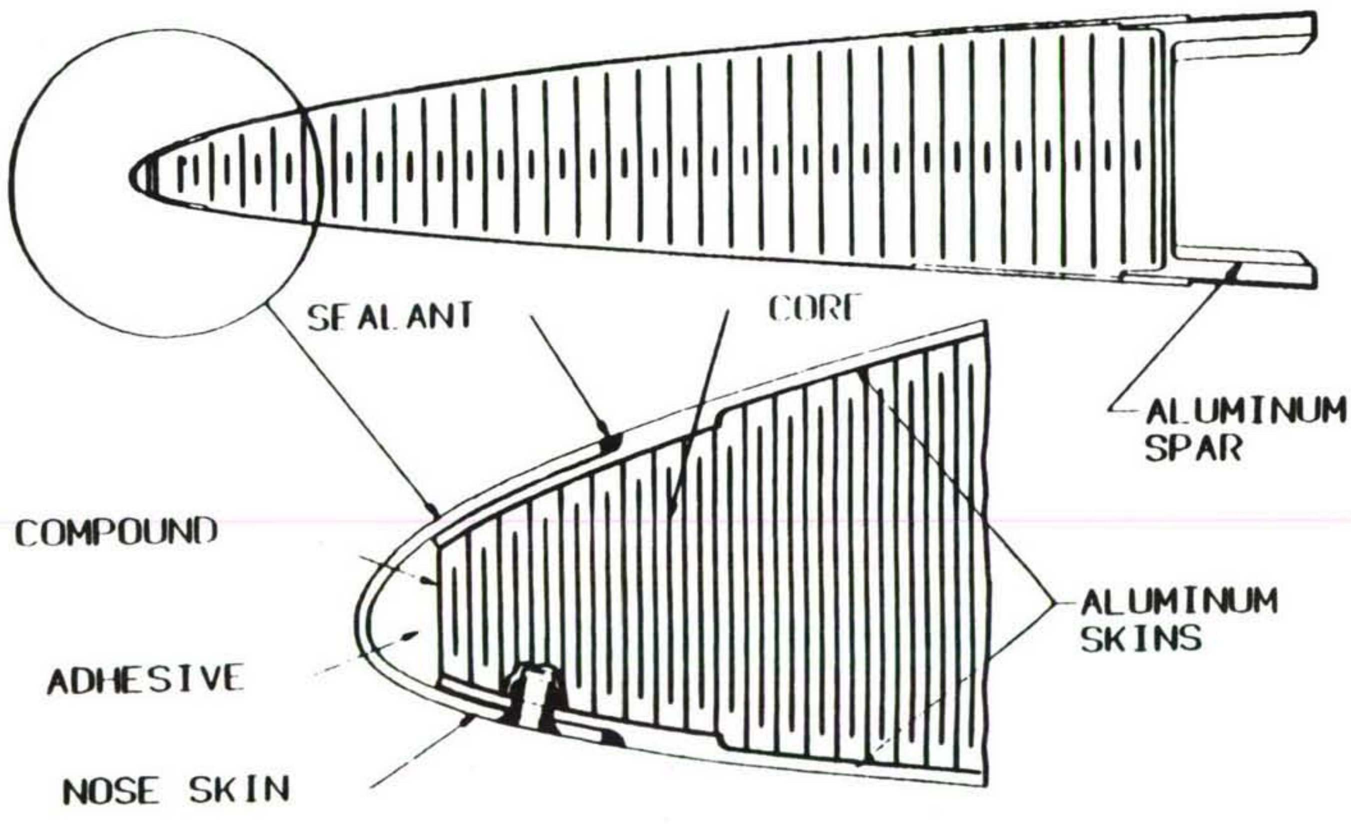
UPPER WING SKIN
SUMMARY OF CRACKING/REPAIRS



LEADING EDGE FLAP
FAILURE LOCATIONS



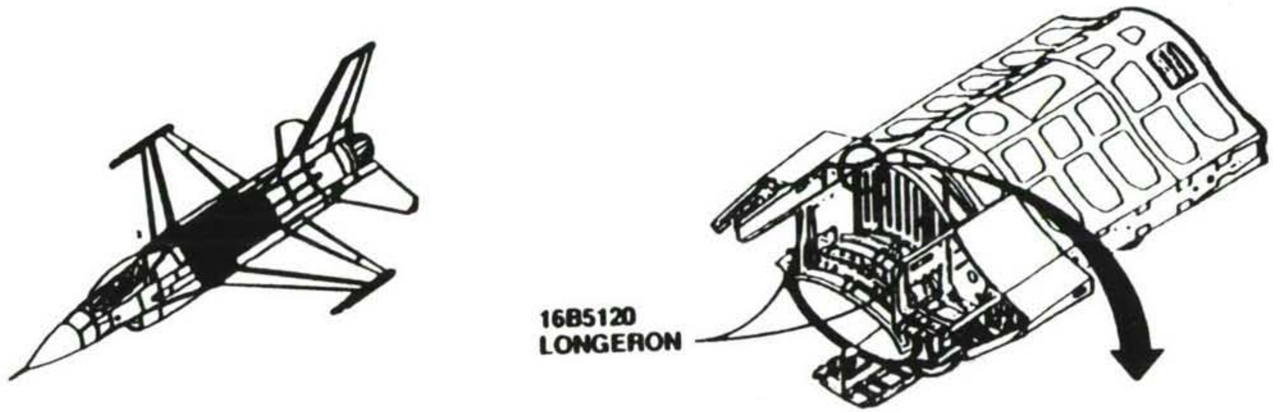
LEADING EDGE FLAP
GEOMETRY

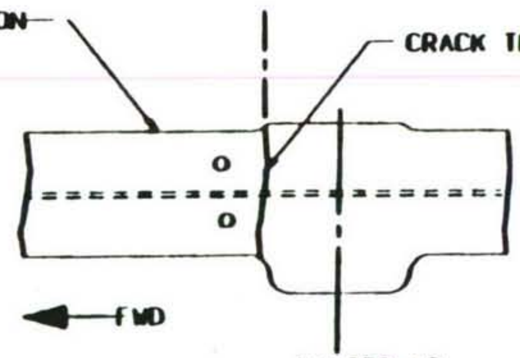


LEADING EDGE FLAP
FAILURE HISTORY

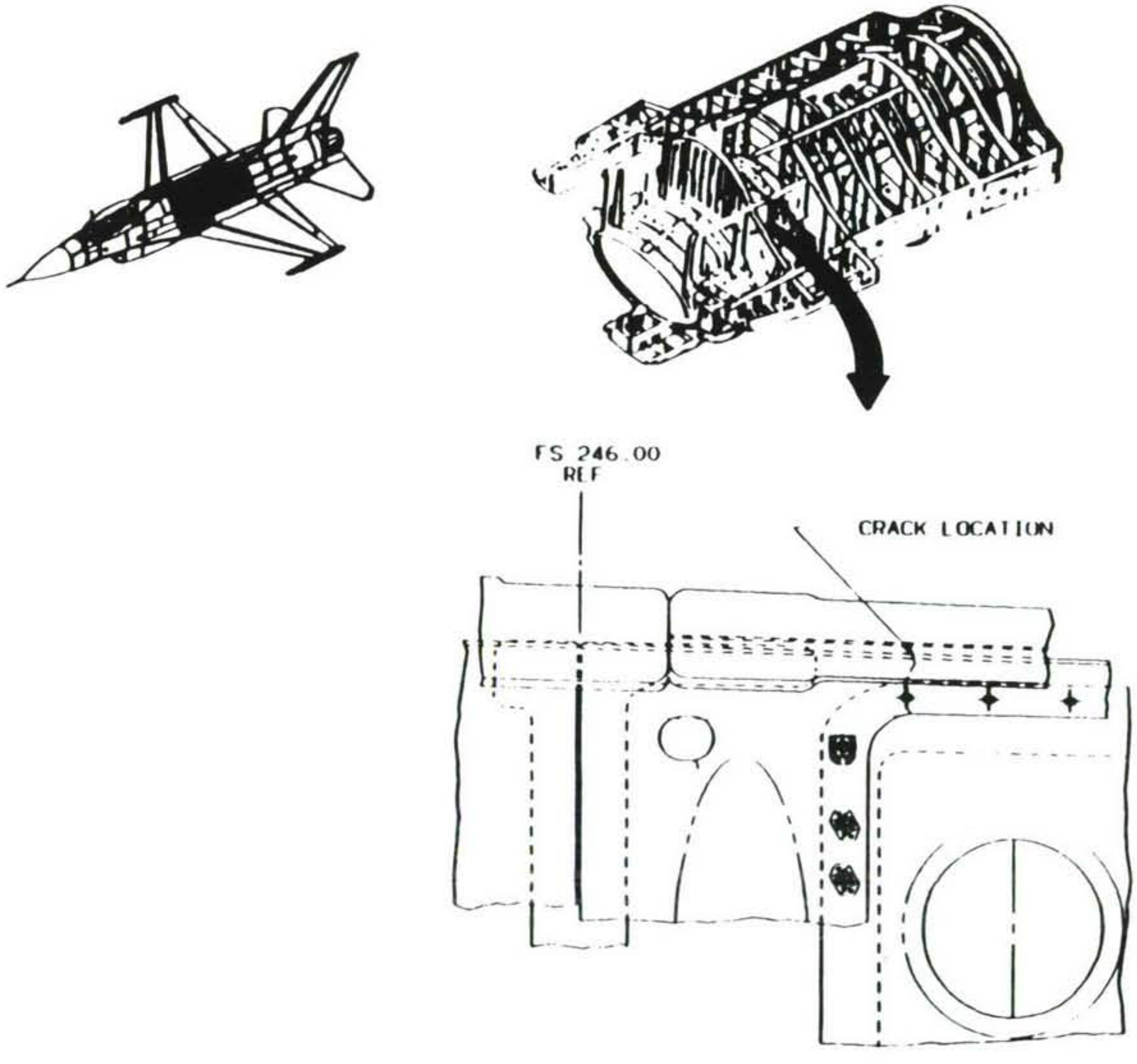
DATE	SIDE	FLT HRS	△FLT HRS	ACTION
OCT 87	LEFT	704	704	REPLACED
JAN 88	RIGHT	1200	1200	CRACK CUT-OUT
APR 88	RIGHT	1800	1800	CRACK CUT-OUT
APR 88	LEFT	1848	1144	BOTH REPLACED
JUN 88	LEFT	2968	1120	REPAIRED
JUL 88	LEFT	3024	1176	REPAIRED
AUG 88	LEFT	3988	2140	BOTH REPLACED

FORWARD LONGERON
FAILURE LOCATIONS





WEB
FAILURE LOCATIONS



FORWARD LONGERON/WEB
FAILURE HISTORY

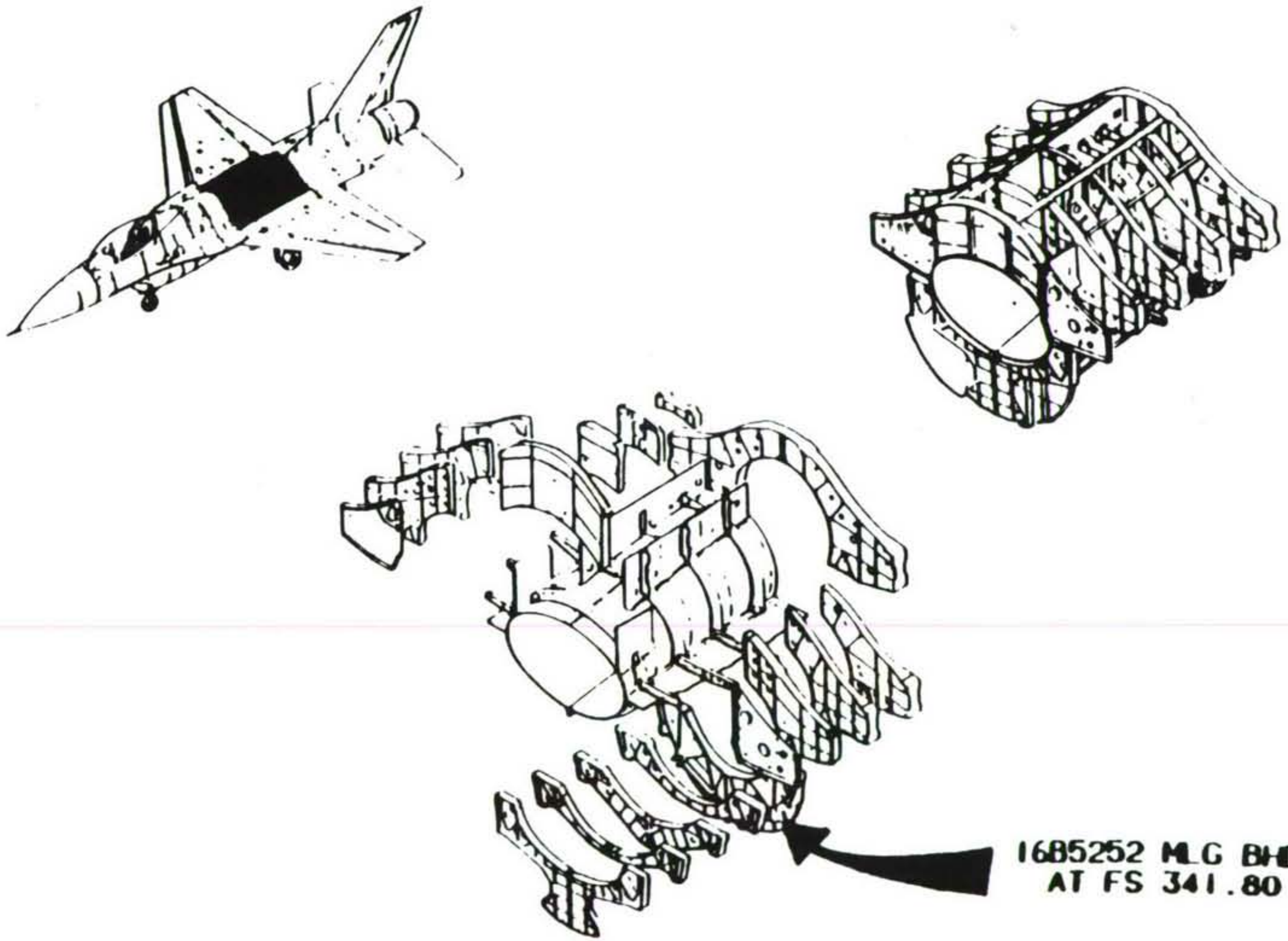
- R/H LONGERON FAILURE AT F.S. 278 AT 2968 HOURS (JUL 88)
  - REPLACED WITH REDESIGNED LONGERON
- L/H LONGERON FAILURE AT F.S. 267.4 AT 3988 HOURS (AUG 88)
  - REPLACED WITH REDESIGNED LONGERON (DIFFERENT FROM ABOVE)
- WEB CRACKS AT 3988 HOURS (AUG 88)
  - REPAIRED WITH DOUBLERS

FORWARD LONGERON/WEB
FAILURE HISTORY CONT'D

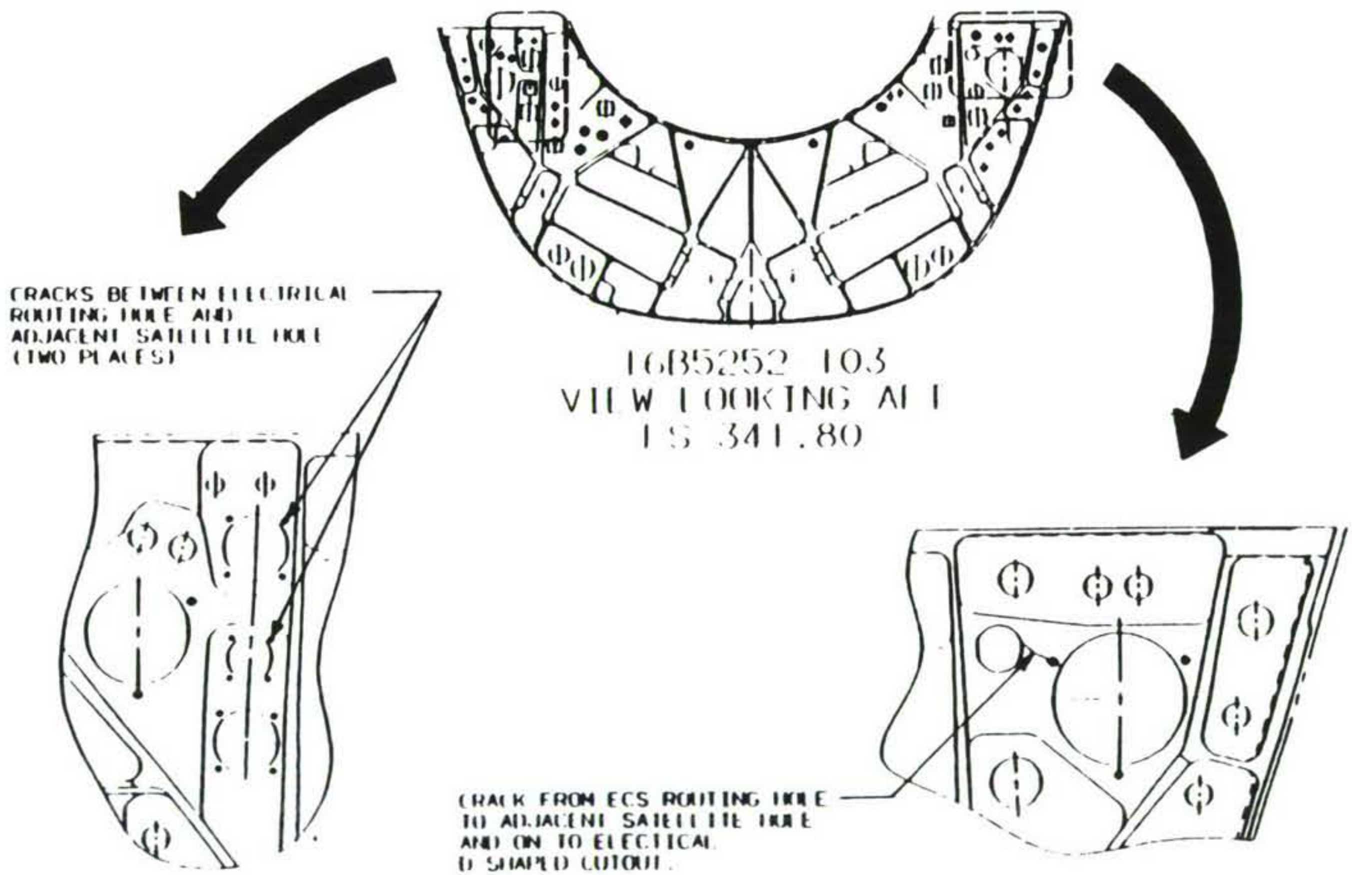
- R/H LONGERON FAILURE AT F.S. 278 AT 5000 HOURS ($\triangle$ = 2032 HOURS)
  - REPLACED WITH REDESIGNED LONGERON

- WEB CRACK AT 5000 HOURS
  - REPAIRED WITH DOUBLERS

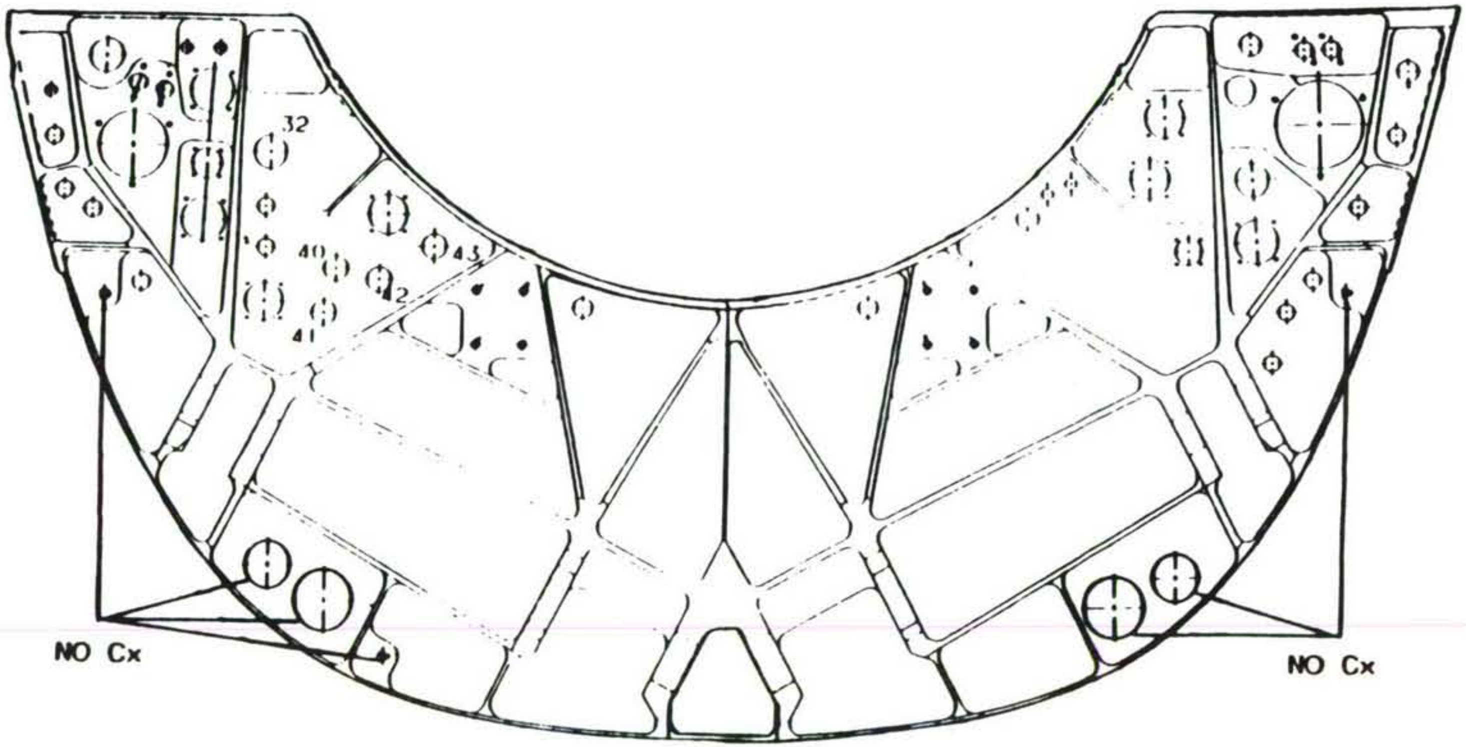
BULKHEAD AT F.S. 341.8
WING CARRY-THROUGH BULKHEAD



BULKHEAD AT F.S. 341.8
FAILURE LOCATIONS



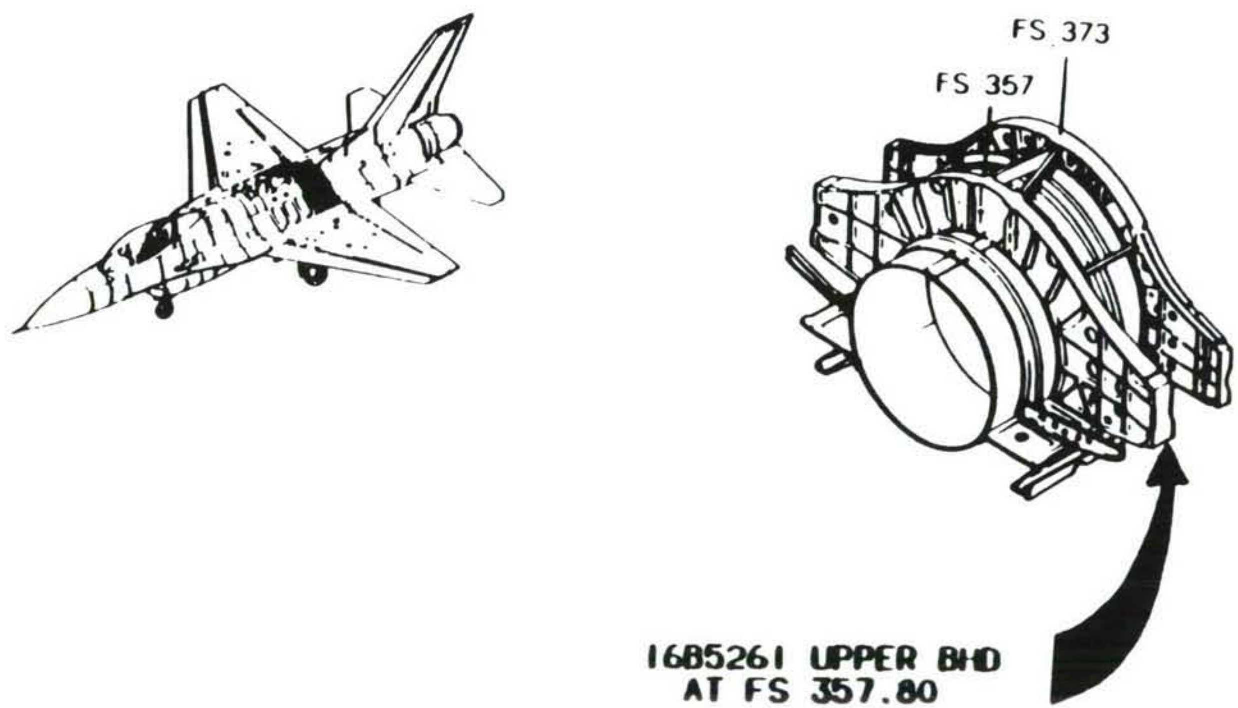
BULKHEAD AT F.S. 341.8
REPAIR



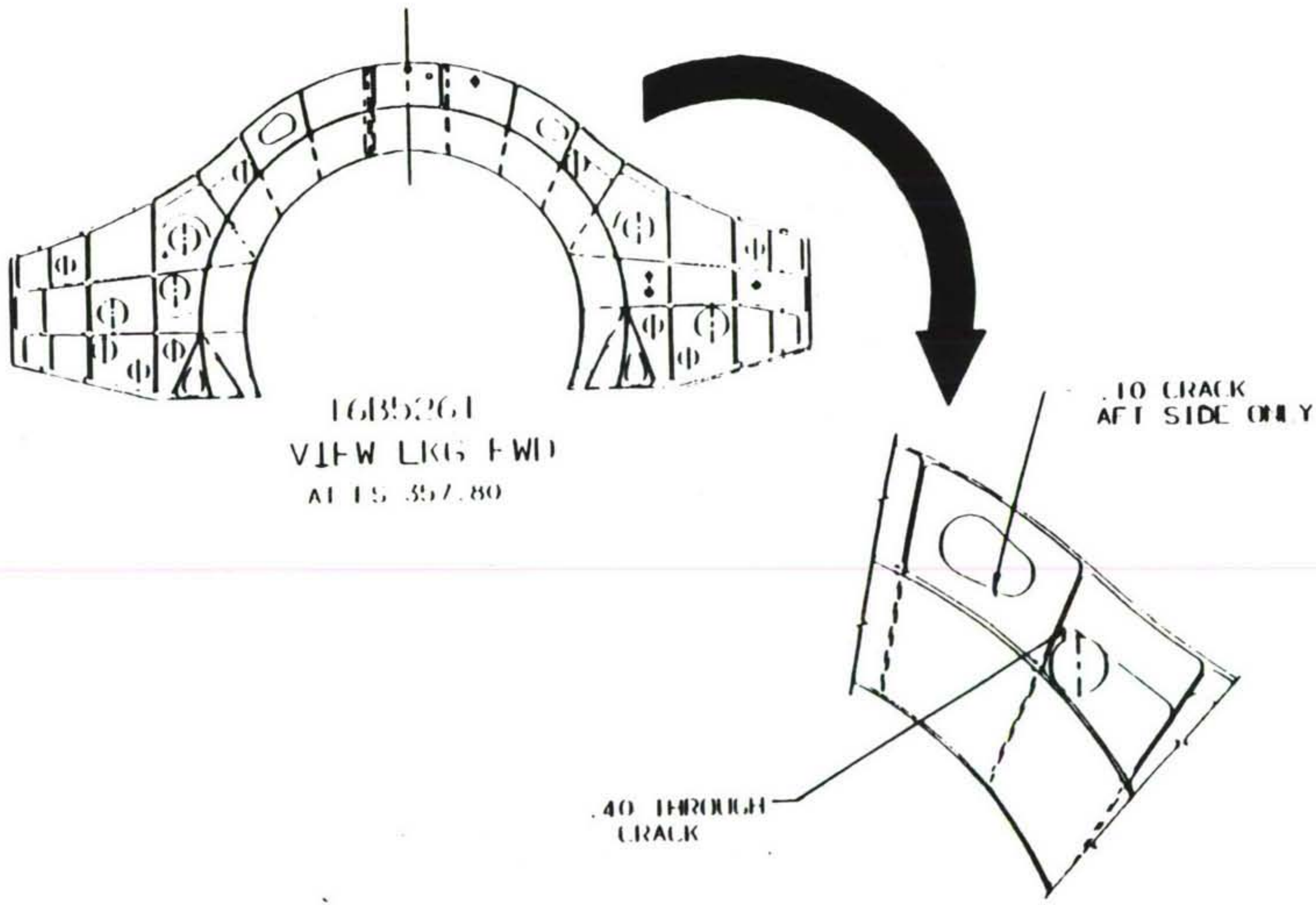
VIEW LKG AFT

ALL HOLES COLDWORKED (Cx) EXCEPT NOTED SEVEN

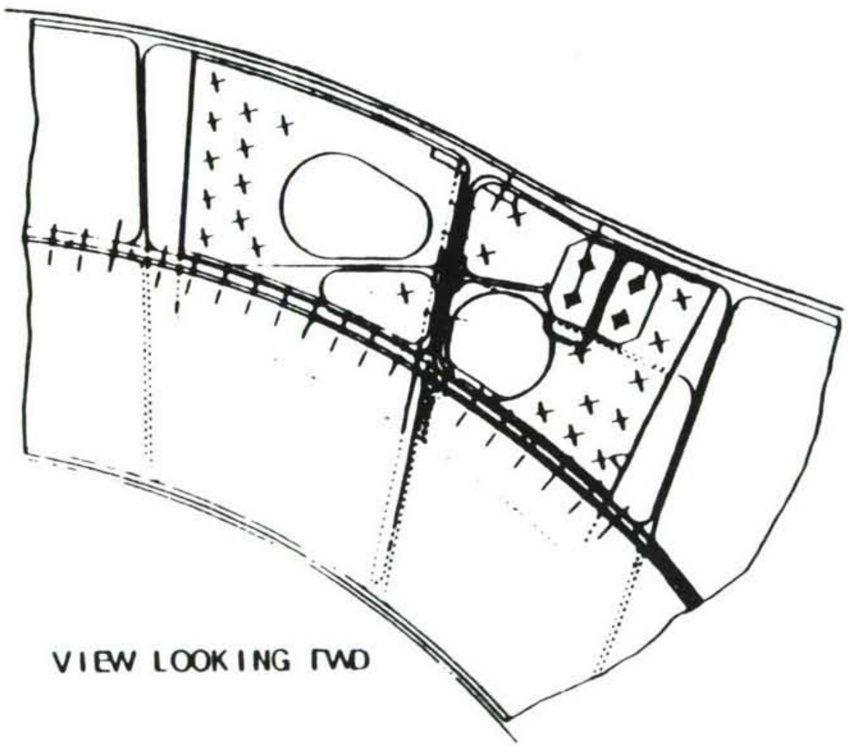
BULKHEAD AT F.S. 357.8
WING CARRY-THROUGH BULKHEAD



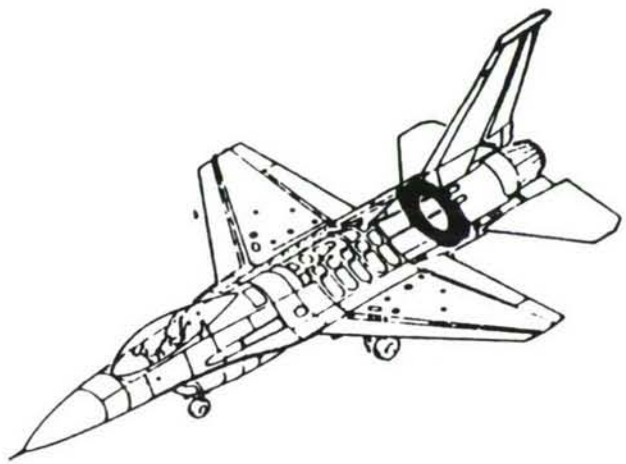
BULKHEAD AT F.S. 357
FAILURE LOCATIONS

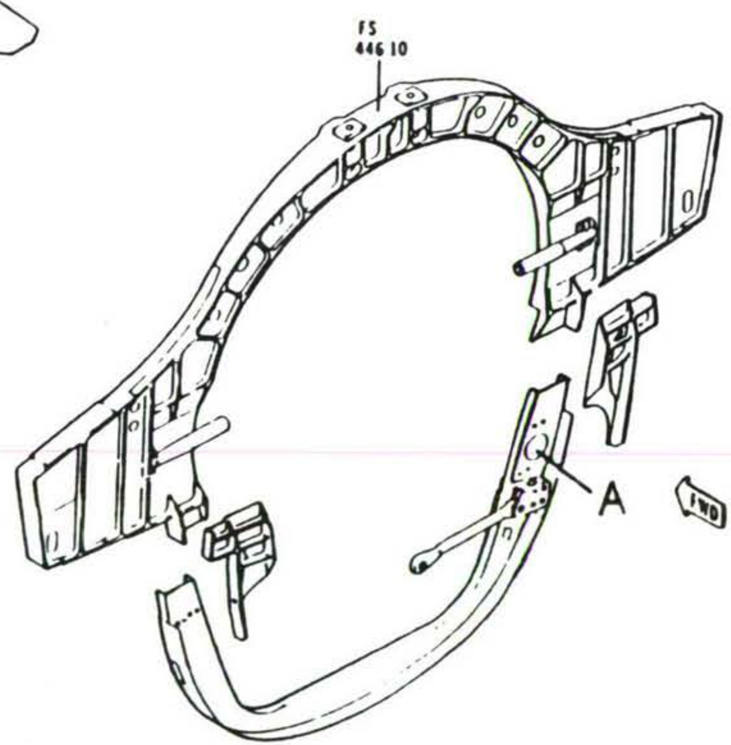


BULKHEAD AT 357.8
REPAIR

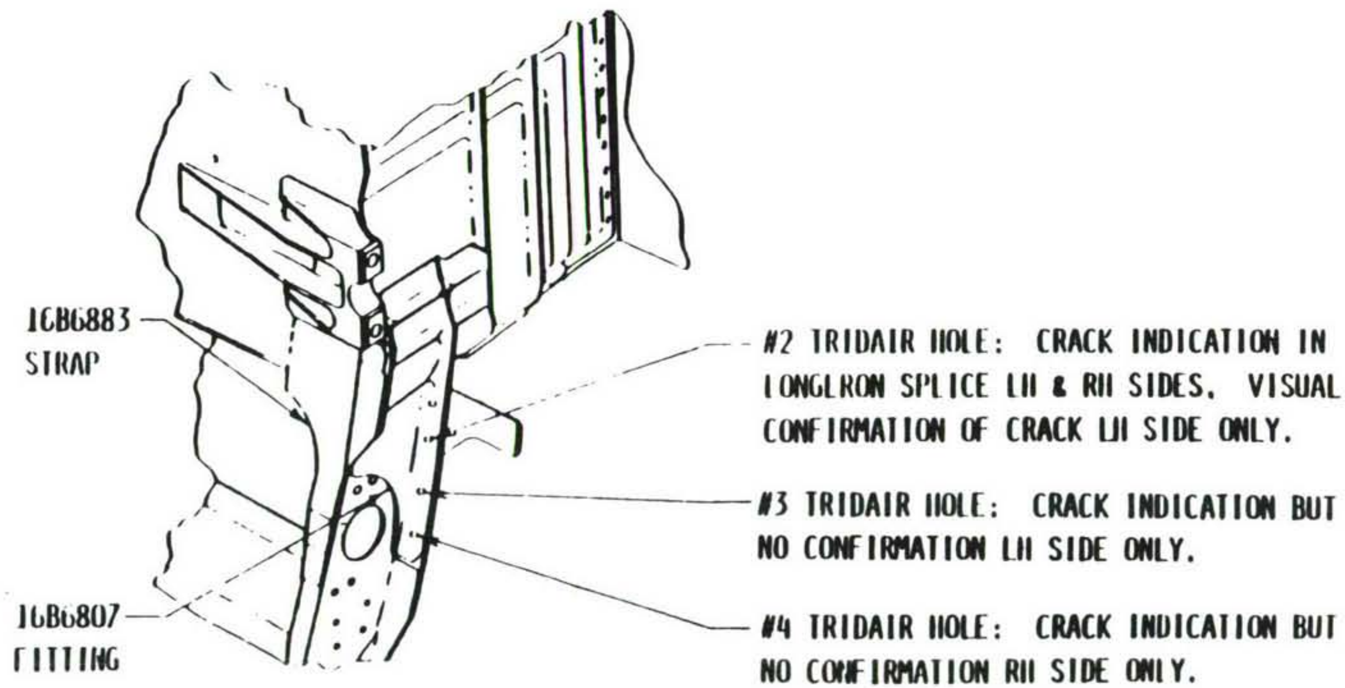


BULKHEAD AT F.S. 446.1

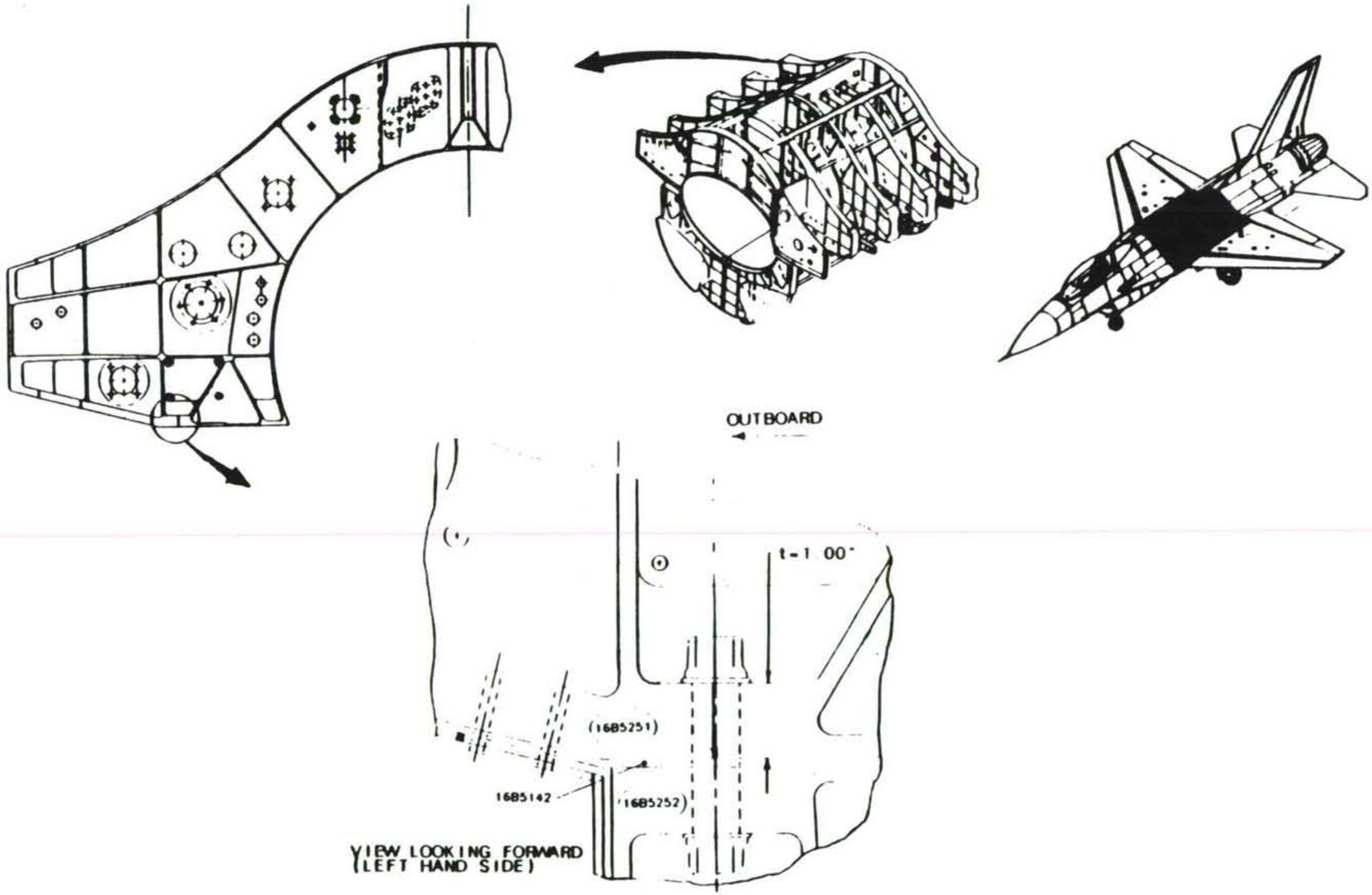




BULKHEAD AT F.S. 446.1
CRACK LOCATIONS



CENTER FUSELAGE BULKHEADS
CRACK LOCATIONS



CENTER FUSELAGE BULKHEADS
CRACKING SUMMARY

- BULKHEAD @ F.S. 309
  - MINOR CRACKING IN FORWARD BOLT HOLE (RHS)
  - FUEL SHELF SKIN CRACKED (LHS & RHS)
- BULKHEAD @ F.S. 325
  - MINOR CRACKING IN FORWARD BOLT HOLE (RHS)
  - FUEL SHELF SKIN CRACKED (LHS & RHS)
- BULKHEAD @ F.S. 341
  - MAJOR CRACKING IN BOTH BOLT HOLES (LHS & RHS)
  - FUEL SHELF SKIN CRACKED (LHS & RHS)
- BULKHEAD @ F.S. 357
  - MINOR CRACKING IN BOTH BOLT HOLES (RHS)
  - FUEL SHELF SKIN CRACKED (LHS & RHS)

CONCLUDING REMARKS

- TEST PROGRAM ESTABLISHING NEW STRUCTURAL ANALYSIS BASELINE

- RETROFIT DESIGN/PLANS FOR FLEET

- TEST RESULTS LEAD THE FLEET FOR TIMELY RETROFIT IMPLEMENTATION

DYNAMIC FATIGUE TESTING OF F-15 VERTICAL TAIL
TO SIMULATE BUFFET ENVIRONMENT

Chart 1 - The F-15 has experienced fatigue cracking of the upper vertical tail due to buffet excitation. This presentation discusses the full scale dynamic test program underway to duplicate the failures and evaluate redesigns.

Chart 2 - Vertical tail buffet is a subsonic phenomenon caused by separated flow from the wing leading edge impinging on the tail. Vertical tail vibration begins at 15° AOA and peaks at 22° AOA. The upper portion of the tail experiences the worst environment. The fact that the steady state load is small during buffet simplifies dynamic test simulation.

Chart 3 - F-15 vertical tail buffet can be defined by two parameters, dynamic pressure and angle of attack. These data are from a sample of average fleet usage recorded by the Signal Data Recorder. For a given AOA, the dynamic response level increases linearly with dynamic pressure.

Chart 4 - The upper portion of the tail has developed cracks in the leading and trailing edges. Cracks typically show up around 1000 flight hours. The leading edge cracking is the biggest problem because it is built-up honeycomb structure that is difficult to repair. Leading edge skin cracks initiate in a chem-mil radius. The left tail cracks much earlier than the right because it has a larger tip pod mass.

Chart 5 - A six step plan was implemented to understand the problem, develop a solution, and verify the solution via testing. (1) The environment and response were determined from flight test and wind tunnel data. (2) The failures were correlated by analysis, leading to a redesign concept. (3) A full scale dynamic test setup was devised to approximate the airplane environment. (4) Prior to test, the equivalent flight hours per test hour were derived by analysis. (5) A baseline test was conducted to validate the test setup and predicted flight hours per test hour. (6) The redesigned tail will be tested to the same environment. This paper concentrates on steps (3) through (5).

Chart 6 - Strain gage and accelerometer response data were recorded in a buffet flight test program. These data were recorded for dynamic pressure/angle-of-attack variations throughout the buffet regime. From these data and operational usage data recorded by the Signal Data Recorder, cycle-by-cycle strain time history spectra were developed for control points on the front and rear spars. These spectra were used to correlate with service failures, develop redesigns, and to tailor the full scale test environment.

Chart 7 - The buffet flight test data and ground vibration test data indicate that one dynamic mode is the main culprit. The critical mode is a combined vertical tail 2nd bending and 1st torsion together with stabilator pitch. The frequency of this mode is 30 hz which causes it to be closely coupled with the vertical tail 32 hz first torsion mode. The interaction of the vertical and horizontal tails in the critical mode indicated that a complete empennage was needed for full scale testing.

Chart 8 - To help understand the problem and investigate solutions, a NASTRAN dynamic model was developed. A coarse grid model of the fuselage, boom and horizontal tail was combined with a fine grid model of the vertical tail. The vertical tail model has increasing detail in the failure areas. The model was used to correlate with airplane response data, correlate with service failures, evaluate redesign concepts, and evaluate alternate full scale test approaches. The total model contained over 18000 degrees of freedom.

Chart 9 - The dynamic model correlates favorably with the airplane natural frequencies and mode shapes. The peak stress areas identified by the model corresponded to the areas cracking in service. However, the model underpredicted the magnitude of peak stress. Additional refinement of the model to improve this correlation was not attempted.

Chart 10 - Through studying alternate designs with the dynamic model, a full scale test setup was devised to accurately simulate the aircraft response. A full empennage was required because of dynamic coupling between the vertical and horizontal tails. To overcome the velocity and force limitations of electrodynamic shakers, a hydraulic shaker system was procured and installed. A narrow band random input spectrum with single point excitation was tailored to excite the structural modes of importance. The input level was adjusted to best duplicate the response levels measured in flight.

Chart 11 - The test article natural frequencies and mode shapes closely duplicated the airplane response. The test article response data were obtained through a GVT with the excitation provided by the single point excitation system to be used for the full scale test.

Chart 12 - To meet the requirement to analytically predict flight hours per test hour prior to test and to estimate test time to failure, the stress field of the tail was mapped with strain gages. The highest stressed areas adjacent to the failure locations were located using multiple gage installations and shaker input somewhat below the desired test level. The peak stress level was then related to the shaker input level and to stress levels at the flight test and lab test control points.

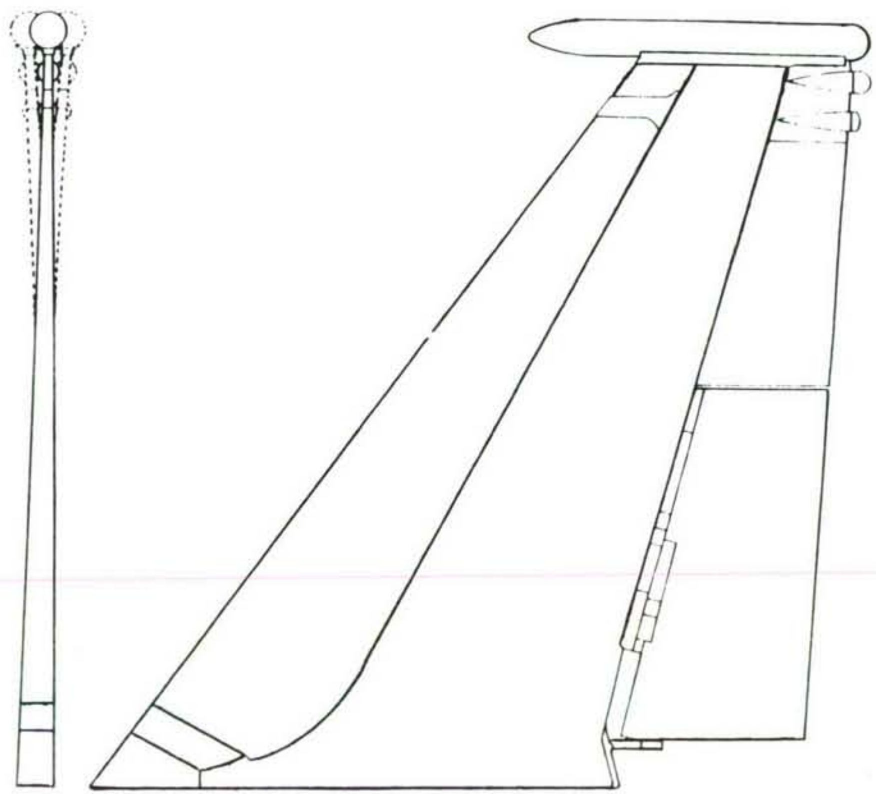
Chart 13 - Because of the requirement to accurately predict test failure time, the analytical fatigue routine was correlated with element tests. The specimens were tested to a strain spectrum recorded during development runs on the test article. Predicted crack initiation lives were compared to test lives for a range of stress levels. The predicted lives were shorter than test lives for all stress levels. Correlation factors dependent on stress level were developed to adjust the analytical predictions.

Chart 14 - Analysis was used to predict flight hours per test hour and test failure time prior to start of test. The analysis was keyed to the leading edge failures which are far more costly to repair than the trailing edge box. Theory and a PROBE finite element model were used to compute the peak K_T. relative to stress at the lab test control point. Strains measured on the lab test article were used to relate the peak K_T. to the stress at the flight control point. Crack initiation lives were then computed using lab test and flight test spectra at the respective control points. The resulting predictions were that the lab test article would crack in 4 hours and that 1 lab test hour was worth 290 flight hours. At this point it was time to test the current design (baseline) tail to failure to try to duplicate service failures and validate the analytical flight hours per test hour.

Chart 15 - The baseline test article was tested until the leading and trailing edges cracked. The aft box cracked first at 2.5 hours. Next, the leading edge skin cracked along the chem mil line at 3.5 hours, very close to the predicted failure time of 4 hours. The crack locations duplicated the service failures. In addition, the leading edge skin failure at 3.5 hours was equivalent to 1015 flight hours per the analytical flight hour per test hour calculations. The 1015 flight hour failure point falls in the middle of the scatter band of service failures. This fact validated the analytical derivation of 290 flight hours per test hour. The next step will be to test the redesigned tail to the same environment. The test goal will be 55 test hours or 55 X 290 = 16000 equivalent flight hours.

Chart 16 - The testing to date has resulted in several conclusions/lessons learned. First, service failures were successfully duplicated with a full scale dynamic test using single point excitation. This was made possible because the steady state loads are negligible on the upper vertical tail when in the damaging buffet regime. Second, the baseline test was invaluable for establishing the test level and correlating with service failures. Third, fatigue analysis methodology required tailoring for buffet spectra via element testing. Fourth, the dynamic model by itself would have overpredicted the failure time due to underpredicting the peak dynamic stress. Additional refinement of the model may have improved this prediction capability but was not attempted. Finally, the dynamic model was an invaluable tool for understanding the problem, evaluating redesigns and developing a test approach.

DYNAMIC FATIGUE TESTING OF F-15 VERTICAL TAIL TO SIMULATE BUFFET ENVIRONMENT



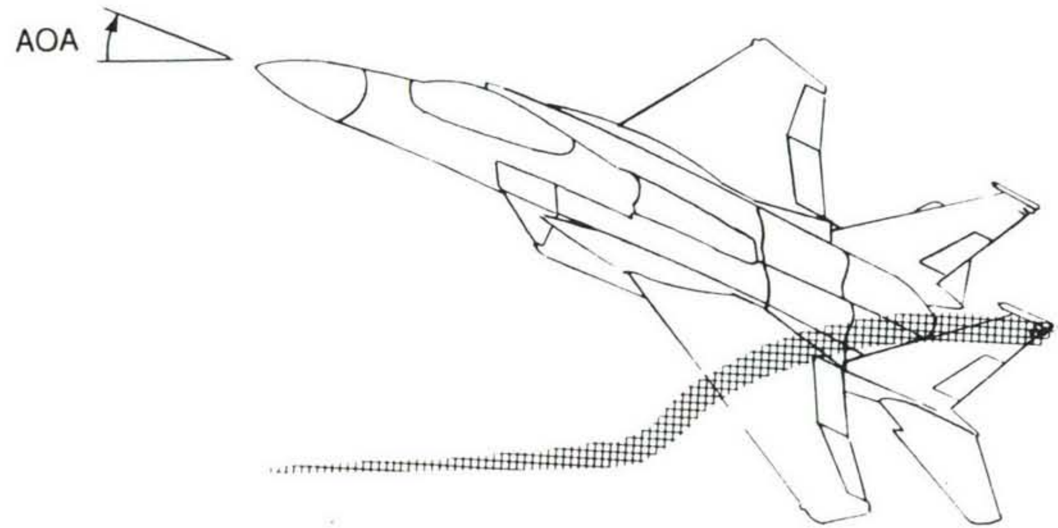
PRESENTED BY:

RONALD MELLIERE
MCDONNELL AIRCRAFT COMPANY

JACK RESNICKY
AERONAUTICAL SYSTEMS DIVISION

USAF STRUCTURAL INTEGRITY
PROGRAM CONFERENCE
5 - 7 DECEMBER, 1989

UPPER VERTICAL TAIL IS SUBJECTED TO BUFFET LOADS

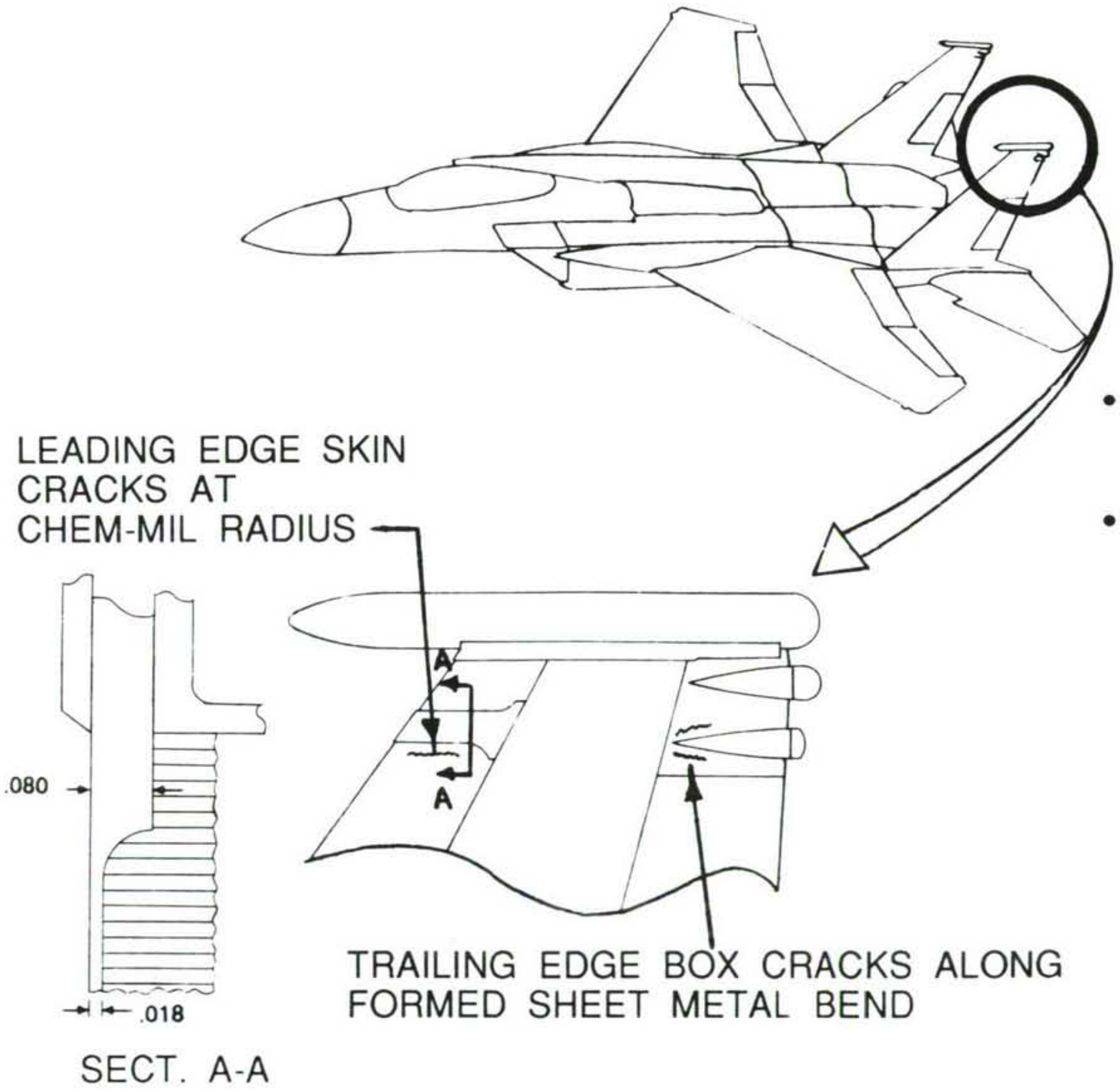


- BUFFET IS SUBSONIC PHENOMENON
- CAUSE IS SEPARATED FLOW FROM WING LEADING EDGE AT HIGH AOA
- TAIL VIBRATION BEGINS AT 15° AOA, PEAKS AT 22° AOA
- SEVERITY AT GIVEN AOA INCREASES WITH DYNAMIC PRESSURE
- STEADY STATE LOAD IS SMALL DURING BUFFET

BUFFET REGIME IS DEFINED BY DYNAMIC PRESSURE AND ANGLE-OF-ATTACK

DYNAMIC PRESSURE (LBS / FT^2)	SECONDS PER 1000 FLIGHT HOURS		
	AOA 15° – 20°	AOA 20° – 25°	AOA 25° – 30°
150 - 200	6420	2360	296
200 - 250	4260	580	12
250 - 300	3210	94	0
300 - 350	1325	102	0
350 - 400	450	19	0
400 - 450	460	0	0
450 - 500	138	0	0
500 - 550	68	0	0

UPPER PORTION OF TAIL HAS DEVELOPED CRACKS DUE TO BUFFET

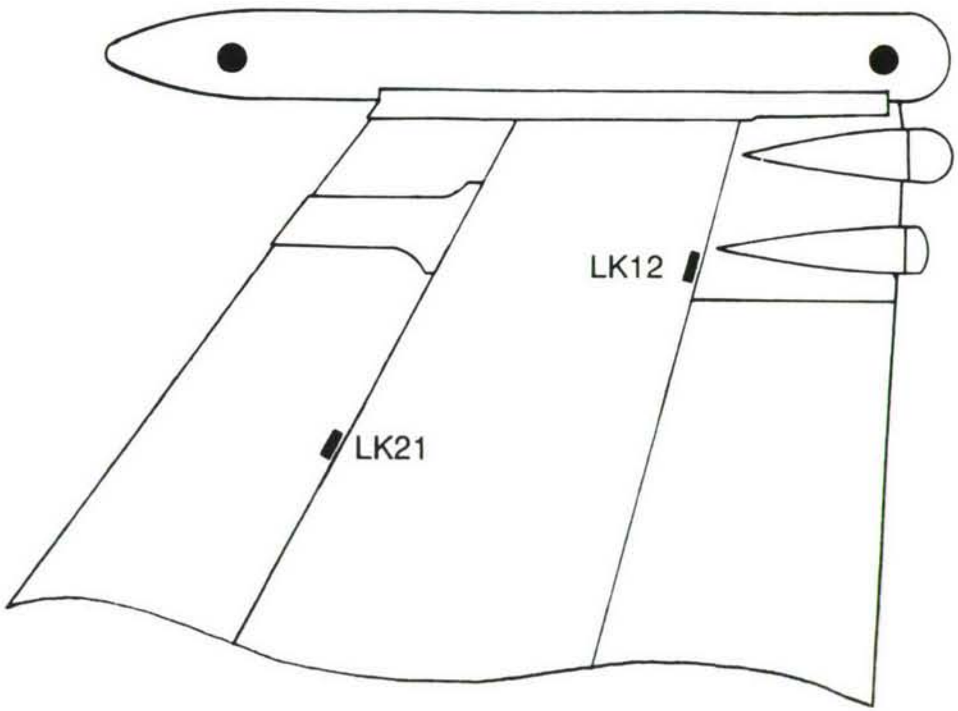


- SERVICE CRACKS OCCUR AT ≈ 1000 FLIGHT HOURS
- LEFT TAILS CRACK MUCH EARLIER THAN RIGHT (LARGER TIP POD)

PLAN OF ATTACK WAS FORMULATED

1. CHARACTERIZE AIRPLANE ENVIRONMENT AND RESPONSE
2. ANALYTICALLY CORRELATE WITH FAILURES AND DEVELOP REDESIGN
3. DEVELOP FULL SCALE DYNAMIC TEST SETUP TO APPROXIMATE AIRPLANE FREQUENCIES, MODE SHAPES, AND RESPONSE LEVELS
4. ANALYTICALLY PREDICT FLIGHT HOURS PER TEST HOUR PRIOR TO TEST
5. TEST BASELINE TAIL TO FAILURE:
   - VALIDATE TEST AND ANALYSES
6. TEST REDESIGNED TAIL TO SAME ENVIRONMENT:
   - QUALIFY FOR 2 LIFETIMES
   - IDENTIFY NEXT HOTSPOTS

ANALYTICAL FATIGUE SPECTRA WERE DEVELOPED FROM FLIGHT DATA



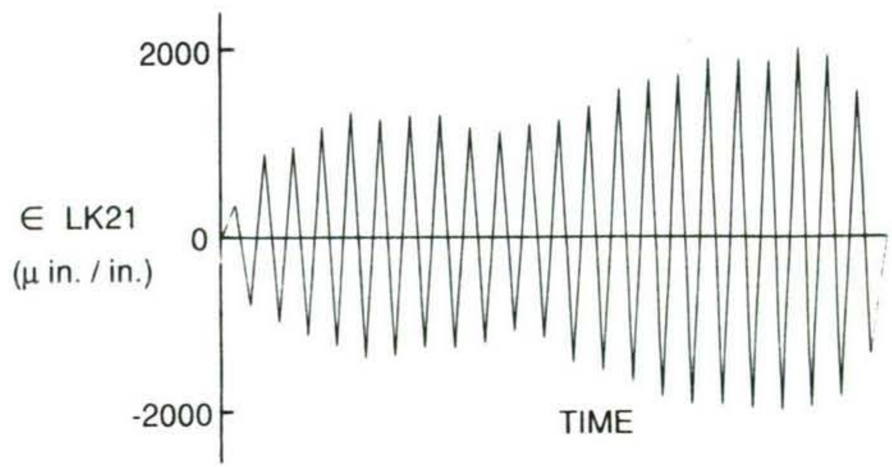
I STRAIN GAGE

● ACCELEROMETER

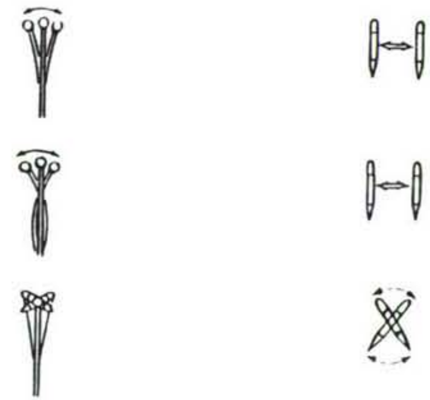
A. STRAIN RESPONSE RECORDED FOR DYNAMIC PRESSURE (q) AND ANGLE-OF-ATTACK (α) VARIATIONS THROUGHOUT BUFFET REGIME

B. SIGNAL DATA RECORDER DEFINED TIME IN EACH q/α REGIME

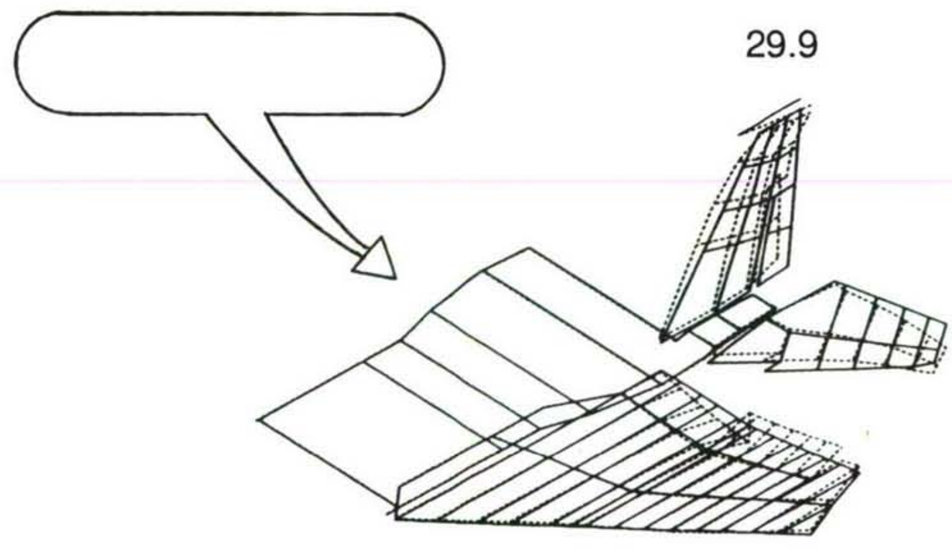
C. CYCLE-BY CYCLE STRAIN HISTORY SPECTRA DEVELOPED FROM A & B

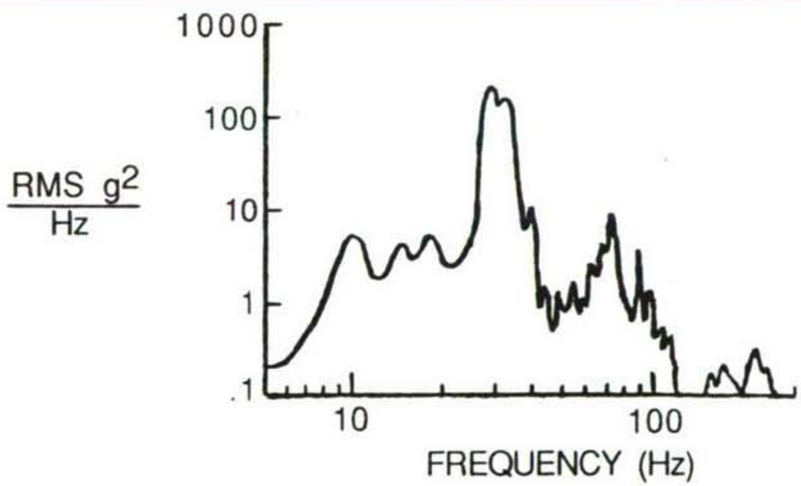


ONE DYNAMIC MODE IS THE MAJOR PLAYER
(BASED ON AIRPLANE DATA - FLIGHT TEST AND GVT)

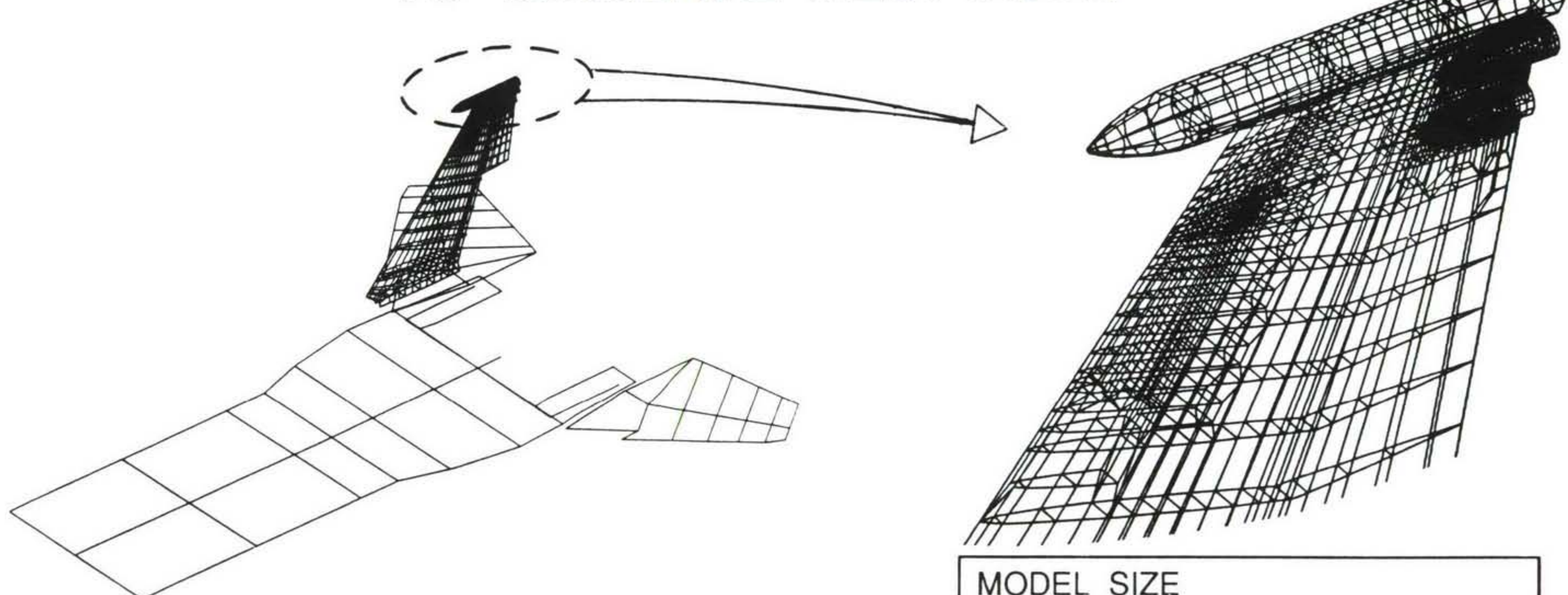


	FREQUENCY (HZ)
• VERTICAL TAIL 1st BENDING	9.2
• VERTICAL TAIL 2nd BENDING	36.3
• VERTICAL TAIL 1st TORSION	32.0
• COMBINED VERTICAL TAIL 2nd BENDING & 1st TORSION, AND STABILATOR PITCH	29.9





NASTRAN DYNAMIC MODEL WAS DEVELOPED TO SIMULATE RESPONSE

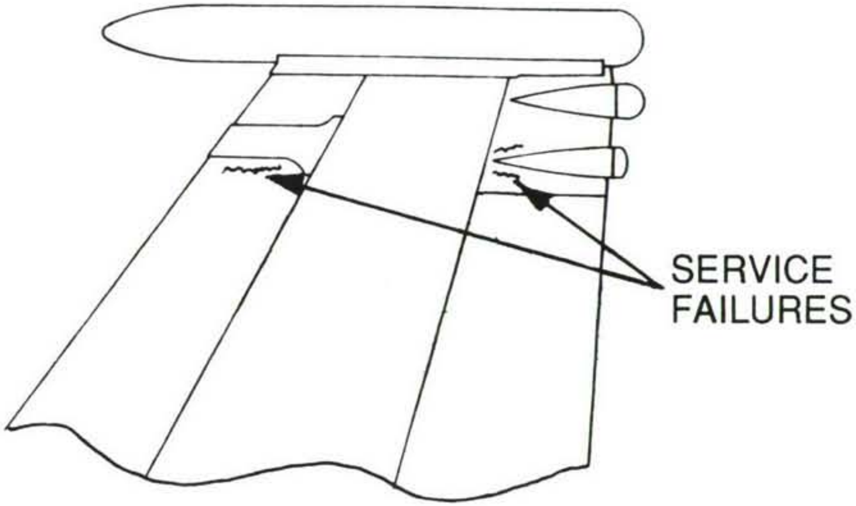


MODEL SIZE	
7040	PLATE ELEMENTS
1200	BAR ELEMENTS
6020	GRID POINTS
18060	DEGREES OF FREEDOM

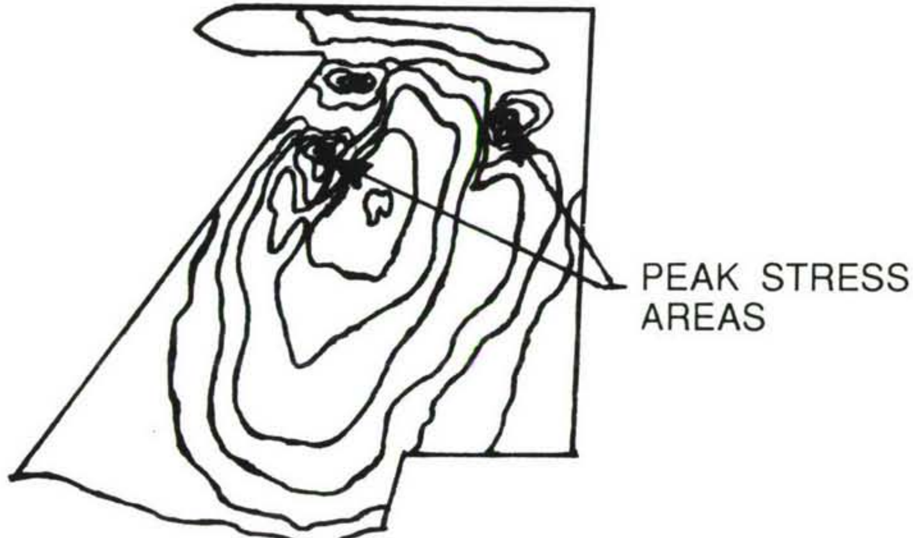
MODEL WAS USED TO –

- CORRELATE WITH AIRPLANE MEASUREMENTS
- CORRELATE WITH SERVICE FAILURES
- EVALUATE REDESIGNS
- EVALUATE TEST APPROACHES

DYNAMIC MODEL CORRELATES WITH AIRPLANE FREQUENCIES AND MODE SHAPES



AIRPLANE



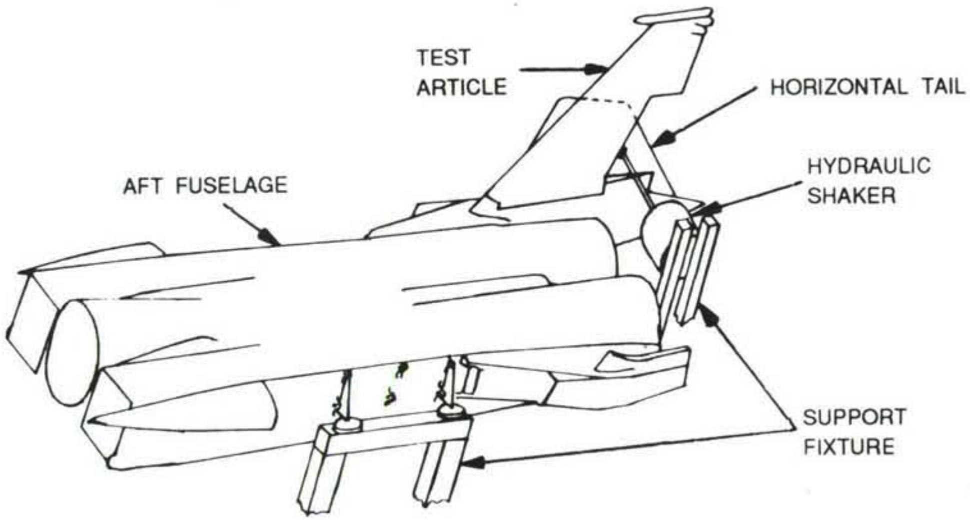
NASTRAN MODEL
(RMS RESPONSE - ALL MODES)

- NATURAL FREQUENCIES -

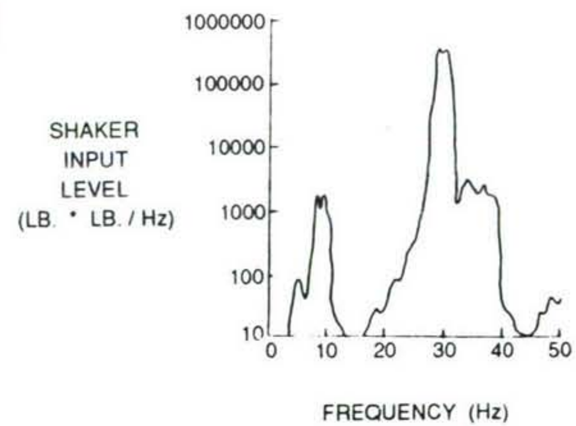
	AIRPLANE	MODEL
VERTICAL TAIL 1st BENDING	9.2	9.3
VERTICAL TAIL 2nd BENDING	36.3	35.7
VERTICAL TAIL 1st TORSION	32.0	33.3
COMBINED VERT. TAIL/STABILATOR	29.9	32.3

- STRESSES -
  - * LOCATIONS OF PEAK STRESS CORRELATE WITH AIRPLANE
  - * MODEL PREDICTED LOWER PEAK STRESSES THAN AIRPLANE

FULL SCALE TEST SETUP WAS DEVISED



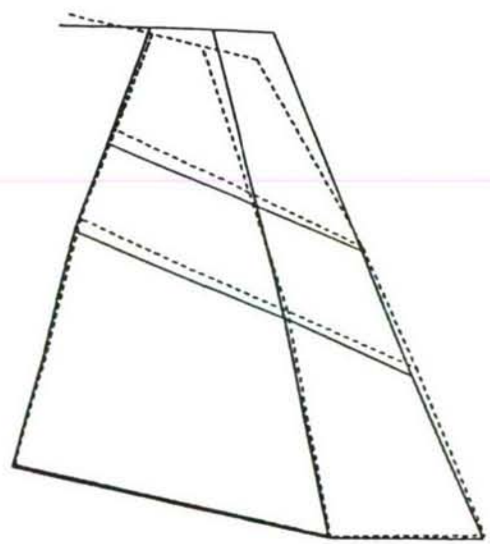
- SINGLE POINT EXCITATION
- FULL EMPENNAGE REQUIRED BECAUSE OF DYNAMIC COUPLING BETWEEN VERTICAL AND HORIZONTAL TAILS
- HYDRAULIC SHAKER USED TO OVERCOME VELOCITY AND FORCE LIMITATIONS OF ELECTRODYNAMIC SHAKERS
- NARROW BAND RANDOM INPUT SPECTRUM DEVELOPED TO EXCITE STRUCTURAL MODES OF IMPORTANCE

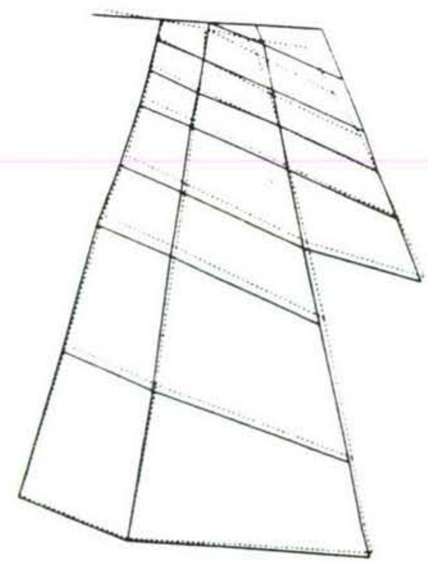


TEST ARTICLE CORRELATED WITH AIRPLANE

	NATURAL FREQUENCY - Hz	
	AIRPLANE GVT	TEST ARTICLE GVT
• VERTICAL TAIL 1st BENDING	9.2	9.6
• VERTICAL TAIL 2nd BENDING	36.3	36.9
• VERTICAL TAIL 1st TORSION	32.0	32.1
• COMBINED VERT. TAIL/STABILATOR	29.9	30.2

MODE SHAPES
CORRELATED
(E.G. 30 Hz MODE)

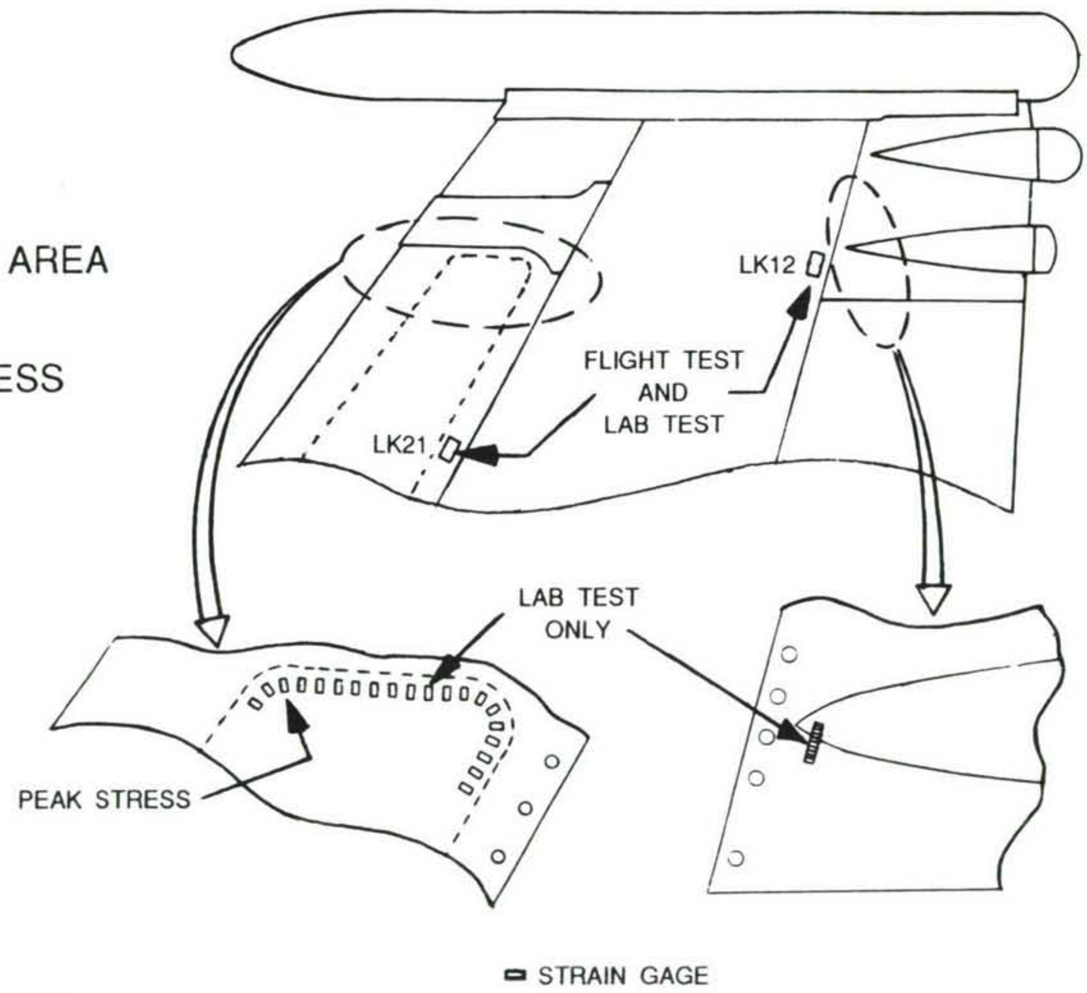

AIRPLANE


TEST ARTICLE

TEST ARTICLE STRESS FIELD WAS MAPPED WITH STRAIN GAGES

TASKS

- DETERMINE HIGHEST STRESSED AREA NEAR FAILURE LOCATIONS
- RELATE FAILURE LOCATION STRESS LEVEL TO -
  - * SHAKER INPUT
  - * FLIGHT TEST CONTROL POINTS (LK21 & LK12)

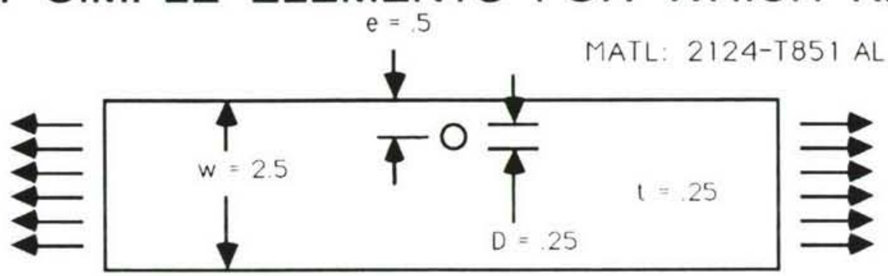


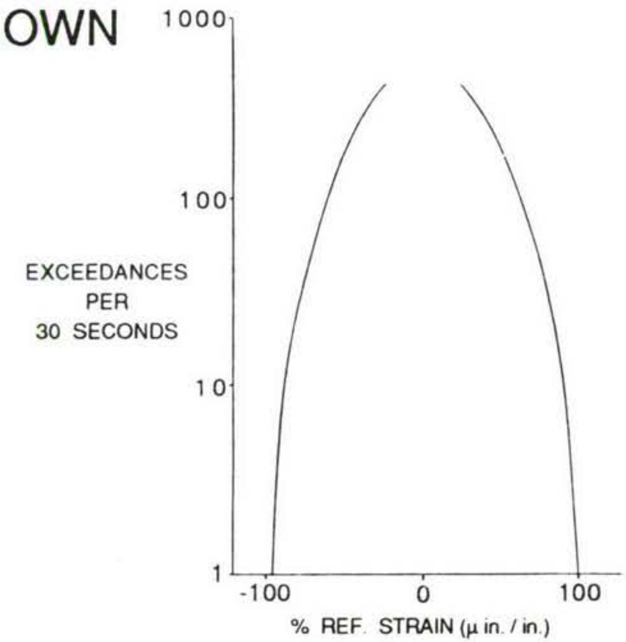
GOAL: ANALYTICALLY PREDICT -

- * TEST TIME TO FAILURE
- * EQUIVALENT FLIGHT HOURS PER TEST HOUR

ELEMENT TESTS WERE RUN TO TAILOR ANALYTICAL FATIGUE PREDICTIONS

- TEST SIMPLE ELEMENTS FOR WHICH Kt IS KNOWN

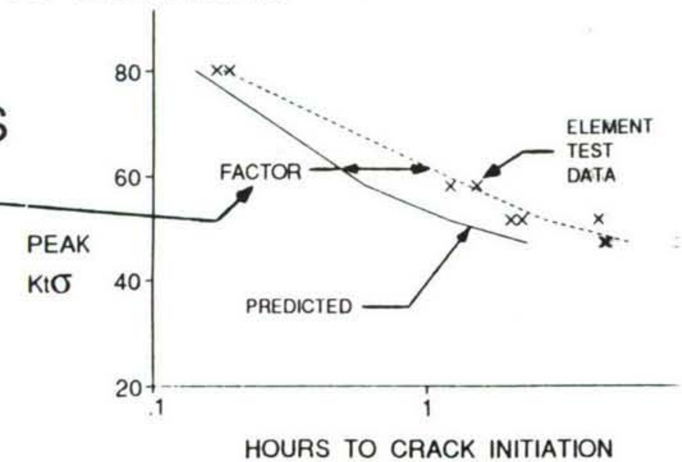




- USE STRAIN SPECTRUM FROM FULL SCALE TEST ARTICLE

- PREDICT CRACK INITIATION LIFE FOR TEST SPECIMENS

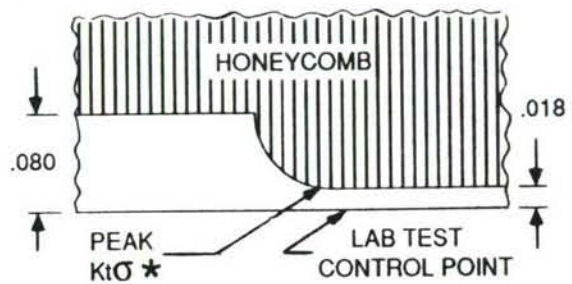
- COMPARE PREDICTED LIVES TO TEST LIVES
  - DEVELOP CORRELATION FACTORS



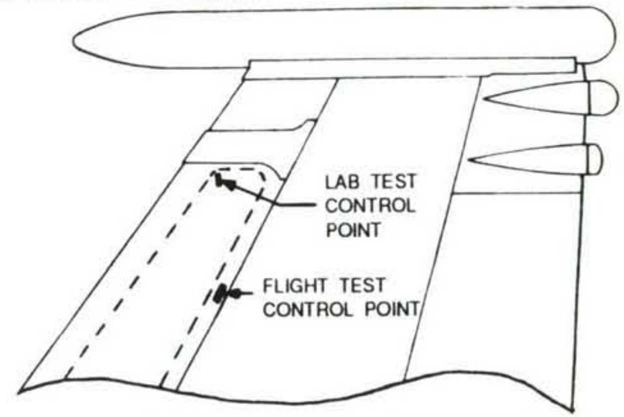
- ADJUST ANALYTICAL PREDICTIONS USING FACTORS

ANALYSIS WAS USED TO PREDICT FLIGHT HOURS PER TEST HOUR

- COMPUTE PEAK $Kt\sigma$ RELATIVE TO LAB CONTROL POINT



$$\longrightarrow \frac{Kt\sigma}{\sigma_{LAB\ TEST\ C.P.}}$$



* USED THEORY AND PROBE MODEL

- USE LAB DATA TO RELATE PEAK $Kt\sigma$ TO FLIGHT CONTROL POINT

$$\frac{Kt\sigma}{\sigma_{FLIGHT\ TEST\ C.P.}} = \frac{Kt\sigma}{\sigma_{LAB\ TEST\ C.P.}} \cdot \left(\sigma_{LAB\ TEST\ C.P.} \Big/ \sigma_{FLIGHT\ TEST\ C.P.} \right)^{*}$$

* DYNAMIC MODEL UNDERPREDICTED THIS FACTOR

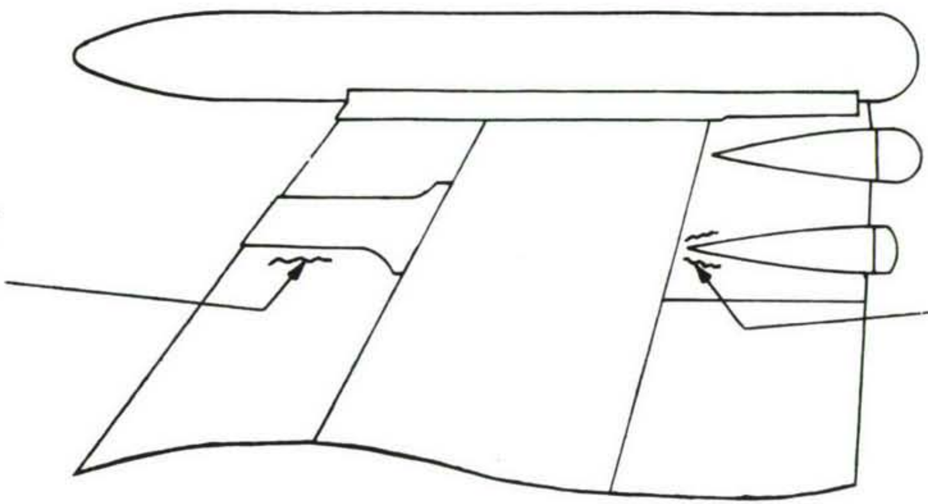
- PREDICT LIVES USING LAB AND FLIGHT SPECTRA

RESULTS:
- LAB TIME TO CRACK INITIATION = 4 HOURS
- FLIGHT TIME TO CRACK INITIATION = 1160 HOURS

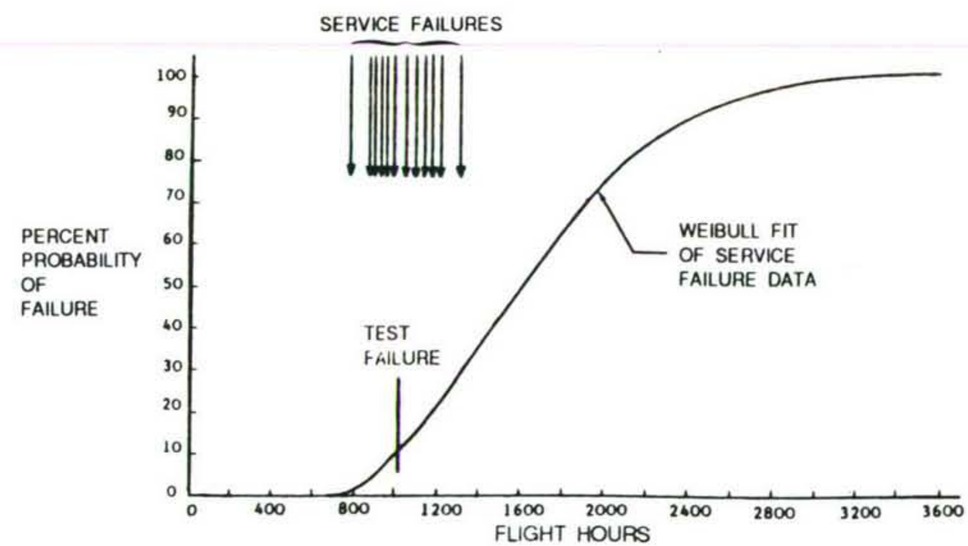
$\therefore$ 1 LAB HOUR = 290 FLIGHT HOURS

BASELINE TEST VALIDATED LOADING AND ANALYSES

- TEST CRACKS DUPLICATED SERVICE FAILURES



- TEST FAILURE TIME MATCHED SERVICE DATA



CONCLUSIONS / LESSONS LEARNED

1. SERVICE FAILURES WERE SUCCESSFULLY DUPLICATED WITH FULL SCALE DYNAMIC TEST USING SINGLE POINT EXCITATION
2. BASELINE TEST DATA WERE EXTREMELY VALUABLE FOR ESTABLISHING TEST LEVEL AND CORRELATING WITH SERVICE FAILURES
3. FATIGUE ANALYSIS METHODOLOGY REQUIRED TAILORING FOR BUFFET SPECTRA
4. DYNAMIC MODEL WAS NOT SUFFICIENT BY ITSELF TO PREDICT FAILURE TIMES
5. DYNAMIC MODEL PROVIDED ENORMOUS INSIGHT INTO DAMAGE MECHANISM AND REQUIRED IMPROVEMENTS

RAPID REPAIR DTA TECHNOLOGY FOR F-16 AIRCRAFT

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STUDY ACCOMPLISHED UNDER CONTRACT F42600-85-D-4910
OGDEN AIR LOGISTIC CENTER (OO-ALC)
HILL AIR FORCE BASE, UT

OUTLINE

- PURPOSE
- BACKGROUND
- APPLICATION TO F-16 AIRCRAFT
- SUMMARY

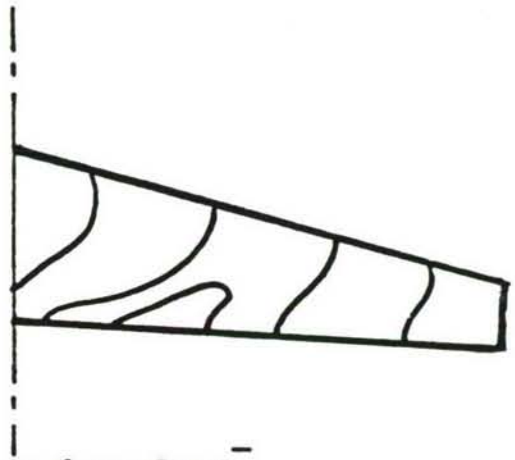
PURPOSE

TO SIMPLIFY THE COMPLICATED ANALYSIS REQUIRED FOR APPLYING DTA TECHNOLOGY TO AIRCRAFT REPAIRS.

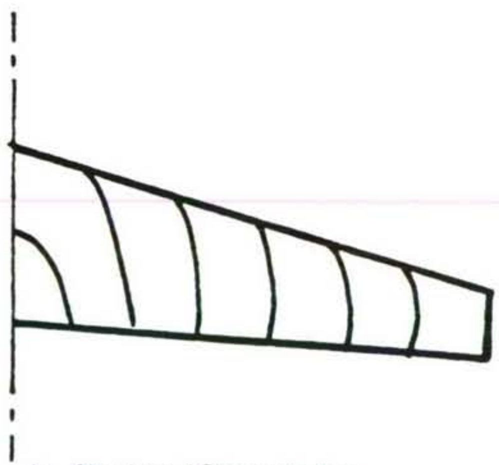
CYCLE-BY-CYCLE (CxC) COMPONENT CRACK GROWTH ANALYSIS

REQUIREMENT	AVAILABILITY
• STRESS INTENSITY FACTOR	• CAN BE GENERATED VIA HANDBOOKS, FINITE ELEMENT, ETC.
• CxC STRESS HISTORY	• NOT NORMALLY AVAILABLE
• MATERIAL PROPERTIES (C.A. $\frac{da}{dN}$, Retardation, ...)	• AVAILABLE
• CRACK GROWTH LIFE CODE	• AVAILABLE BUT MAY REQUIRE EXCESSIVE TIME TO EXERCISE

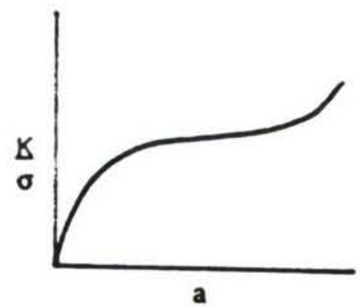
SUGGESTED APPROACH TO RAPIDLY CALCULATE CRACK GROWTH LIFE AT MULTIPLE LOCATIONS IN A MAJOR AIRCRAFT COMPONENT



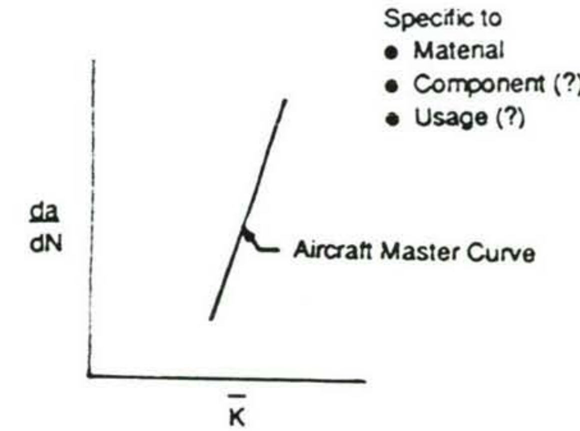
a. Contour Plot of $\overline{\sigma}$



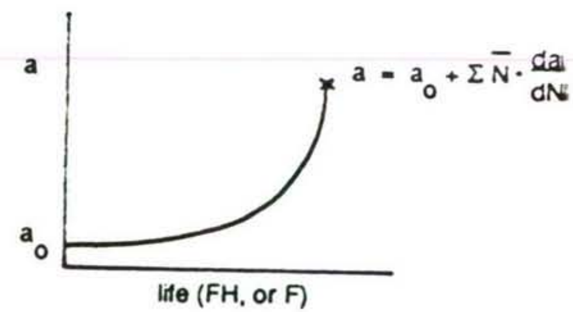
b. Number of Stress Cycles per Unit of Operation ($\overline{N}$)



c. Structural Data for Each Location

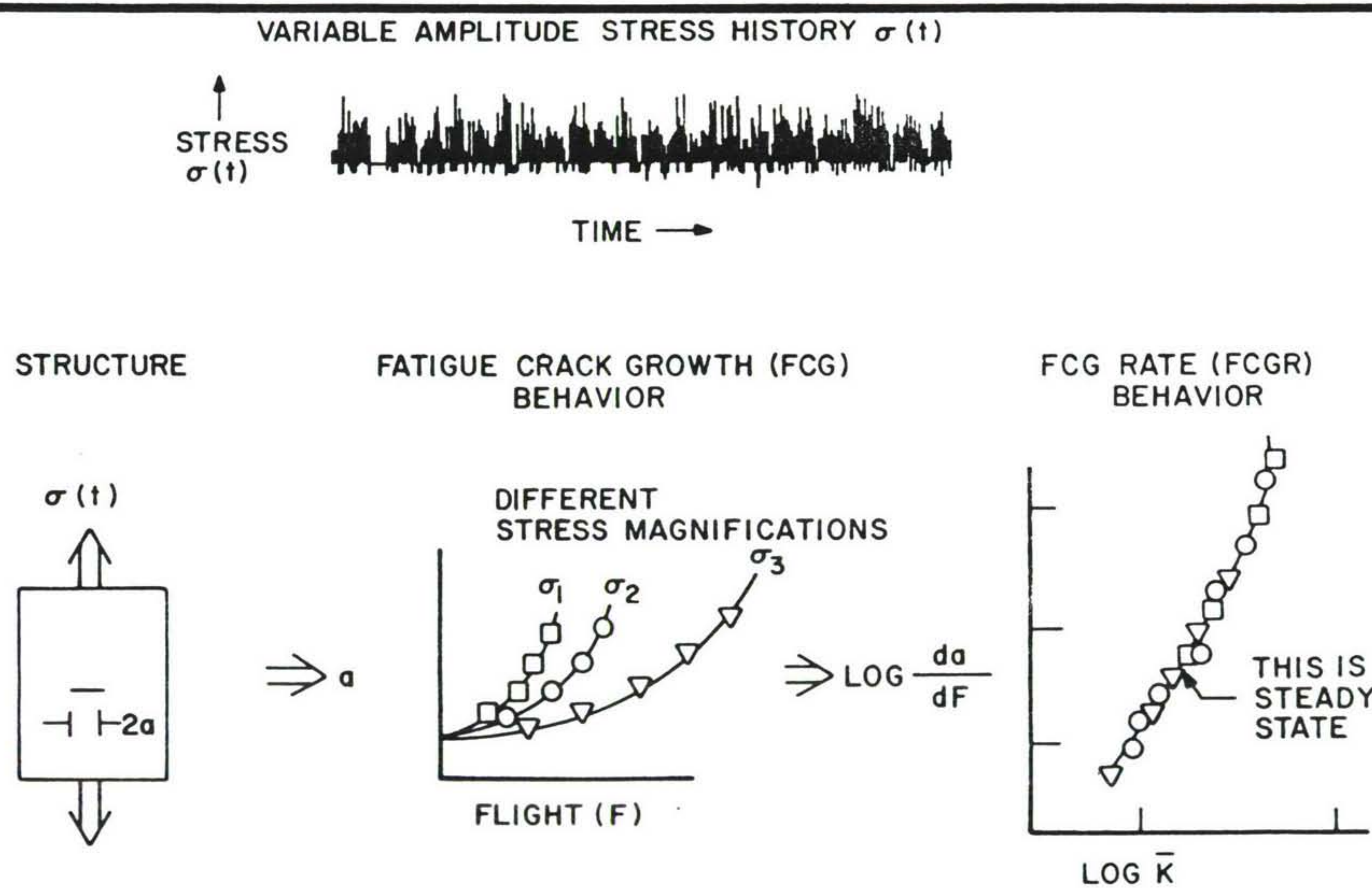


d. FxF FCGR Behavior (all locations)



e. Crack Growth Behavior for Each Location

WHAT IS STEADY-STATE BEHAVIOR?



STEADY STATE BEHAVIOR

- FxF FATIGUE CRACK GROWTH RATE (FCGR) BEHAVIOR IS INDEPENDENT OF STRESS LEVEL SCALING AND CRACK GEOMETRY

- BEHAVIOR IS SIMILAR TO THAT OBSERVED UNDER CONSTANT AMPLITUDE LOADING

- BEHAVIOR CAN BE DESCRIBED AS A FUNCTION OF A CHARACTERISTIC STRESS INTENSITY FACTOR ($\overline{K}$)

CHARACTERISTIC STRESS INTENSITY FACTOR ($\bar{K}$)

$$\bar{K} = \bar{\sigma} \cdot \left[\frac{K}{\sigma}\right]$$

WHERE

$\bar{\sigma}$ = CHARACTERISTIC STRESS FOR SPECTRUM

$\left[\frac{K}{\sigma}\right]$ = STRESS-INTENSITY FACTOR COEFFICIENT

CHARACTERISTIC STRESS ($\bar{\sigma}$) PARAMETERS

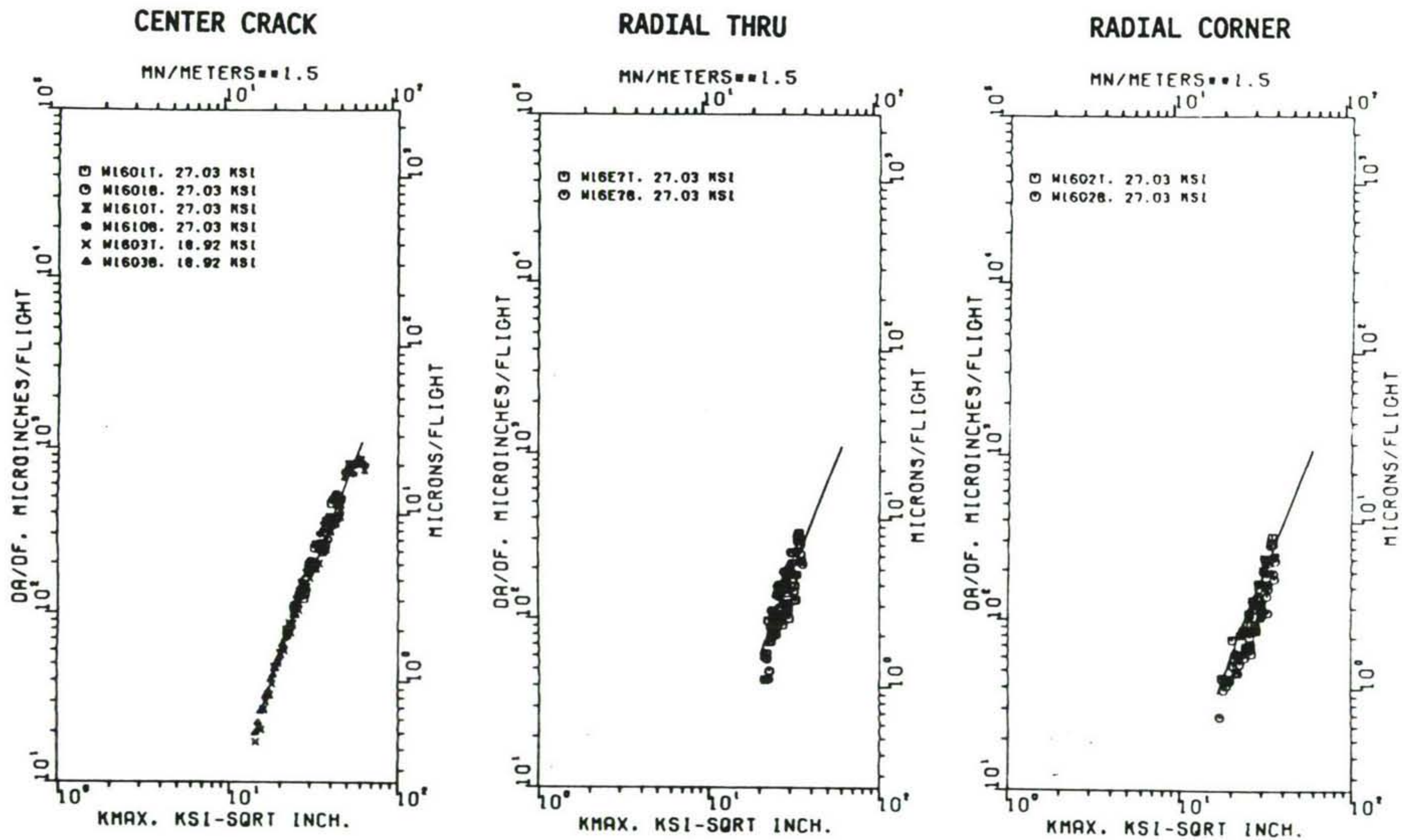
- MAXIMUM SPECTRUM STRESS (σ_{MAX})

- RMS MAXIMUM STRESS (σ_{MAX}^{RMS})

- RMS STRESS RANGE ($\Delta\sigma_{RMS}$)

- SINGLE VALUE CHARACTERIZATION OF ALL STRESSES IN HISTORY

FXF STEADY STATE BEHAVIOR FOR ONE STRESS HISTORY



10

OBSERVATIONS ABOUT STEADY STATE BEHAVIOR

- FxF RATE CURVES CAN BE GENERATED EITHER EXPERIMENTALLY OR ANALYTICALLY
- IF GENERATED ANALYTICALLY, THEN PREDICTION METHODOLOGY MUST BE VERIFIED BY EXPERIMENTAL DATA
- UNLESS CHARACTERISTIC STRESS ($\bar{\sigma}$) IS CHOSEN CAREFULLY, DIFFERENT STRESS HISTORIES WILL HAVE DIFFERENT FxF RATE CURVES

STEADY STATE CONCEPT APPLIED TO AIRCRAFT COMPONENTS

- FIND A STRESS PARAMETER WHICH MAKES IT POSSIBLE TO RELATE FxF CRACK GROWTH RATE BEHAVIOR INDEPENDENT OF LOCATION AND USAGE
- THIS STRESS PARAMETER WILL CHARACTERIZE THE PRIMARY DAMAGING ELEMENTS IN INDIVIDUAL HISTORIES
- APPLY AND VERIFY USE OF SIMPLIFYING FxF ANALYSIS FOR F-16 AIRCRAFT COMPONENTS

F-16 AIRCRAFT DATA

- 500 FLIGHT HOUR STRESS HISTORIES
  - 20 WING (16 BASELINE)
  - 12 FUSELAGE (6 BASELINE)
  - 5 TAIL (3 BASELINE)

- MAXIMUM SPECTRUM STRESSES
  VARIED FROM 17.5 TO 41.8 KSI

- CYCLE COUNTS / 500 FLIGHT HOURS
  VARIED FROM 1200 TO 20,600 CYCLES

- MATERIALS AND ENVIRONMENTS CONSIDERED
  - 7475-T7351
  - 2124-T851
  - PH 13-8MO
  - HIGH HUMID AIR
  - SUMP TANK WATER
  - JP4 FUEL

CHARACTERISTIC STRESS ($\bar{\sigma}$) FOR F-16 AIRCRAFT

- RANGE-PAIR STRESS HISTORIES
- TRUNCATE (SCALE DESIGN LIMIT STRESS TO 30 KSI)
  - $\Delta\sigma_i < \Delta\sigma_{TR} = 4$ KSI
  - $\sigma_{MAX_i} < \sigma_{MAX_{TR}} = 0$
- COUNT REMAINING CYCLES (~14000 CYCLES/500 FLT HRS)
- APPLY RMS STRESS RANGE FORMULA AND SET $\bar{\sigma} = \Delta\sigma_{RMS}$

$$\Delta\sigma_{RMS} = \left[\frac{\sum_{i=1}^{N} [\Delta\sigma_i]^2}{N} \right]^{1/2} = \bar{\sigma}$$

CHARACTERISTIC STRESS INTENSITY FACTOR ($\bar{K}$)

$$\bar{K} = \bar{\sigma} \cdot \left[\frac{K}{\sigma}\right]$$

WHERE

$\bar{\sigma}$ = CHARACTERISTIC STRESS FOR SPECTRUM

$\left[\frac{K}{\sigma}\right]$ = STRESS-INTENSITY FACTOR COEFFICIENT

AND FOR THE F-16

$\bar{\sigma} = \Delta\sigma_{rms}$, THE RMS STRESS RANGE

$\bar{K} = \Delta K_{rms}$, THE RMS STRESS INTENSITY FACTOR RANGE

FxF FATIGUE CRACK GROWTH RATE

CRACK GROWTH RATE PER BLOCK

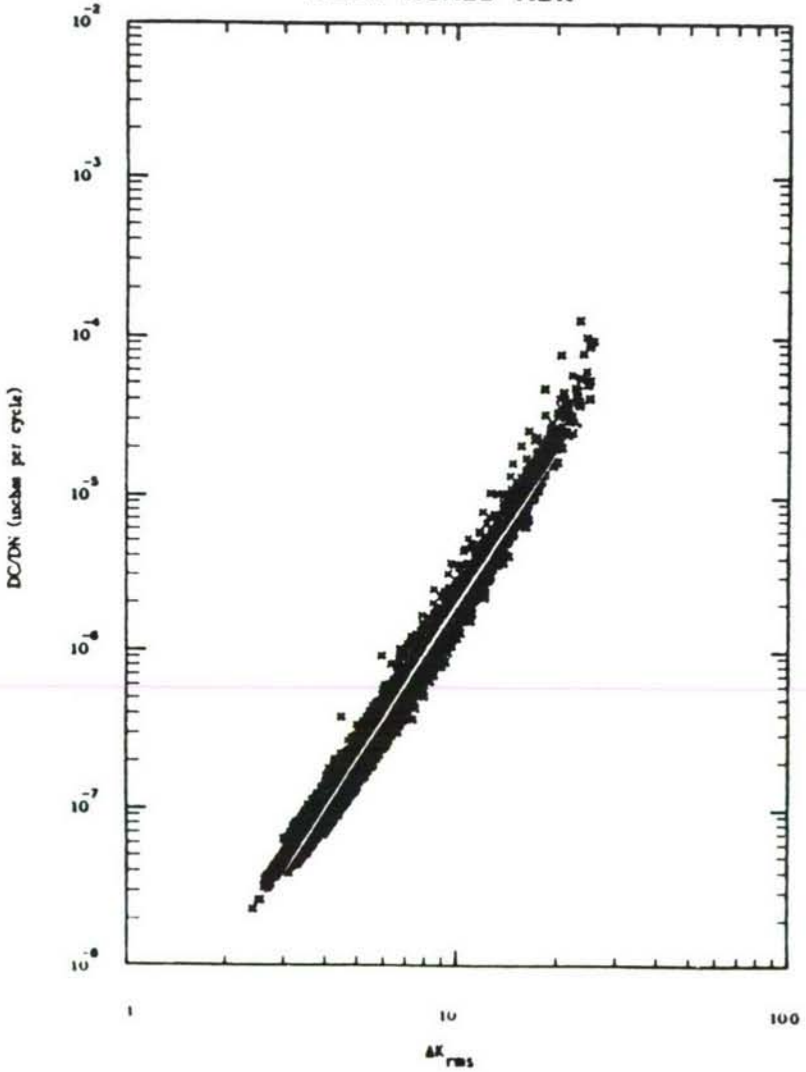
$$\frac{dc}{dB} = \frac{\Delta c}{\Delta B}$$

CRACK GROWTH RATE PER CYCLE (XXX CYCLES/BLOCK)

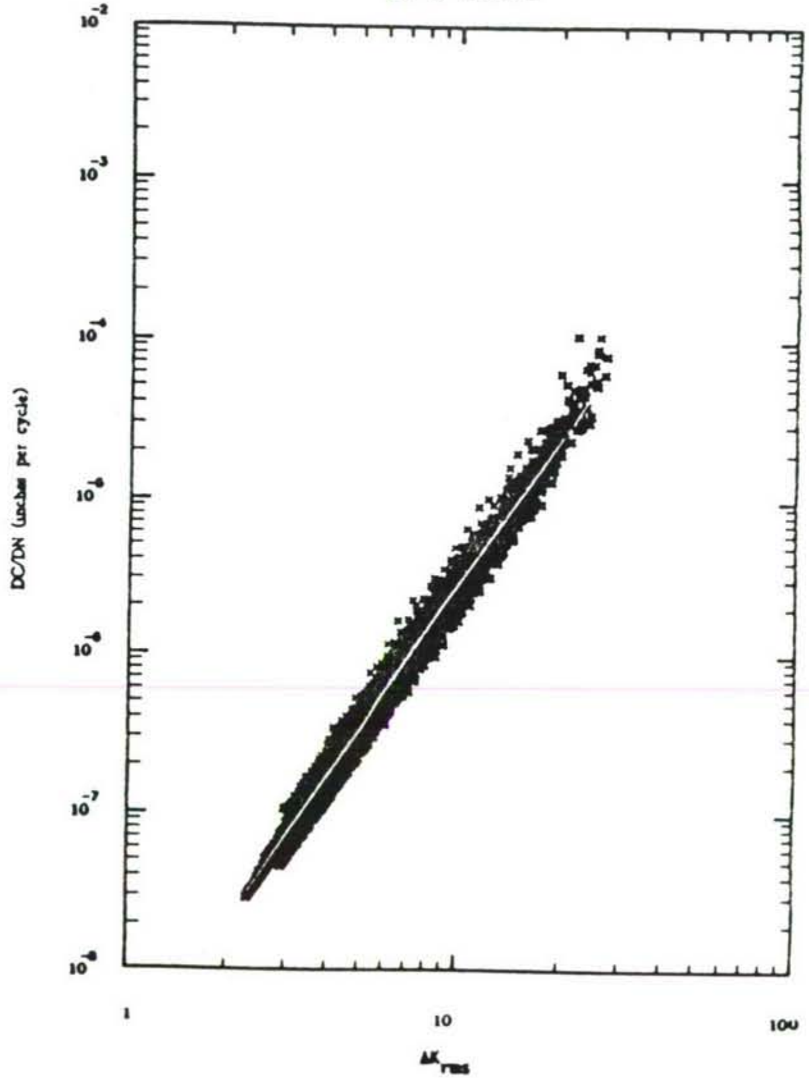
$$\frac{dc}{dN} = \frac{\Delta c}{\Delta B} \cdot \text{DIVIDED BY} \quad \frac{\text{XXX CYCLES}}{\text{BLOCK}}$$

F-16 FxF FATIGUE CRACK GROWTH RATE MASTER CURVES FOR 7475-T7351 ALUMINUM

HIGH HUMID AIR



JP4 FUEL



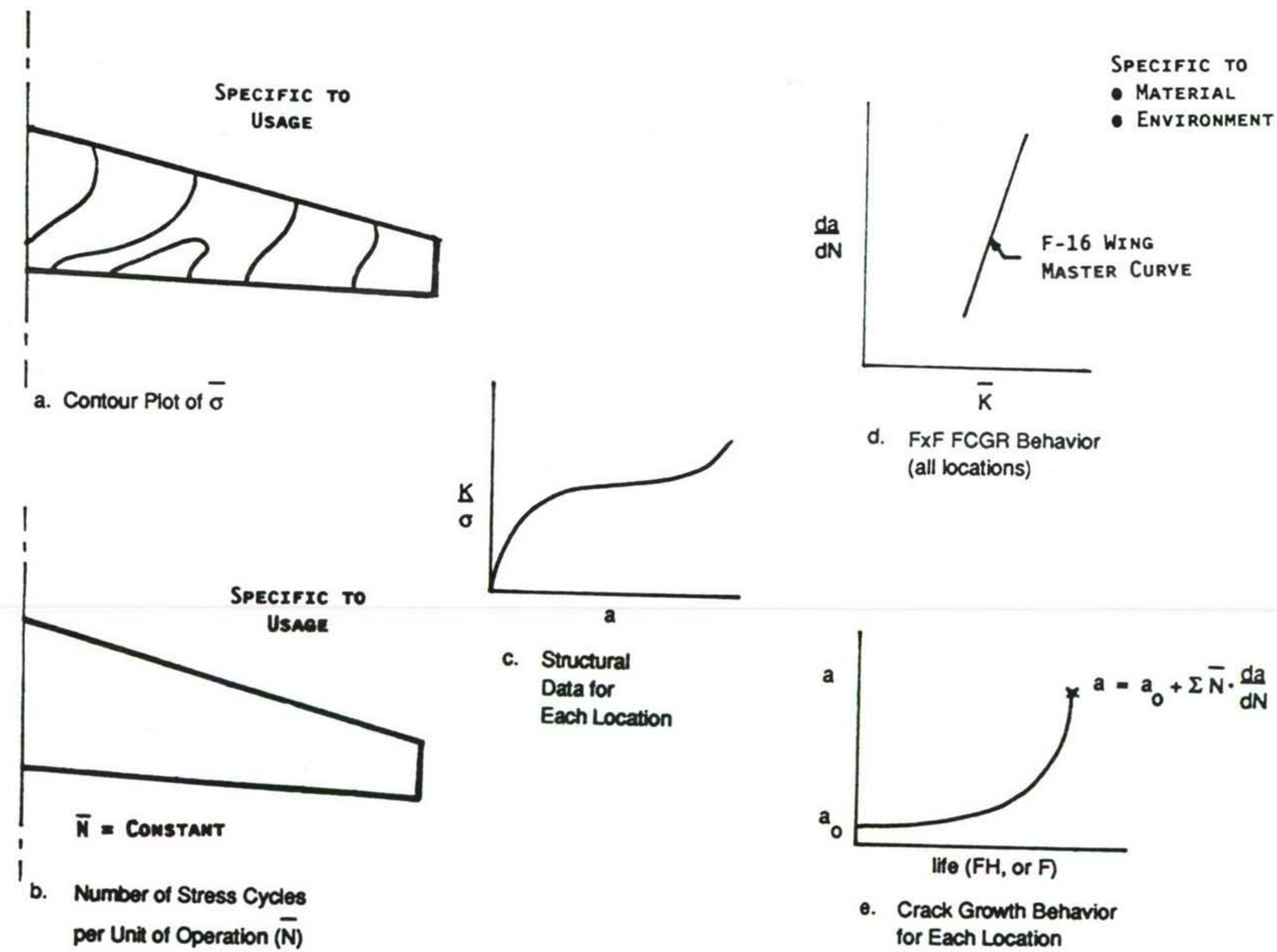
SUMMARY OF OBSERVATIONS

- SIX MASTER CURVES WERE DEVELOPED TO COVER ALL MATERIAL AND ENVIRONMENTAL COMBINATIONS

- ALL 20 WING STRESS HISTORIES AND EIGHT (OF THE 12) FUSELAGE HISTORIES RESULTED IN BEHAVIOR THAT FELL WITHIN EACH MASTER CURVE'S SCATTER BAND

- TAIL HISTORIES RESULTED IN FxF BEHAVIOR THAT WAS UNIQUELY DIFFERENT, PROBABLY BECAUSE OF NEGATIVE STRESS RATIO EFFECTS AND THE METHODOLOGY USED TO MODEL THESE EFFECTS

RAPID CRACK GROWTH LIFE CALCULATIONS FOR MULTIPLE LOCATIONS IN THE F-16 LOWER WING



COMPONENT CRACK GROWTH ANALYSIS REQUIREMENTS

CYCLE-BY-CYCLE (CxC)

- STRESS INTENSITY FACTOR ANALYSIS
- CxC STRESS HISTORY
- MATERIAL PROPERTIES

  (C.A. $\frac{da}{dN}$, Retardation, ...)
- COMPLICATED CRACK GROWTH LIFE CODE

FLIGHT-BY-FLIGHT (FxF)

- STRESS INTENSITY FACTOR ANALYSIS
- FxF STRESS HISTORY
- MATERIAL PROPERTIES

  (FxF $\frac{da}{dN}$, ...)
- SIMPLE CRACK GROWTH LIFE CODE

FLIGHT-BY-FLIGHT
COMPONENT CRACK GROWTH ANALYSIS
ADVANTAGES

- ELIMINATES THE NEED FOR PRE-EXISTING CxC STRESS HISTORIES AT ALL LOCATIONS IN STRUCTURE
- SIMPLIFIES THE LIFE ANALYSIS PROCESS
- COMPUTATIONALLY MORE EFFICIENT
  - LESS CPU
  - QUICKER TURNAROUND
- PRODUCES SAME LEVEL OF ACCURACY AS CxC ANALYSIS

SUMMARY

- A RAPID DTA ANALYSIS PROCEDURE HAS BEEN DEVELOPED TO PROVIDE MAINTENANCE ENGINEERS WITH THE TECHNOLOGY FOR ESTIMATING CRACK GROWTH IN REPAIRED STRUCTURES.

- THE PROCEDURE MINIMIZES COMPUTATIONAL DIFFICULTIES ASSOCIATED WITH CONDUCTING DETAILED CYCLE-BY-CYCLE CRACK GROWTH LIFE ESTIMATES.

NLR

Subject to the conditions as stated on the attached NLR form.

RNLAF EXPERIENCES WITH EPAF-ASIP FOR F-16 AIRCRAFT

PRESENTED AT

1989 USAF Structural Integrity Program Conference
5 - 7 December 1989
San Antonio, Texas

by

D.J. Spiekhout

NATIONAL AEROSPACE LABORATORY, NLR
Amsterdam, The Netherlands

NLR

CONTENTS BRIEFING

1. BACKGROUND EPAF-ASIP

2. OPERATIONAL STRAIN MEASUREMENTS
   (single and multi channel)

3. CRACK GROWTH INVESTIGATION AT NLR
   (calculations and specimen tests)

4. CRACK SEVERITY INDEX CONCEPT

5. CONSEQUENCES FOR MAINTENANCE RNLAF
   (load monitoring)

1. BACKGROUND EPAF-ASIP

- 1979, introduction F-16 A/B in RNLAF
- 1984, start of EPAF-ASIP program
- carried out by General Dynamics
- for 4 European Air Forces (EPAF)
- use of USAF-ASIP results
- input from Flight Load Recorder (FLR) and Mechanical Strain Recorder (MSR) results
- only one Fleet Structural Maintenance Plan (FSMP) per air force

NLR INVOLVEMENT:

- participate in EPAF-ASIP meetings
- prepare 'input RNLAF' for EPAF-ASIP
    * mission mixture
    * fuel/store configuration per mission type
    * TO weight distribution
    * mission content (severity)

  sources:

    * additional FLR measurements
    * NLR strain measurements with "Spectrapot"
    * squadron administration books

- discussion with GD about use of 'RNLAF input' and subsequent analysis

2. OPERATIONAL STRAIN MEASUREMENTS BY NLR

SINGLE CHANNEL STRAIN MEASUREMENTS F-16

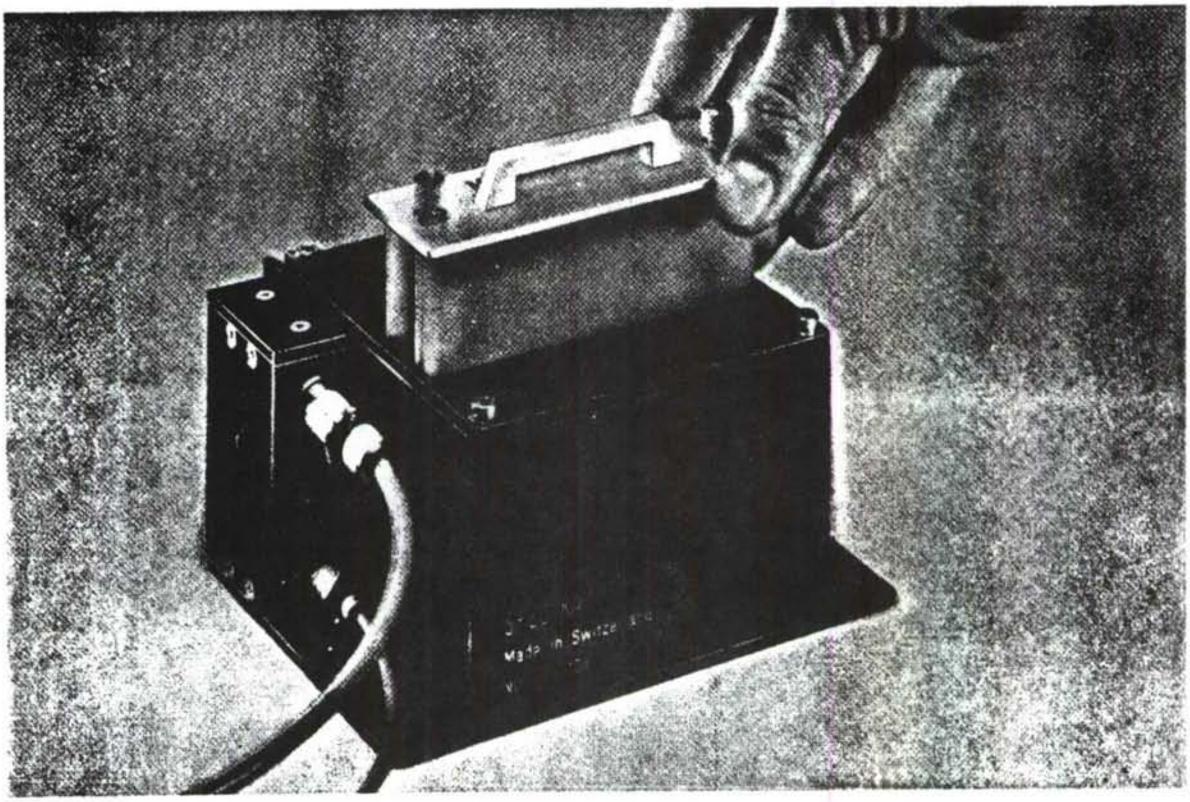
PURPOSE OF PROGRAM:

* input for EPAF-ASIP
* monitoring of changes in usage and loading experience during operational flights of the RNLAF

IN MEASURING PROGRAM:

* measuring with strain gage bridge at MSR location (calibrated with MSR, converted to GD calculated "MSR strain")
* "Spectrapot" instrumentation (storing peaks/valleys on solid state memory)
* since 1984 at all F-16 air bases (started as a limited load monitoring program)
* load data and debriefing form data per flight available in data base (about 2000 flights per mid 1989)

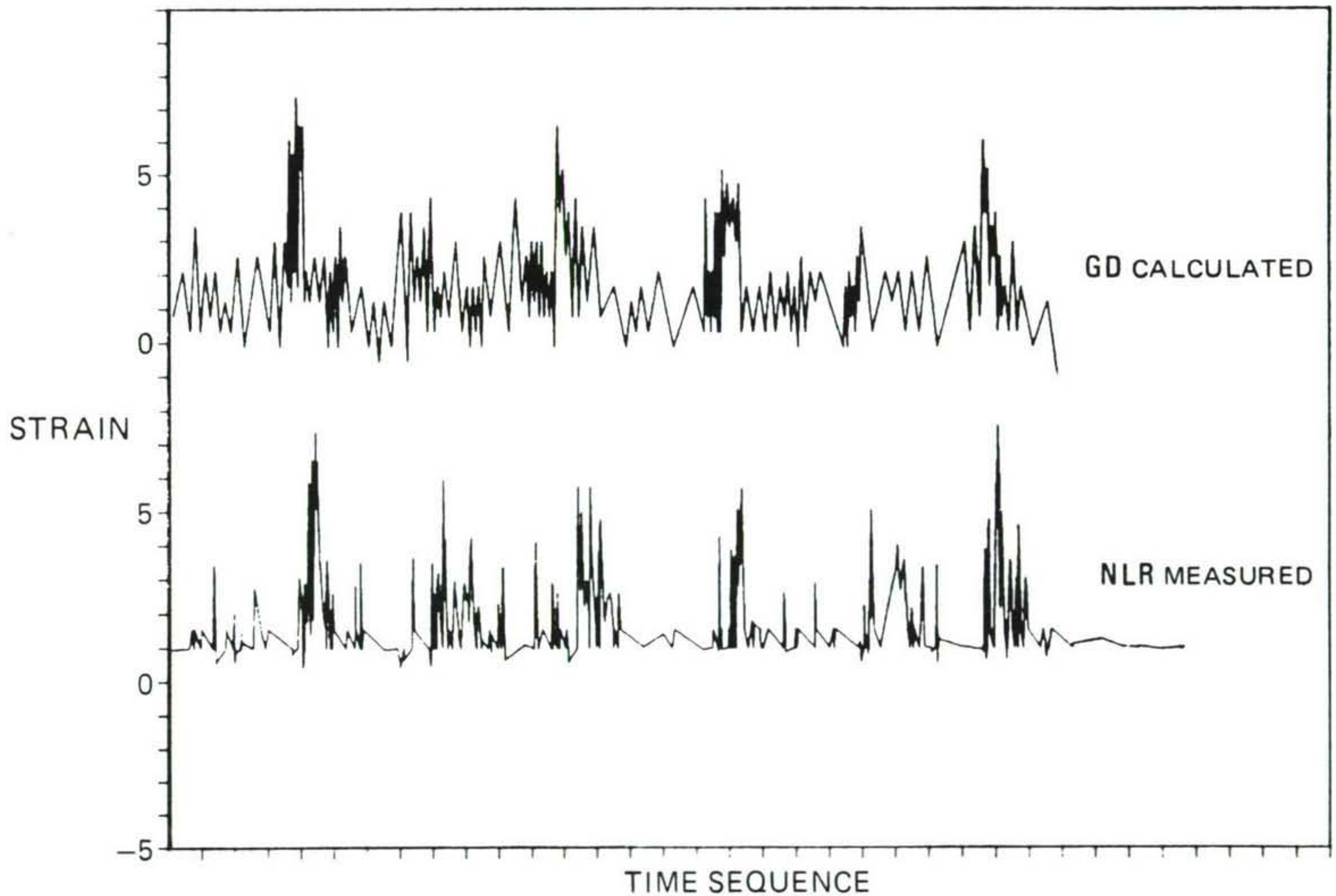
● SPECTRAPOT DATA COLLECTOR



- 1 CHANNEL (32 STEPS)
- SIGNAL CONDITIONING/POWER SUPPLY BRIDGE
- DATA REDUCTION IN FLIGHT:
  - PEAK/TROUGH SEARCH
  - RANGE FILTER
- TIME MARK
- SPV/RTRF
- SOLID STATE MEMORY

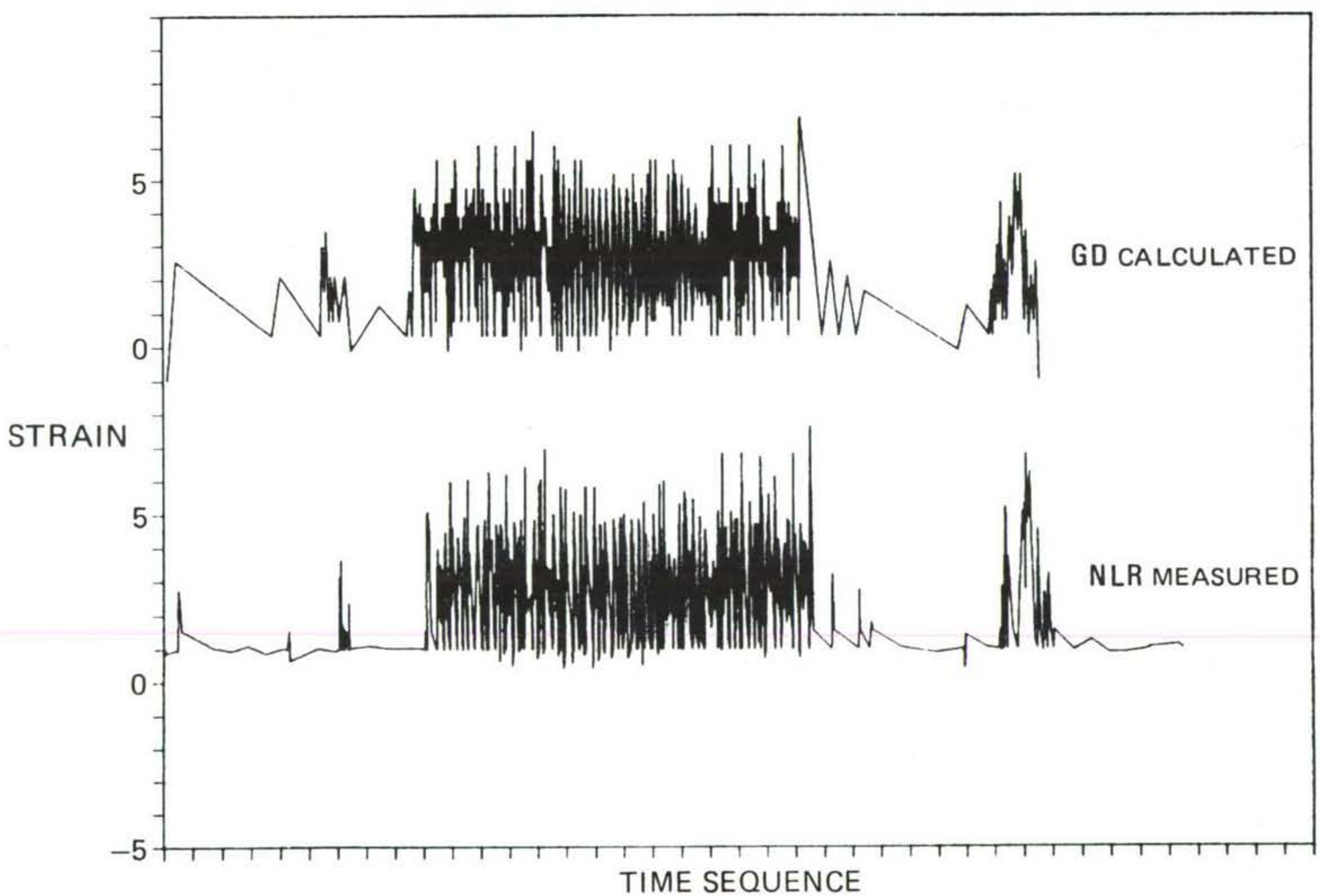
EXAMPLE OF GD CALCULATED AND NLR MEASURED STRAIN AT MSR LOCATION

ACT MISSION TYPE



EXAMPLE OF GD CALCULATED AND NLR MEASURED STRAIN AT MSR LOCATION

RANGE MISSION TYPE



F-16 WING STRAIN MONITORING

12-03-1985

GROUND PERSONNEL

SQUADRON: 322 ☐ 306 ☐
323 ☐ 311 ☐
313 ☐ 312 ☐
315 ☐ ...... ☐

LOCATION: LEEUWARDEN ☐
VOLKEL ☐
DE PEEL ☐
TWENTHE ☐

TYPE OF AIRCRAFT: F16–A ☐
F16–B ☐
RF–16 ☐
.... ☐

DATE ☐ MEMORY NR ☐ AIRCRAFT NR ☐ TOTAL HOURS BEFORE FLIGHT ☐

EXTERNAL STORES CONFIGURATION BEFORE FLIGHT

WINGTIP	1	9	OUTBOARD	2	8	OUTBOARD	3	7	INBOARD	4	6	CENTERLINE	5
launcher			empty			empty			empty			empty	
aim 9			u.w launcher			pylon only			pylon only			rack/adaptor	
....			aim 9			ter			dry 370			dry 300	
....			....			suu 20			1/2 full 370			full 300	
....			....			ver			full 370			ecm pod	
....			....			....			....			recce pod	
												....	

PILOT

FLIGHT DURATION ☐ T/O TIME ☐ MAX. HUD G–LEVEL ☐

BASIC MISSION	

if RANGE, AGNAV, AGCOM →

	NUMBER OF PASSES			
	LEVEL	DIVE ANGLE		LOFT
		$\leq 15^0$	$> 15^0$	
GUNNERY				
ROCKETRY				
BOMBING				

if ACT, PI OR QRA → NUMBER OF SETUPS ☐

if DART → NUMBER OF PASSES ☐

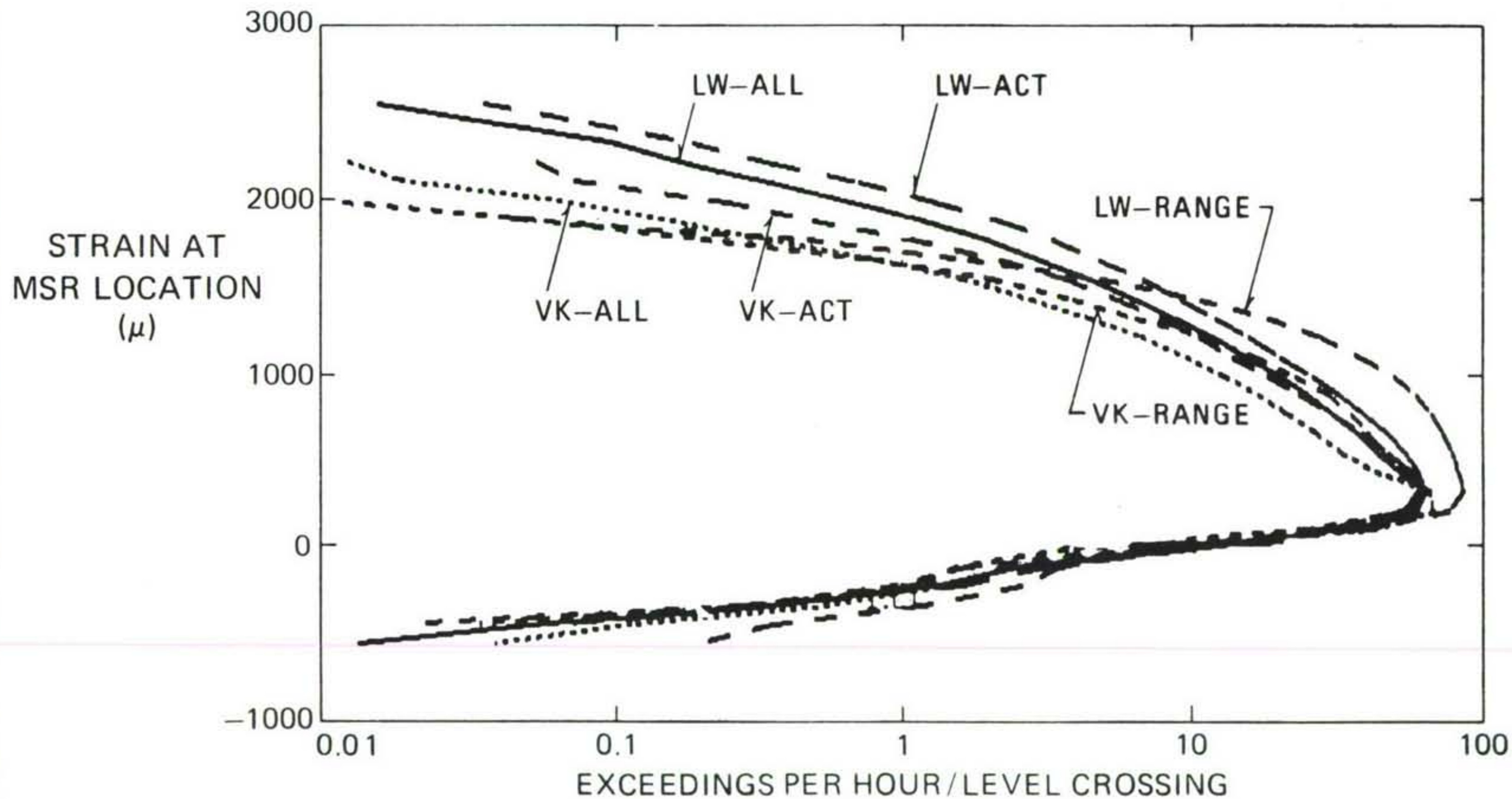
if ADDITIONS TO BASIC MISSION

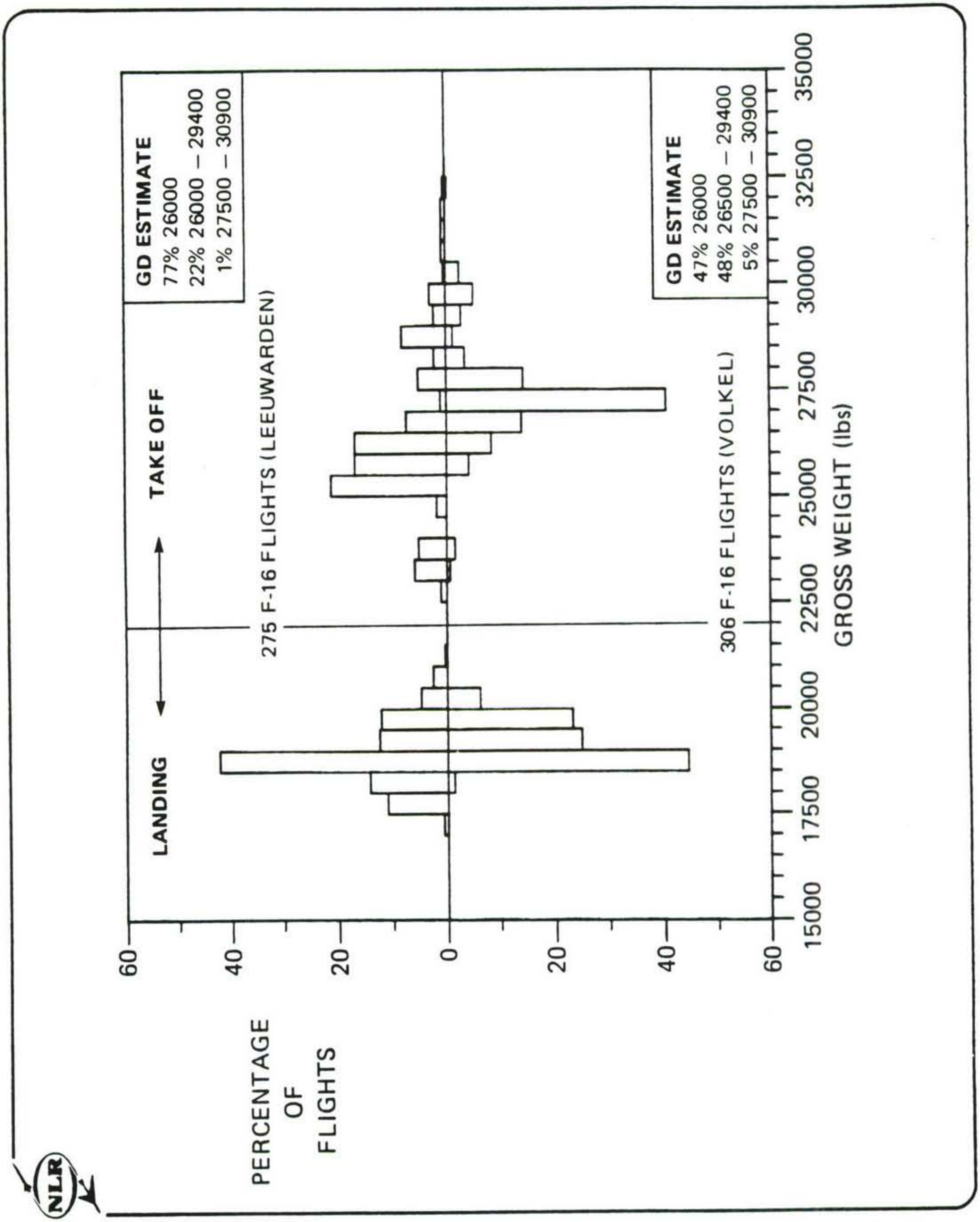
ACT ☐ PI ☐ .... ☐

MISSION EVENTS										

REMARKS: ....

STRAIN SPECTRA PER MISSION TYPE FOR LW AND VK AIR BASE




NLR
LANDING
TAKE OFF
275 F-16 FLIGHTS (LEEUWARDEN)
306 F-16 FLIGHTS (VOLKEL)
GD ESTIMATE
77% 26000
22% 26000 – 29400
1% 27500 – 30900
GD ESTIMATE
47% 26000
48% 26500 – 29400
5% 27500 – 30900
PERCENTAGE
OF
FLIGHTS
60
40
20
0
20
40
60
15000
17500
20000
22500
25000
27500
30000
32500
35000
GROSS WEIGHT (lbs)

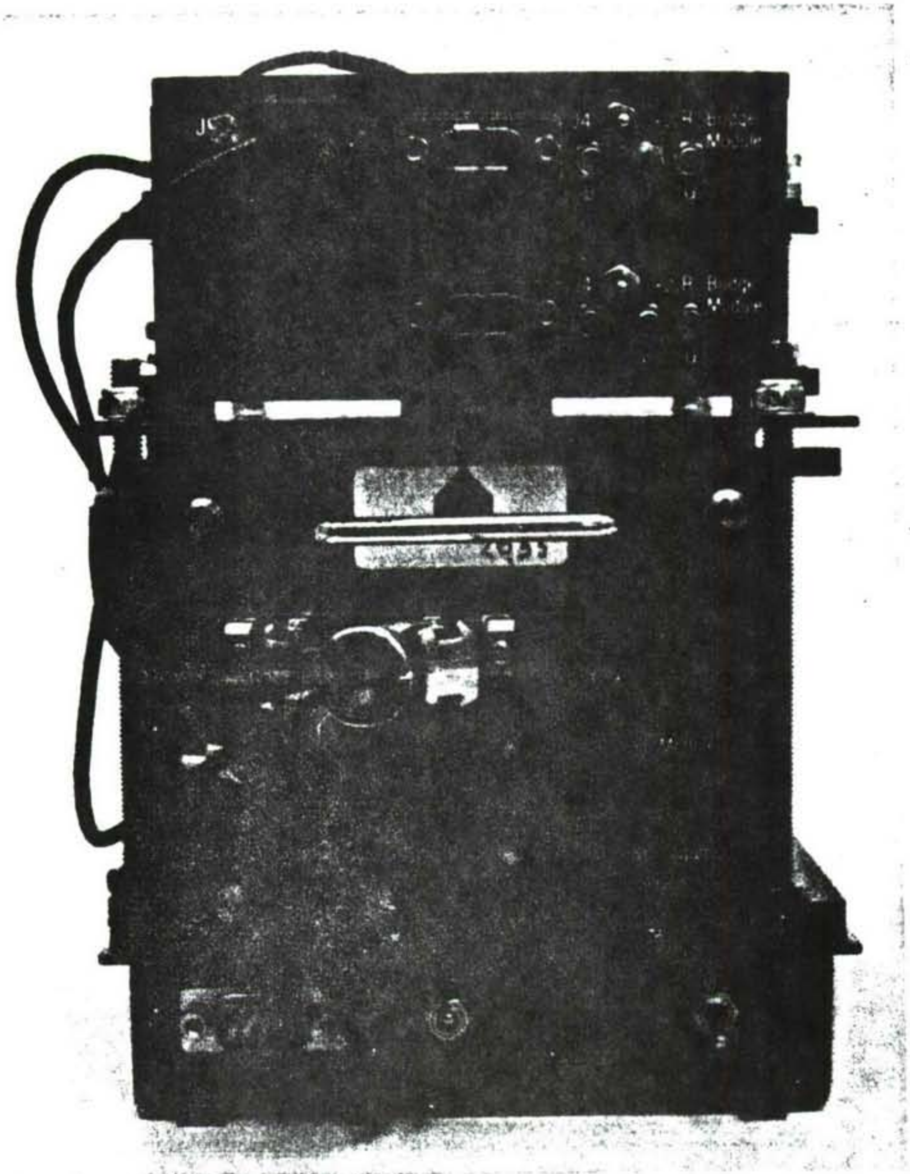
MULTI CHANNEL STRAIN MEASUREMENTS F-16

PURPOSE OF PROGRAM:

* **to find a correlation between the severity at the MSR location and the severity at another point of the structure**
* **to see if there is an influence of mission type**

IN MEASURING PROGRAM:

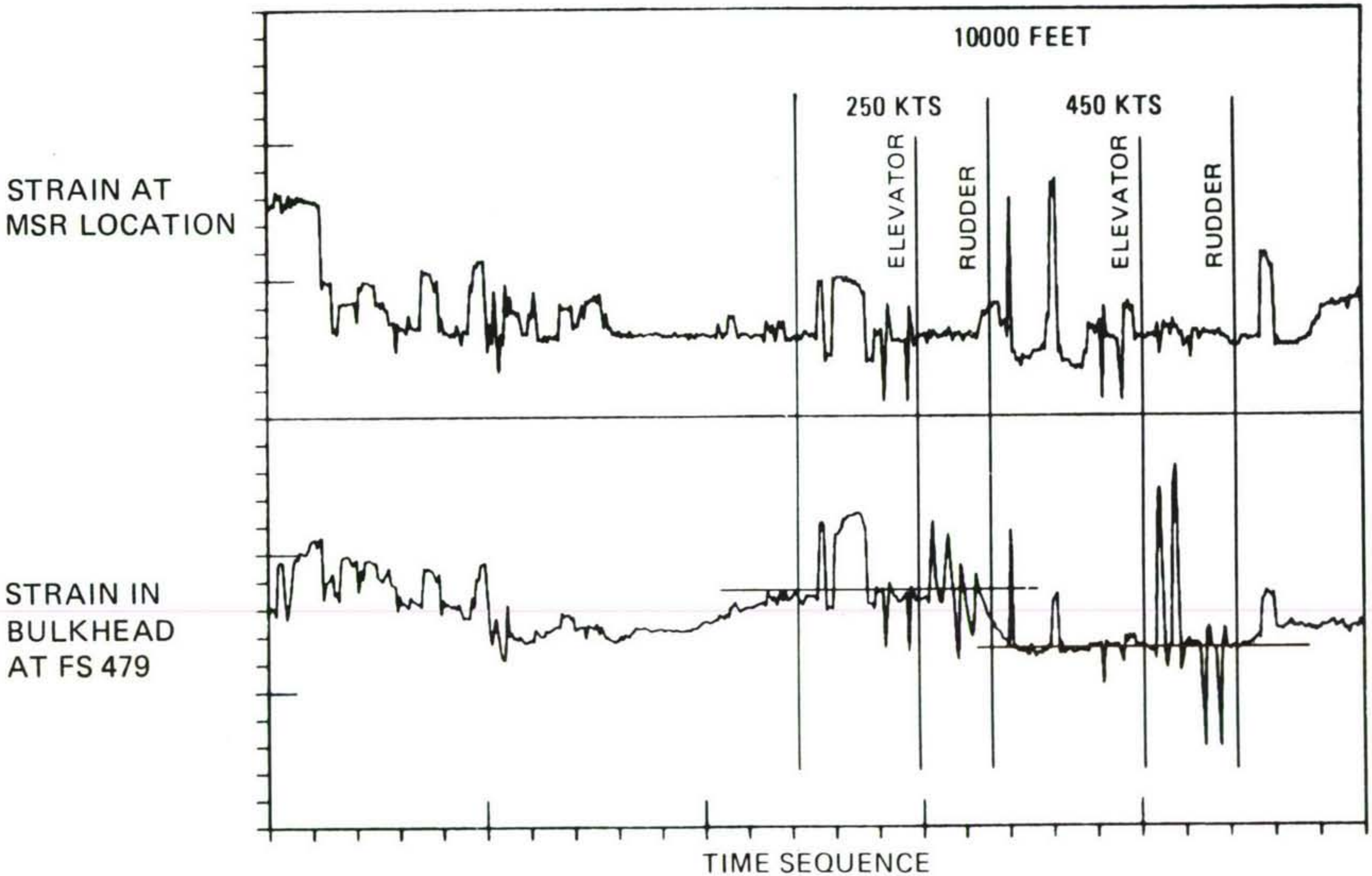
* **measuring:**
  . strain gage at MSR location
  . strain gage from FLR (aft fuselage, FS 479)
  . vertical acceleration
* **instrumentation: Spectrapot-4C data collector**
  (data reduction to peaks and valleys)
* **at Leeuwarden AFB with 3 aircraft**
* **per flight load data and debriefing form available**



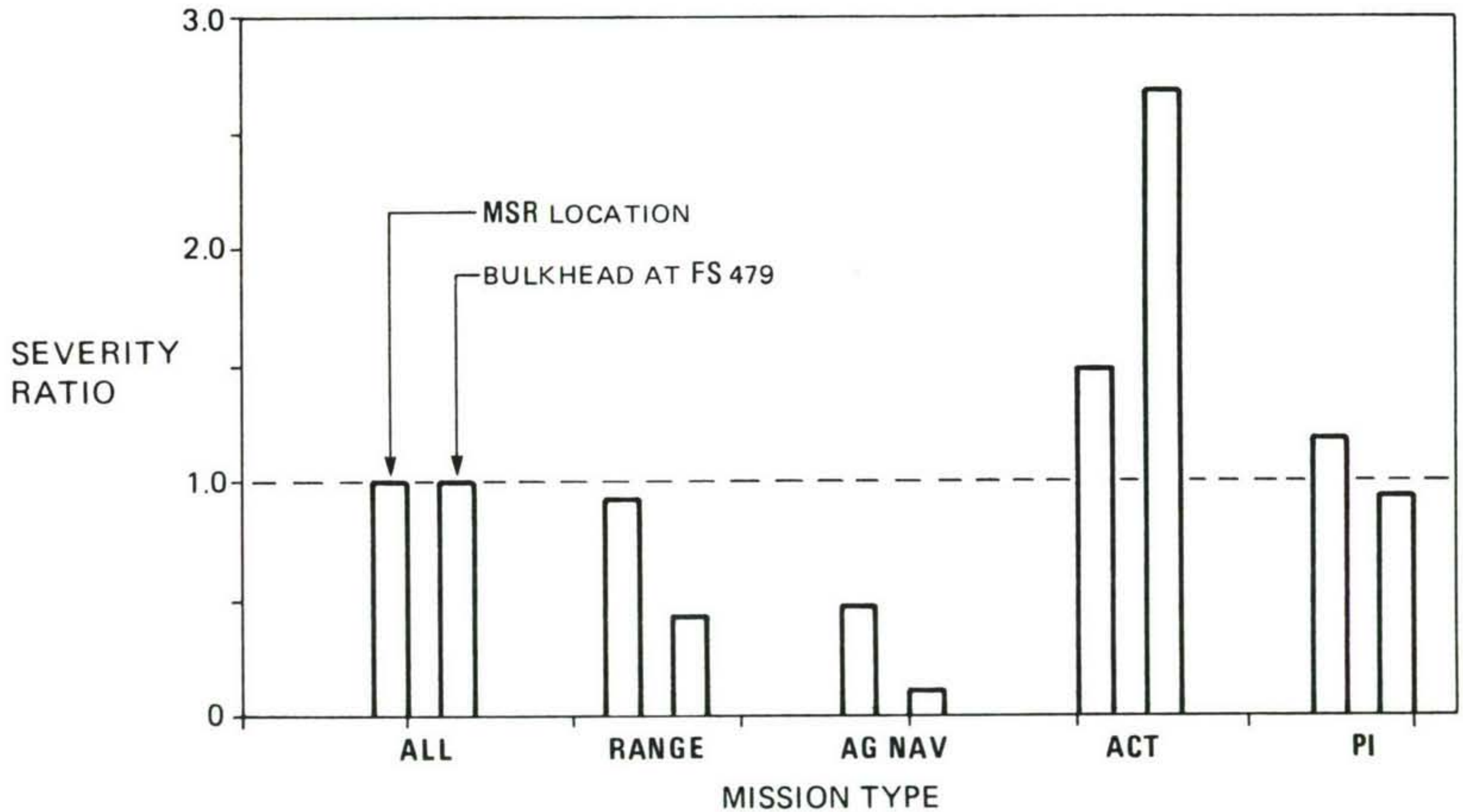
**SPECTRAPOT – 4C DATA COLLECTOR
USED FOR
RNLAF LOAD MONITORING**

3 CHANNELS: "**MSR**" STRAIN
"**FS-479**" STRAIN
VERTICAL ACCELERATION

EXAMPLE OF ELEVATOR/RUDDER INPUT ON MEASURED STRAINS
IN MULTI CHANNEL MEASURING PROGRAMME



RELATIVE SEVERITIES OF DIFFERENT MISSION TYPES



REMARKS ON MEASURING PROGRAMS

* SINGLE CHANNEL STRAIN MEASUREMENTS
  large differences in usage and loading environment between air bases
  -> FSMP per air base necessary

* MULTI CHANNEL STRAIN MEASUREMENTS
  the relative damage per mission type is not the same for the two considered structural locations
  (MSR at FS 325,8, FLR strain gage at FS 479,5)

* FLR AND MSR RECORDINGS
  - administrative data FLR not usable
    (mission type, weight information)
  - applied counting methods questionable
  - use of EPAF-FLR results preferred to USAF-FLR

➔ EPAF-FSMP ONLY VALID FOR USAGE AND LOADING ENVIRONMENT AS DETERMINED FOR EPAF-ASIP PER AIR BASE
(based on 1985 results)

➔ USE OF MSR FOR IAT MAY NOT BE ACCURATE ENOUGH FOR 'EXTREME' STRUCTURAL LOCATIONS.

3. CRACK GROWTH INVESTIGATION AT NLR

PURPOSE:

- **to advise RNLAF on 'quality' of 'parametric' approach results as applied by GD for EPAF-ASIP**
- **establish relative severity between LW and VK AFB**
- **establish relative severity between GD sequences for more 'Control Points' (CP)**

STEPS IN 'PARAMETRIC APPROACH' PROCEDURE BY GD

- **about 2450 flights in EPAF FLR data base**
- **for 8 EPAF common mission types**
  (4 mission types: light/heavy)
- **from FLR: parameter time histories**
  * load time histories
  * stress time histories
  (both corrected for fuel/store usage)
- **with USAF-ASIP results**
  crack growth curve per EPAF mission type per CP
- **'parametric approach' per air base**
  * EPAF mission type mixture
  * 'average MSR severity'

RESULT

average crack growth curve per control point and per air base

AVAILABLE LOAD SEQUENCES

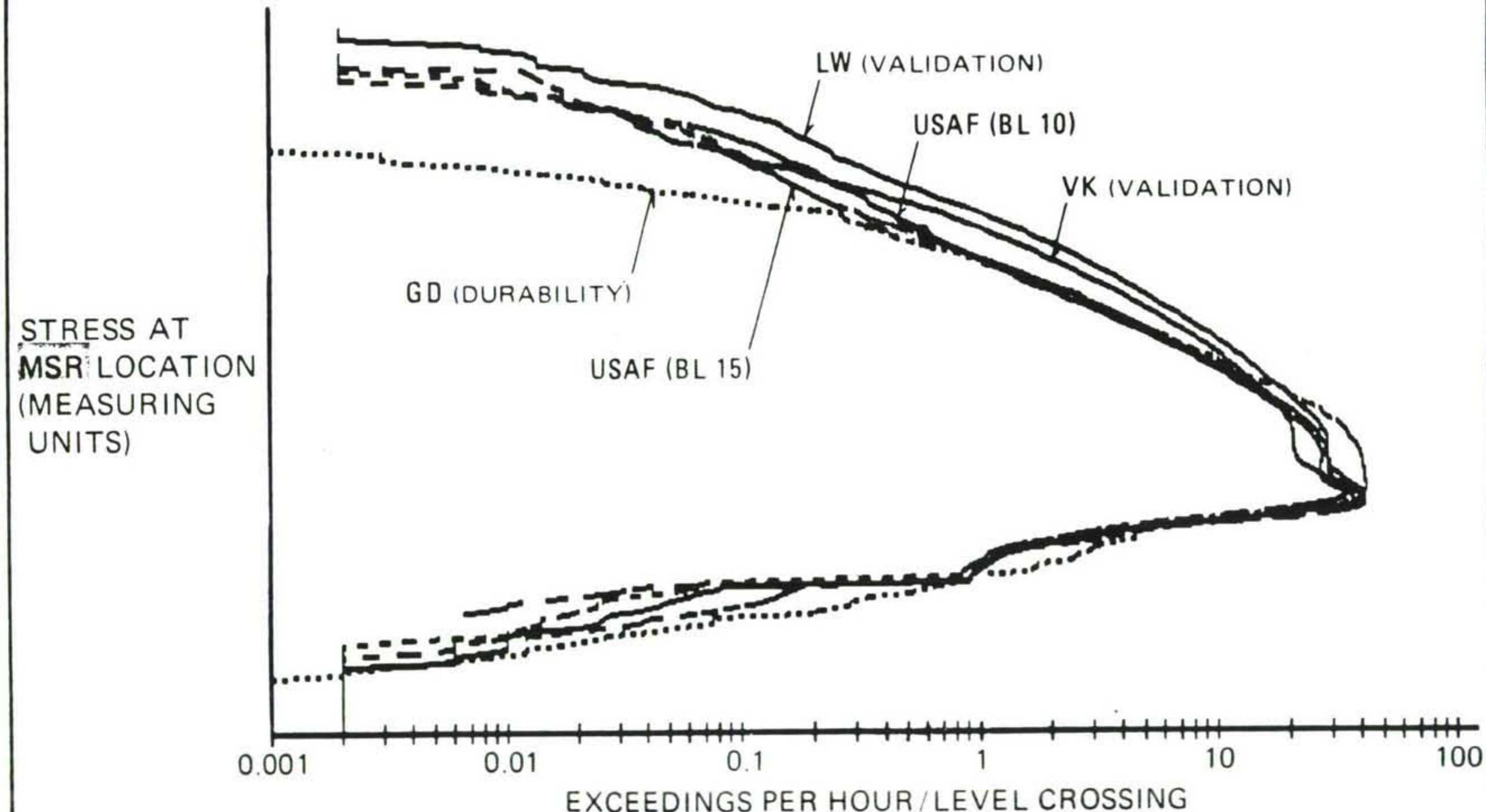
MADE BY GD (FROM FLR)

- **for 7 locations**
  * MSR location
  * wing (root, BL 102)
  * horizontal tail (small, big)
  * fuselage (FS 189, FS 479)
- **for 8 mission mixtures**
  * validation sequences
    . RNLAF (LW, VK AFB)
    . RNoAF (BD AFB)
    . EPAF
  * A/B durability test (16bl, 2bl) at GD
  * USAF (BL10, BL15)
  * C/D durability test at GD

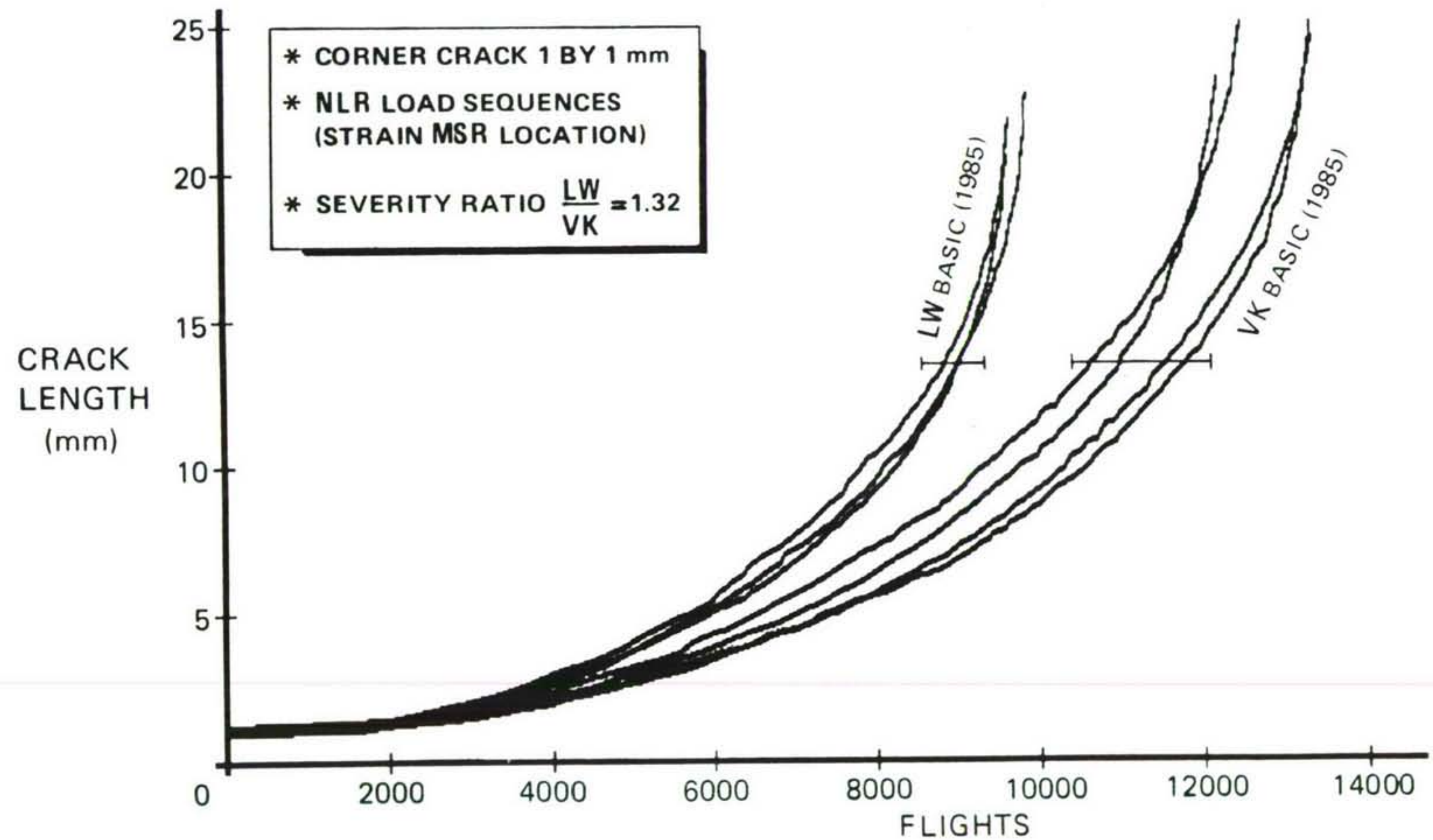
MADE BY NLR (FROM SPECTRAPOT)

- **LW BASIC (6 variations)**
- **VK BASIC**

COMPARISON OF STRESS SPECTRA FROM GD CALCULATED SEQUENCES

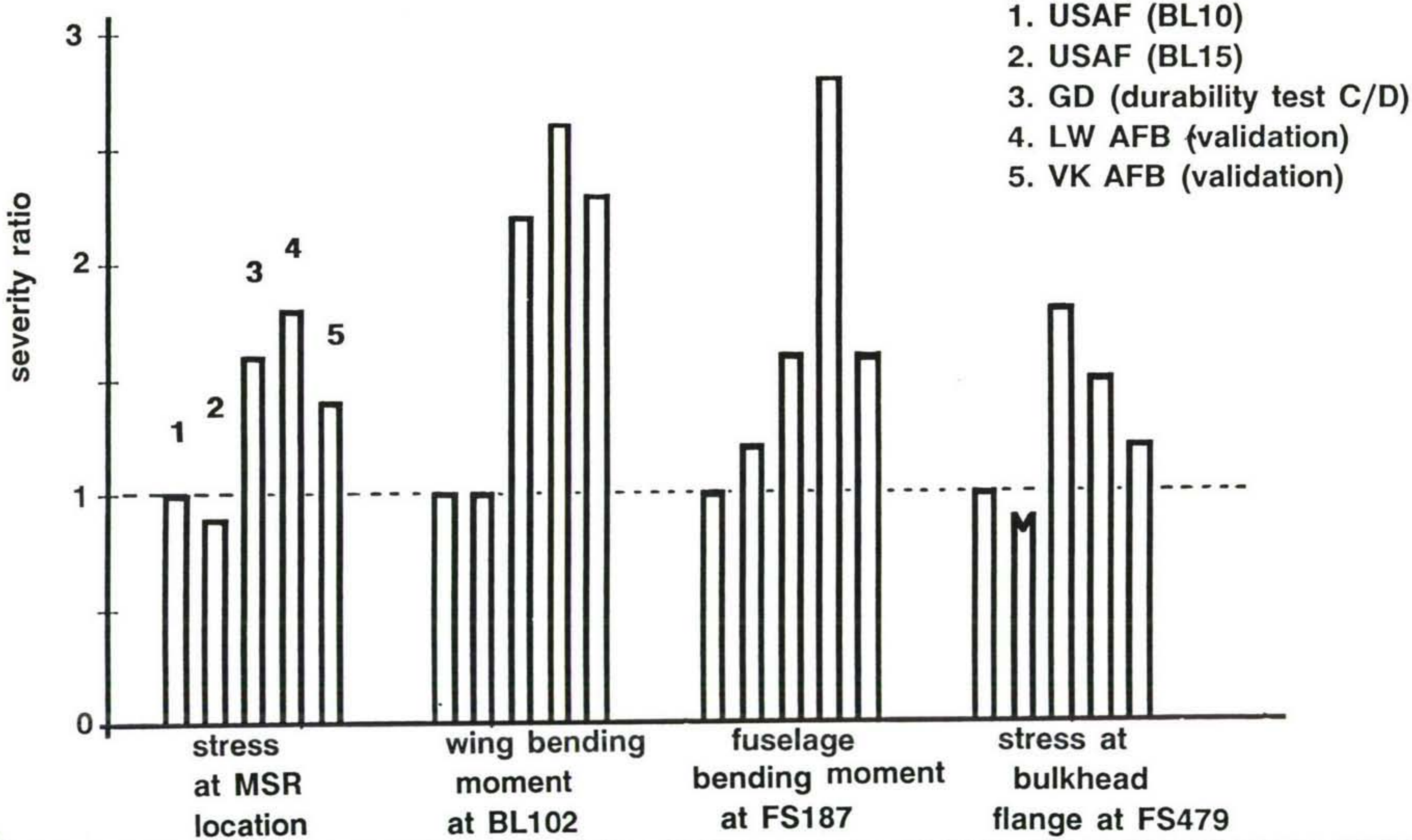


CRACK GROWTH TEST RESULTS FOR SEVERITY RATIO LW AND VK AIR BASE



NLR CRACK GROWTH CALCULATION RESULTS FOR DIFFERENT USAGES

* based on GD load sequences



CRACK GROWTH CALCULATION RESULTS FOR ACTUAL RNLAF OPERATIONAL USAGE
(based on GD parametric approach)

LOCATION	USAF BLOCK 10	LW AFB	VK AFB	SEVERITY RATIO
WING BL 50	1.0	5.5	4.3	1.3
WING BL 71		2.9	2.3	1.3
WING BL 102		5.4	4.5	1.2
FUSELAGE FS158		1.9	1.2	1.6
FUSELAGE FS 479	1.0	2.7	2.0	1.4

* based on flight hours

* usaf block 10 is reference for severity ratio

REMARKS ON CRACK GROWTH INVESTIGATIONS

- 'parametric approach' procedure acceptable for RNLAF

- 'parametric approach' procedure gives conservative results for most validated Control Points

- in general GD crack growth results more conservative than NLR results

- RNLAF usage and loading environment is more severe than:
  * A/B durability test at GD
  * C/D durability test at GD
  * operational USAF F-16 A/B
  * other EPAF countries

4. CRACK SEVERITY INDEX (CSI) CONCEPT

- in 1986 developed by NLR for quantification of recorded stress spectra in terms of crack growth potential

- crack closure and associated crack growth retardation are taken into account

- the CSI is a relative damage figure for a specific structural location and loading spectrum

- GD has adopted method for use in EPAF-ASIP

CSI VALIDATION WITH CRACK GROWTH TESTS

CSI applied on wing root bending moment spectra from Spectrapot measurements (MSR location)
(in this case: CSI = 1.0 for a batch of 488 F-16 flights flown in 1985)

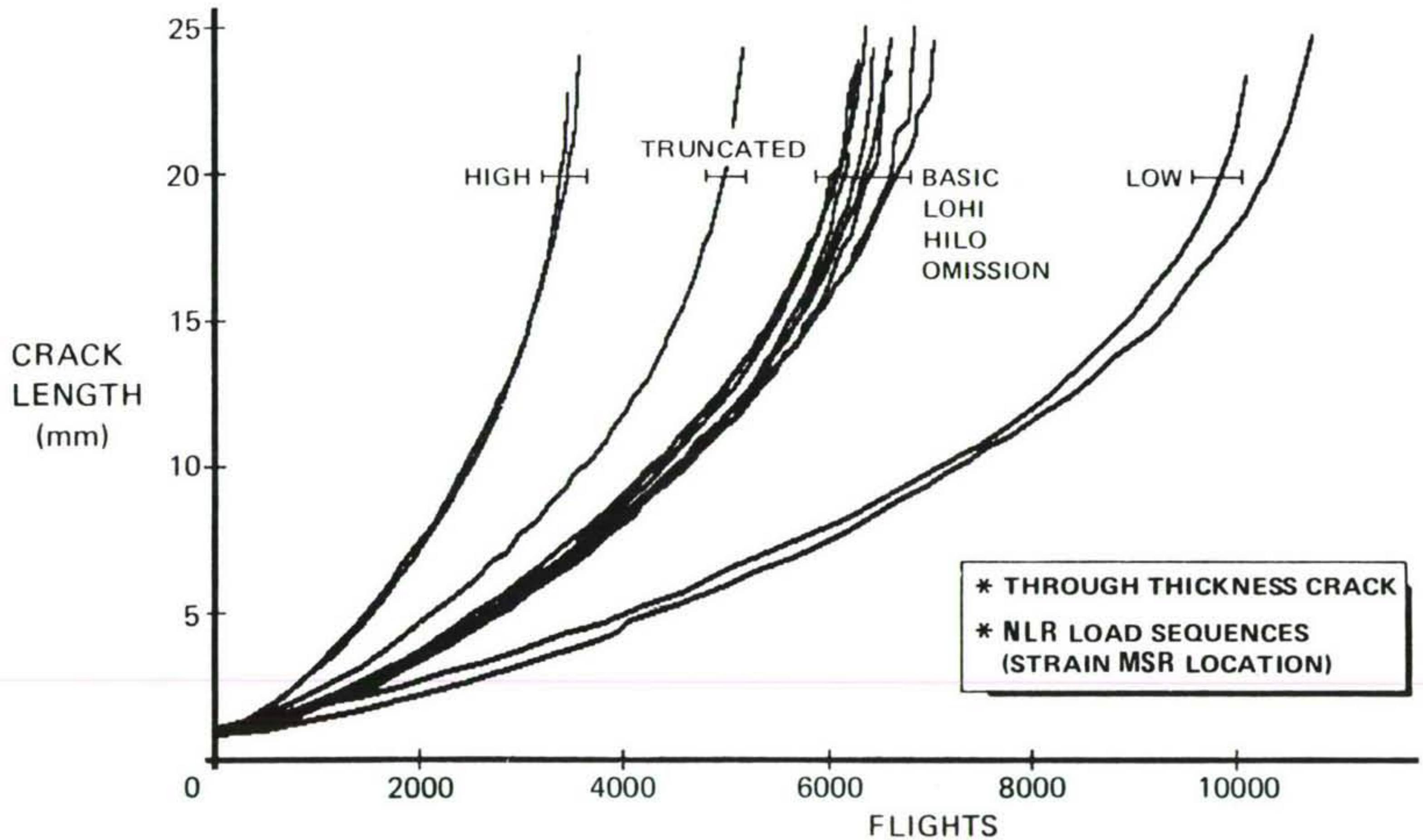
TEST SEQUENCES:

- BASIC	**200 flights LW air force base** mission mixture 1985 CSI (per hour) = 1.289

VARIATIONS:

- HIGH	**CSI = 2 * CSI (basic)** (flights with CSI < 1.24 deleted)
- LOW	**CSI = 0.5 * CSI (basic)** (flights with CSI > 1.73 deleted)
- LO HI	**CSI = CSI (basic)** (increasing CSI per flight)
- HI LO	**CSI = CSI (basic)** (decreasing CSI per flight)
- OMISSION	**CSI ± CSI (basic)** (cycles below opening stress level deleted)
- TRUNCATED	**CSI ± CSI (basic)** (truncation of a few high peaks)

CRACK GROWTH TEST RESULTS FOR CSI VALIDATION



5. CONSEQUENCES FOR MAINTENANCE RNLAF

- **many maintenance areas have shortage of economic service life and/or an initial inspection period shorter than expected**

- **a number of maintenance areas are very difficult to inspect**
  * very small critical crack size (< 0.25 inch)
  * very short repeat inspection period

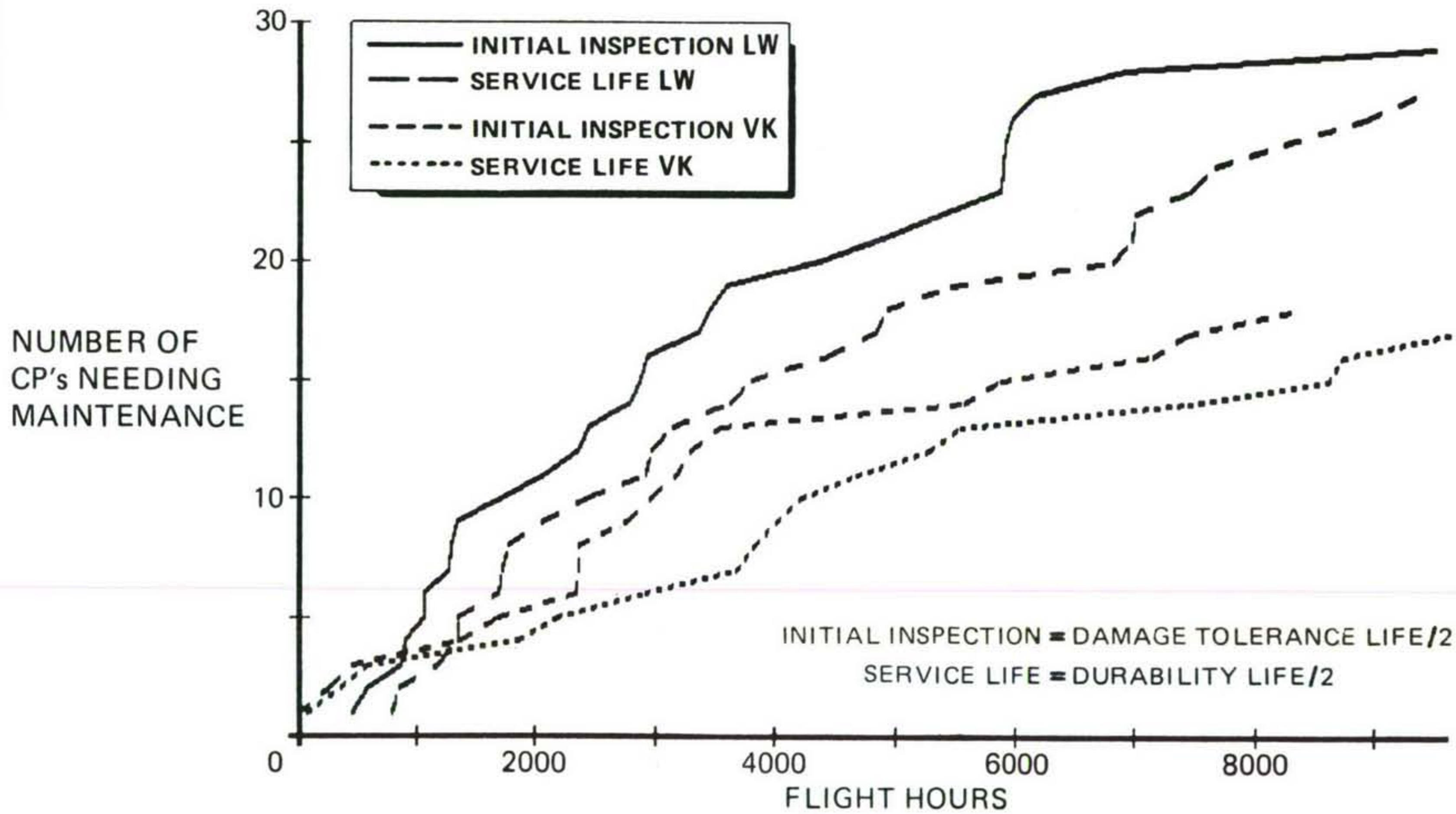
- **implementation of EPAF-FSMP in normal maintenance procedures**

LARGE INCREASE MAINTENANCE BURDEN FOR RNLAF

- **manhours**
  (inspection, reparation, modification, replacement)
- **downtime aircraft**
- **spares**

➔ **For many structural parts a modification is necessary to prevent repeat inspections**
(cold working holes in bulkheads)

OVERVIEW OF ADDITIONAL MAINTENANCE FOR RNLAF



NLR

FUTURE LOAD MONITORING AND AIRPLANE TRACKING FOR F-16

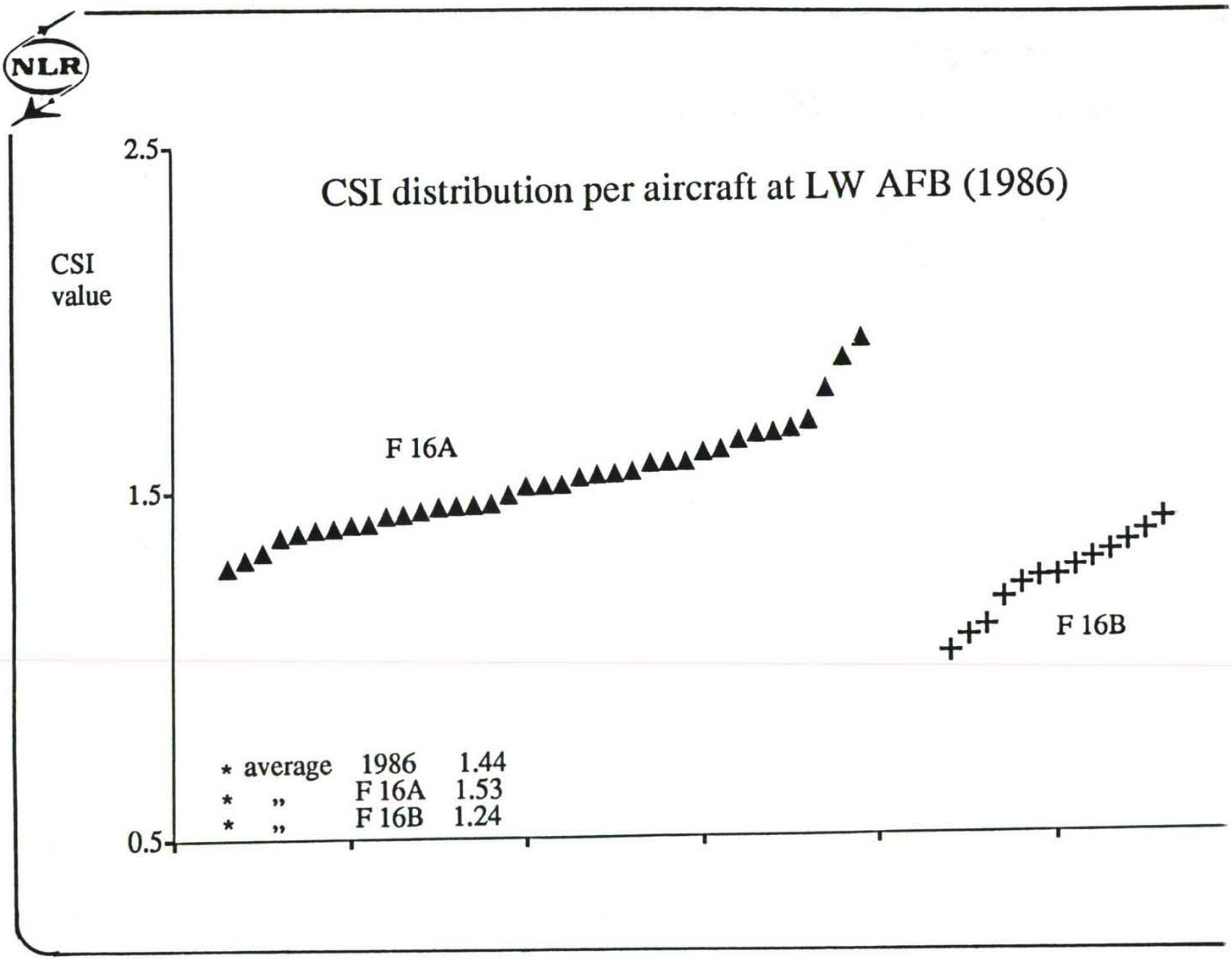
- **FLR and MSR not in use anymore by RNLAF**

- **NLR proposes a replacement by Spectrapot instrumentation. This means an extension of the present 'limited' program to 3 aircraft per squadron:**

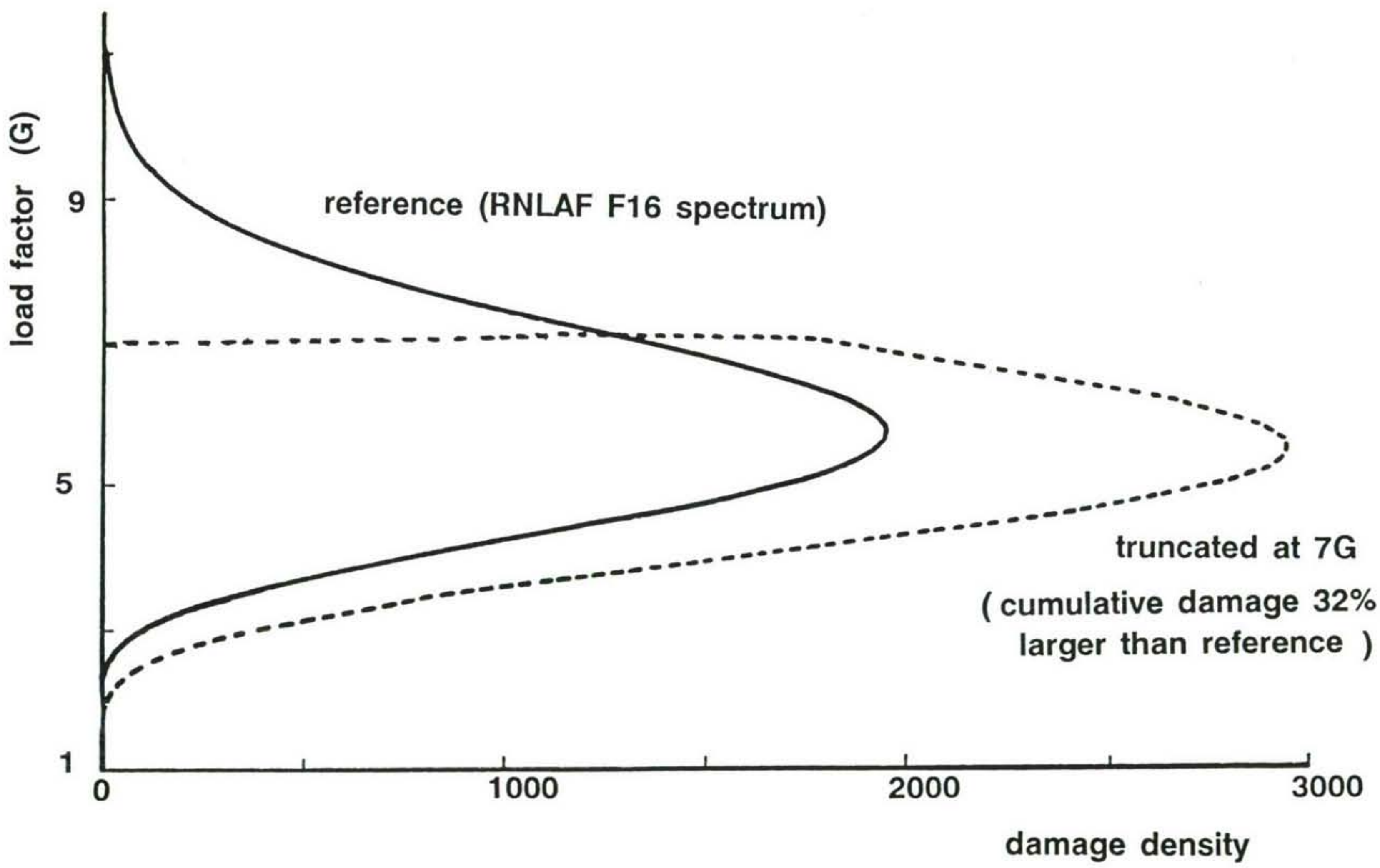
  (recording MSR strain sequences/debriefing form)

- **by keeping track of the average severity per mission type and the mission mixture per base/squadron/aircraft inspection procedures can be adapted**

- **future:**
  * introduction SFDR for load monitoring?
  * more cooperation with other EPAF air forces?
  * decrease of damaging flying by changing:
    . mission content
    . fuel/store configuration
    . less or less severe G loading


NLR
CSI distribution per aircraft at LW AFB (1986)
CSI value
2.5
1.5
0.5
F 16A
F 16B
* average 1986 1.44
* " F 16A 1.53
* " F 16B 1.24

COMPARISON OF DAMAGE DENSITIES FOR TWO SPECTRA (MSR location)



A Proposed MIL-STD
For USAF NDE System
Reliability Assessment

by
Charles Annis
Alan Berens
Peter Hovey
Sharon Vukelich
and others

MIL-STD for USAF NDE
System Reliability Assessment

This document is the result of the cooperative efforts of many people and organizations. The Air Force/Industry NDE Reliability Working Group Members were:

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Ward Rummel	Martin Marietta / Denver
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Bob Stone	San Antonio-ALC/MMFR
Sharon Vukelich	Aero Systems Division/YZEE

with special thanks to Becky Harrison, P&W, for typesetting the entire document.

Purpose

Provide testing and evaluation procedures for assessing and comparing NDE system capability for gas turbine engine component inspections.

Proposed MIL-STD Organization

1. Scope
2. Reference Documents
3. Definitions
4. General Requirements
5. Specific Requirements
6. Notes
7. Appendices

NDE Systems Discussed

- Eddy Current
- Fluorescent Penetrant
- Ultrasound
- Magnetic Particle

NDE Systems Classification

- "hit/miss" systems, which produce only qualitative information as to the presence or absence of a flaw.

- "a-hat vs. a" systems, which also provide some quantative measure of the size of the indicated flaw.

Figure C-1.

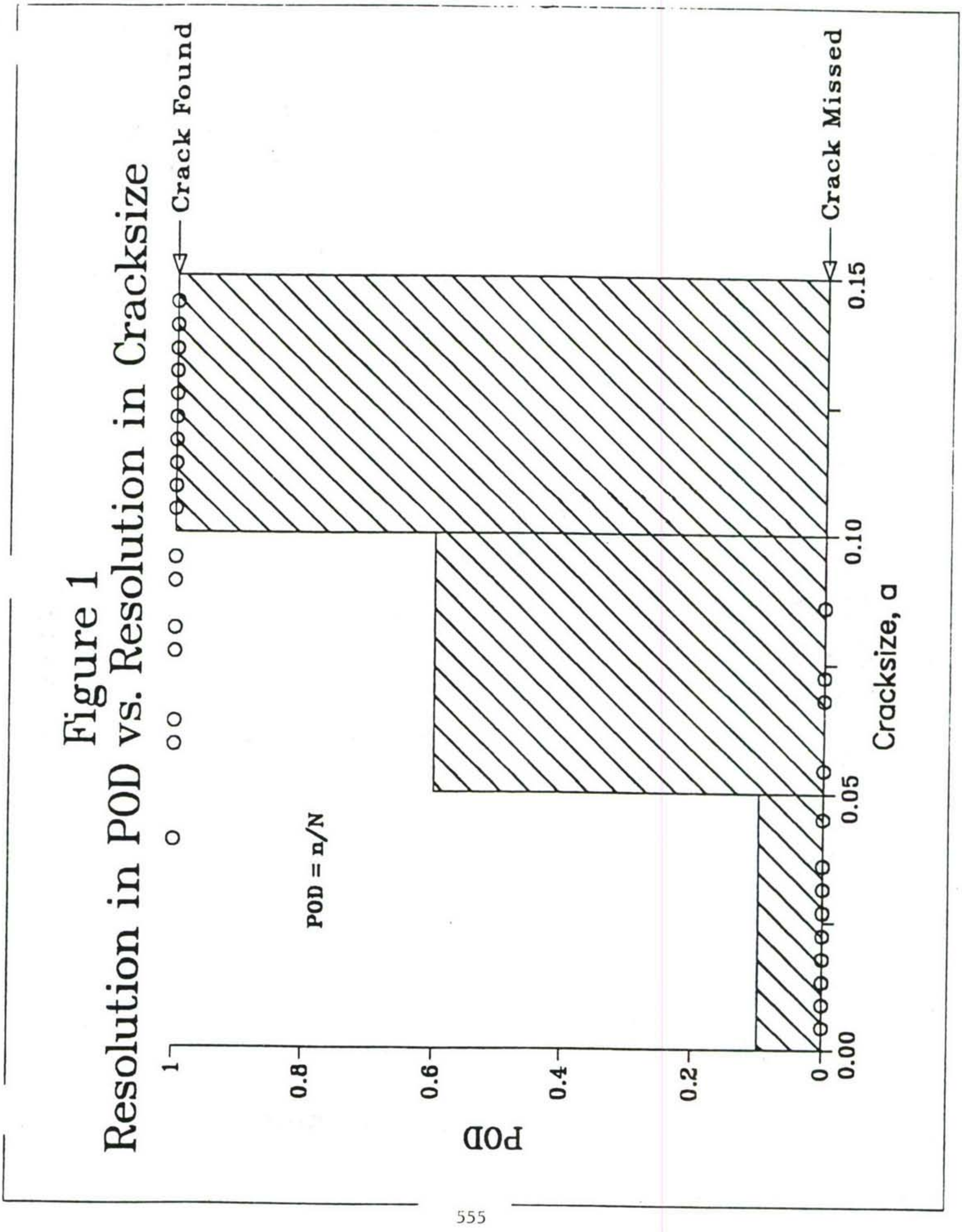
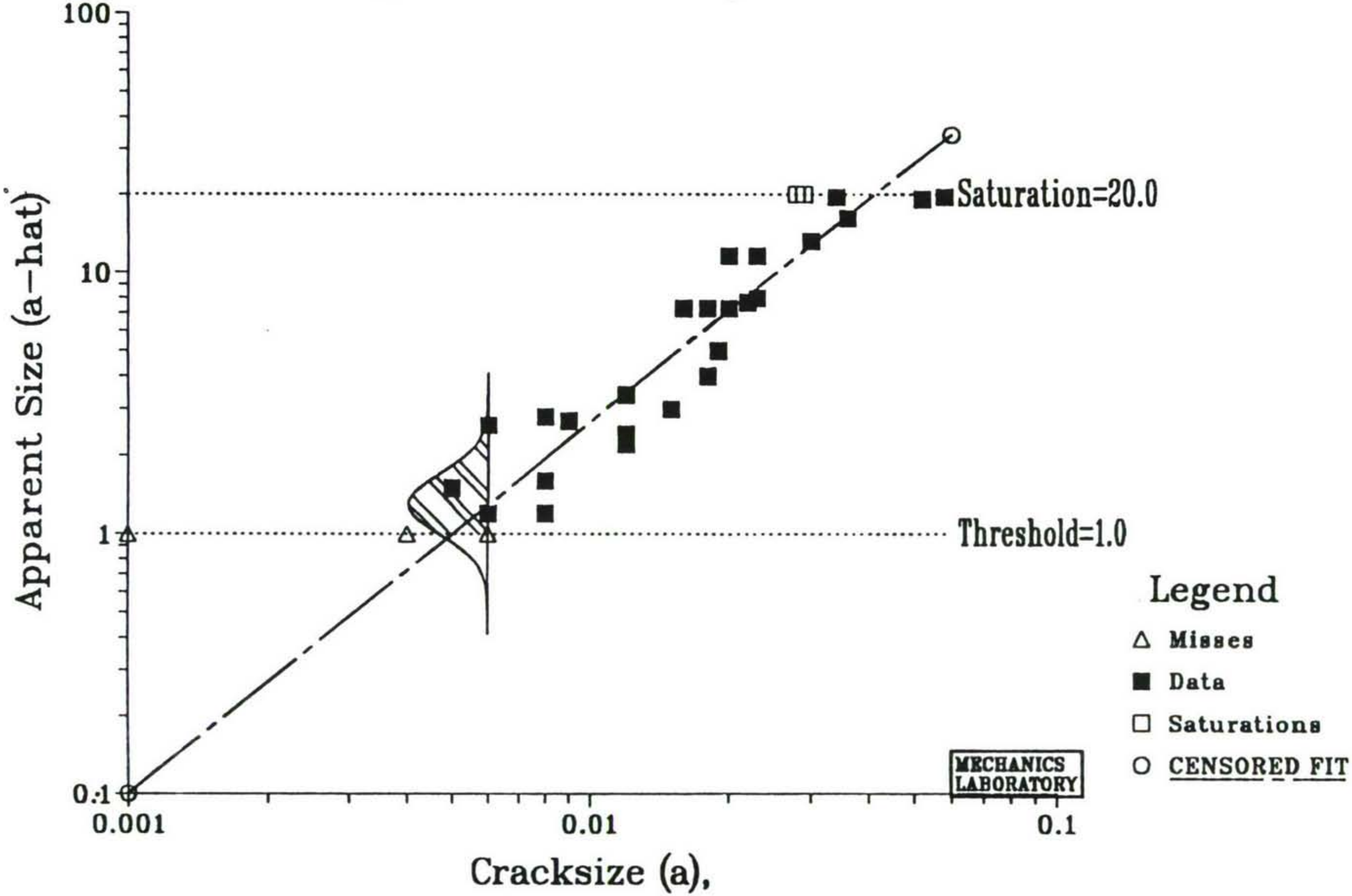
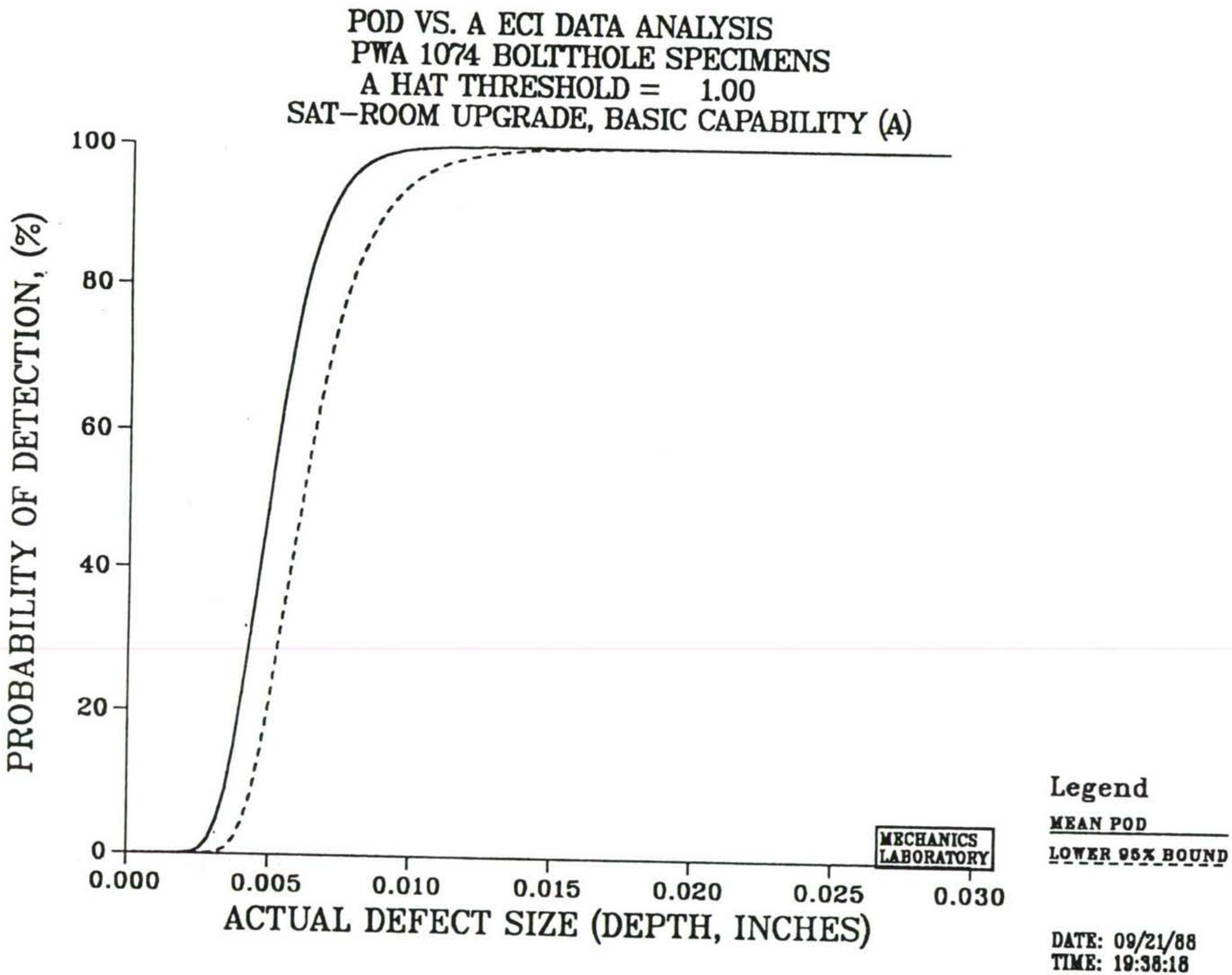


Figure 1
Resolution in POD vs. Resolution in Cracksize
Crack Found
Crack Missed
POD = n/N
POD
1
0.8
0.6
0.4
0.2
0
Cracksize, a
0.00
0.05
0.10
0.15

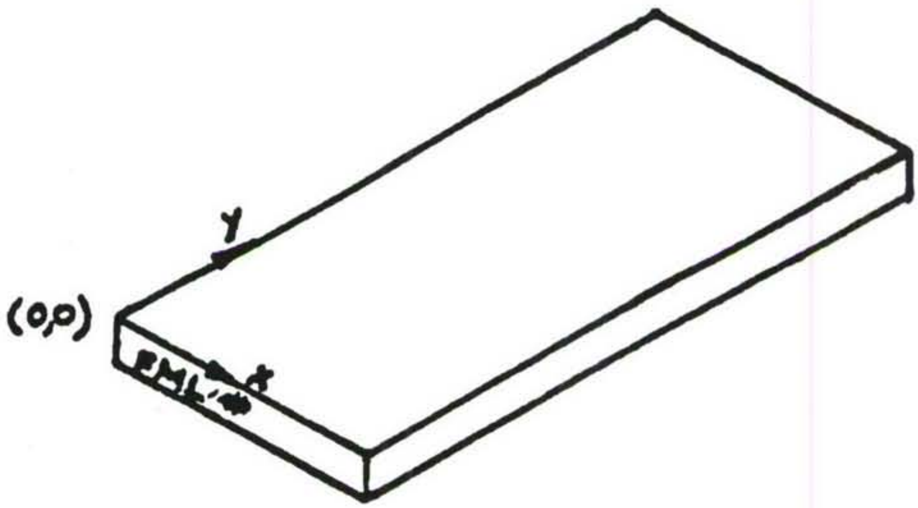
Figure 2
Large Bolthole Specimens
Shaded Region is Probability of Detection



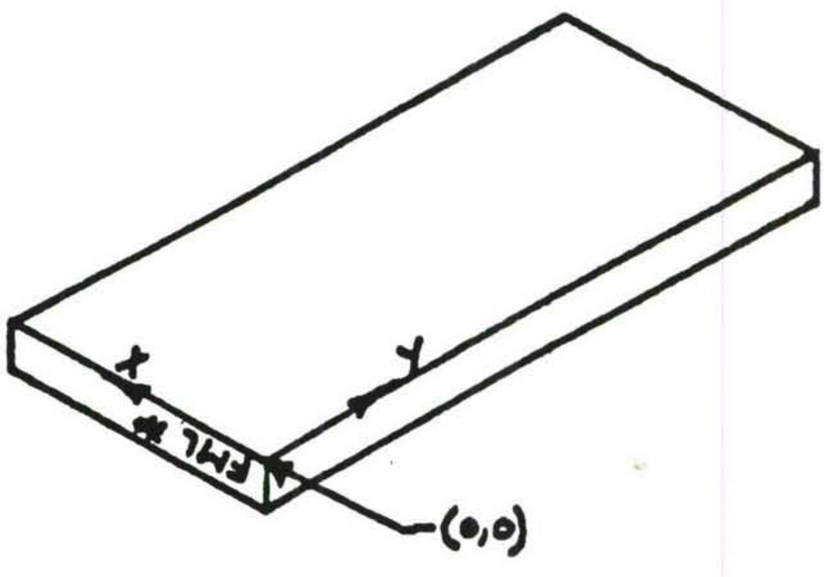

POD VS. A ECI DATA ANALYSIS
PWA 1074 BOLTTHOLE SPECIMENS
A HAT THRESHOLD = 1.00
SAT–ROOM UPGRADE, BASIC CAPABILITY (A)
PROBABILITY OF DETECTION, (%)
100
80
60
40
20
0
0.000
0.005
0.010
0.015
0.020
0.025
0.030
ACTUAL DEFECT SIZE (DEPTH, INCHES)
MECHANICS LABORATORY
Legend
MEAN POD
LOWER 95% BOUND
DATE: 09/21/88
TIME: 19:38:18

Appendices

FLAW LOCATION REFERENCE



X-Y REFERENCE WITH SPECIMEN IN UP POSITION



X-Y REFERENCE WITH SPECIMEN IN DOWN POSITION

Figure B-6.

F100-PW-229 ENSIP MANUFACTURING INSPECTION RELIABILITY TEST WEB/BORE SURFACE SPECIMEN FPI TEST DATA SHEET

Specimen FML # ____________________ Operator ____________________

Matrix/Test ____________________ Date ____________________

Facility ____________________ Inventory No. ____________________

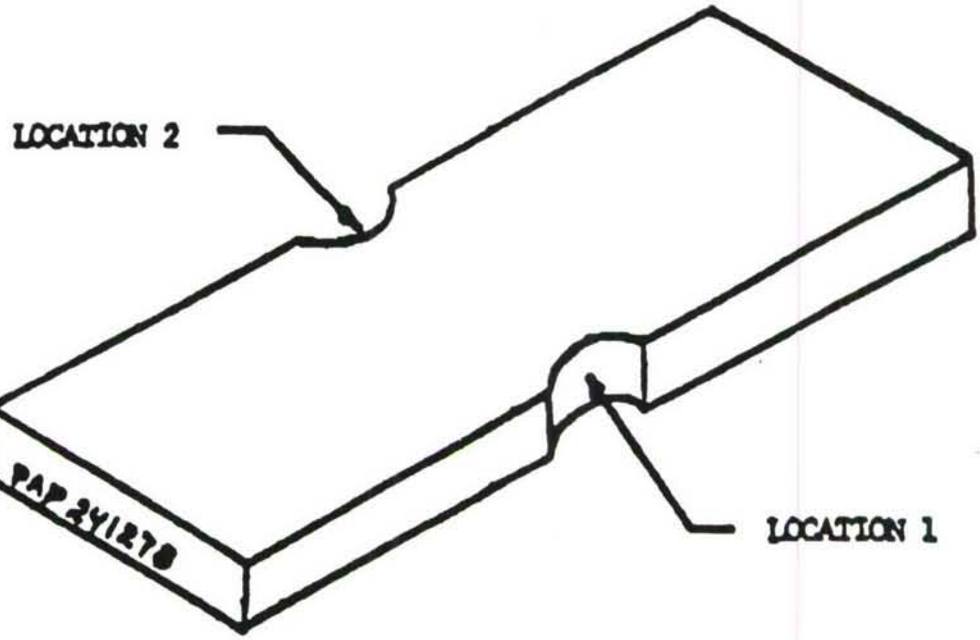
FPI System ____________________

	LOCATION *			
S/N	UP	DOWN	X	Y

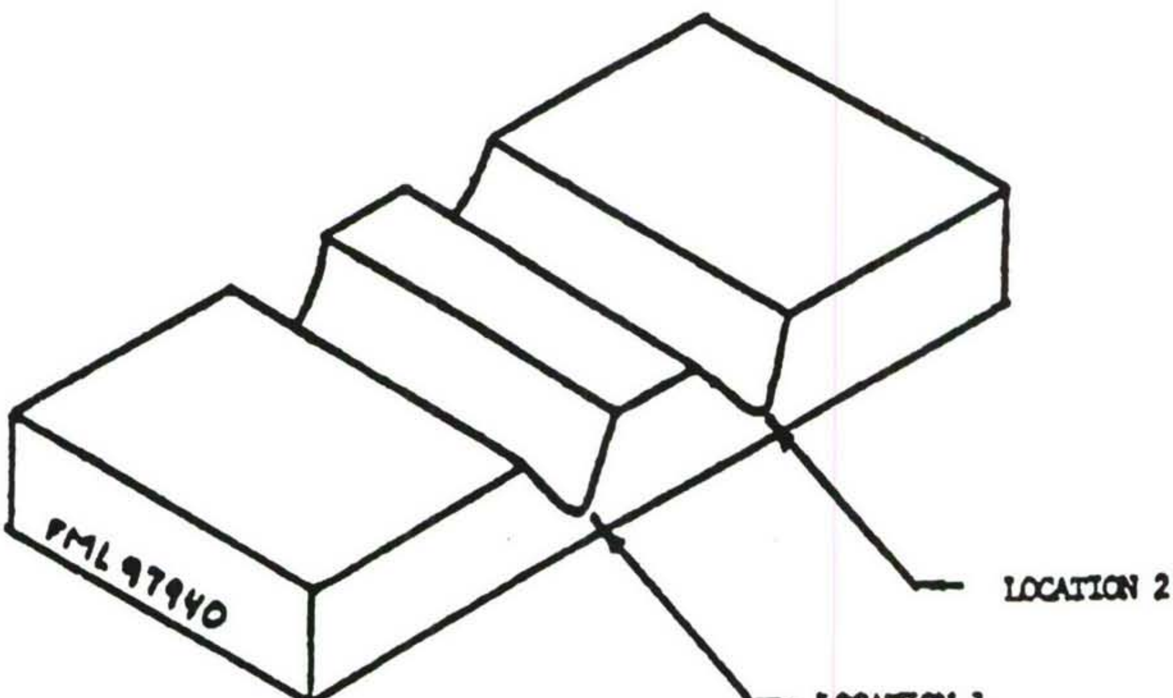
	LOCATION *			
S/N	UP	DOWN	X	Y

* SEE SKETCH FIGURE B-6 TO REFERENCE FLAW LOCATION WITH S/N

FLAW LOCATION REFERENCE



LOCATION REFERENCE OF SCALLOP SPECIMEN



LOCATION REFERENCE OF WEB/BORE FILLET SPECIMEN

Figure B-8

Figure 5-1

EDDY CURRENT
DATA SHEET

Date: ____________
Operator ID: ____________

Part Number ________ Serial Number ____________ Alloy ____________

Engine ____________ Part Name ____________ Surface Roughness ____________

*Attach Specification Sheet System Operating Ambient Temperature ________________

State other Equipment Environmental Constraints ____

Test Frequency ____________ Scan Speed ____________ Filtering ______________

Horizontal Gain ____________ Vertical Gain __________ Lift-Off-Technique ______

Coil Output Impedence

<u>Probe</u>

Contact ______________ Noncontact ____________

Differential ____________ Absolute ______________ Others _______________

Pancake ______________ Toroid Coil ____________ Others _______________

Coil Diameter __________ Shielding ______________

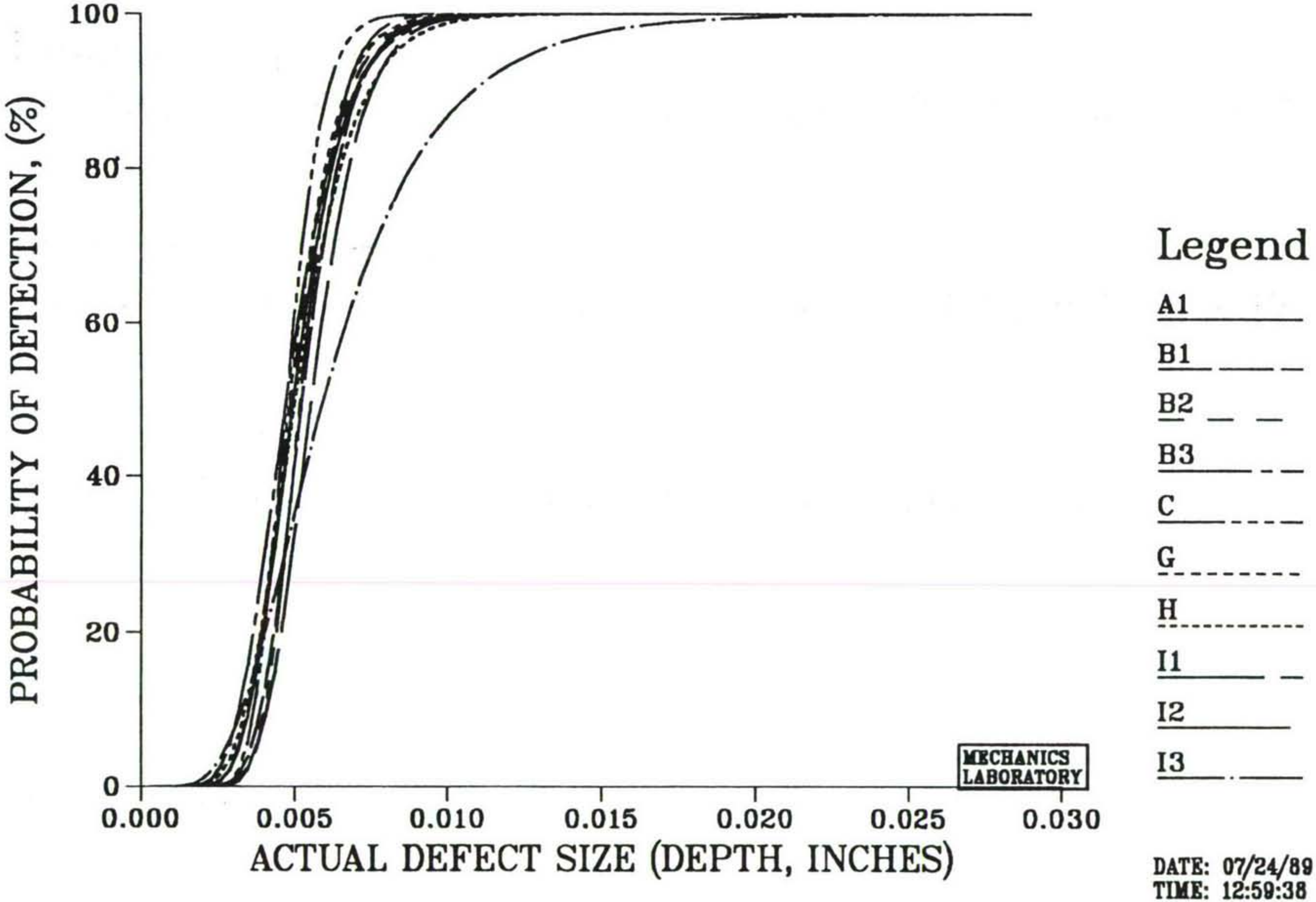
Scanning Technique __________________ Digitization ___________________

Calibration Level __________________ Inspection Threshold ___________________

Attach a sketch of the inspection setup. Include part orientation with respect to flaw orientation and eddy current direction.

Describe technique for analyzing, rejecting, and recording a defect signal.

Composite Plot for Semi–Automated Inspections Showing Inspection I3 To Be Different



Summary:

- The Proposed (first generation) MIL-STD is completed. (August, 1989)

- Upgrades to keep pace with advancing technologies should be addressed annually.

- Contact Sharon Vukelich, ASD, YZEE, for a copy of the proposed MIL-STD.

NDE for Engine Structural Integrity Programs

1989 USAF Structural Integrity Conference
December 5 - 7, 1989
San Antonio, Texas

W.L. Herron
GE Aircraft Engines
Cincinnati, Ohio

GE Aircraft Engines

T2191.12 · 891101

Outline

- Background
  - Eddy Current applications, capabilities, and limitations
- Signal processing
  - Goals
  - Approach
- Waveform analysis
- Image analysis
- Process evaluation

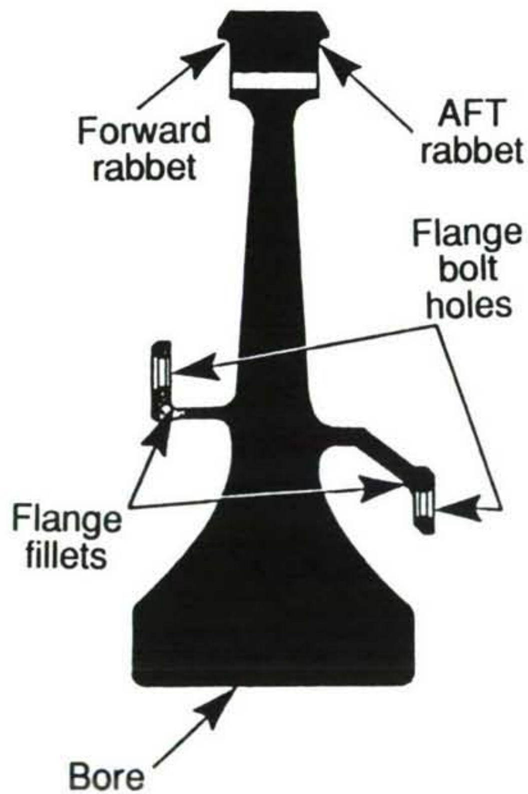
Background

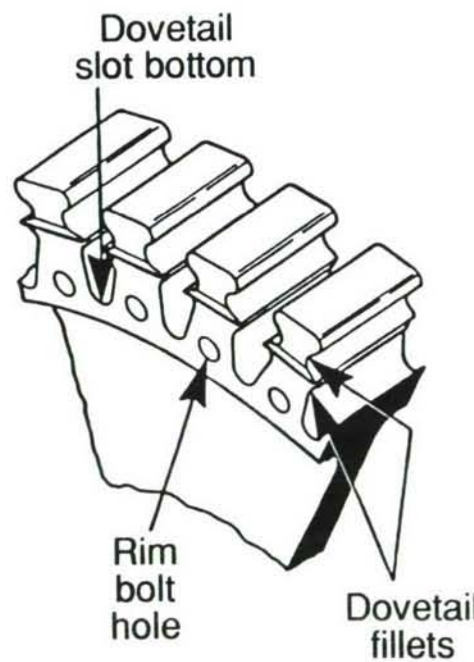
- Engine Field Management under the USAF Engine Structural Integrity Programs (ENSIP) relies heavily on Eddy Current inspection to detect small cracks

- The same inspections used in the field are implemented in the factory to:
  - Assure part inspectability
  - Assure that there are no pre-existing cracks or stress risers that will cause premature crack initiation/growth

Background (continued)

- The Eddy Current inspections focus on the areas of highest part stress.



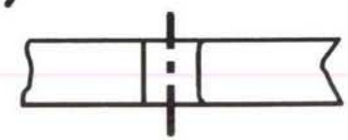


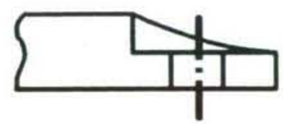
Background (Continued)

- Eddy Current detects local conductivity changes caused by cracks, tool marks, and handling damage

- It is also sensitive to some part geometry and to within-tolerance dimensional variations

  - Ends of skew holes 

  - Non-uniform breakedge 

  - Counterbore proximity 

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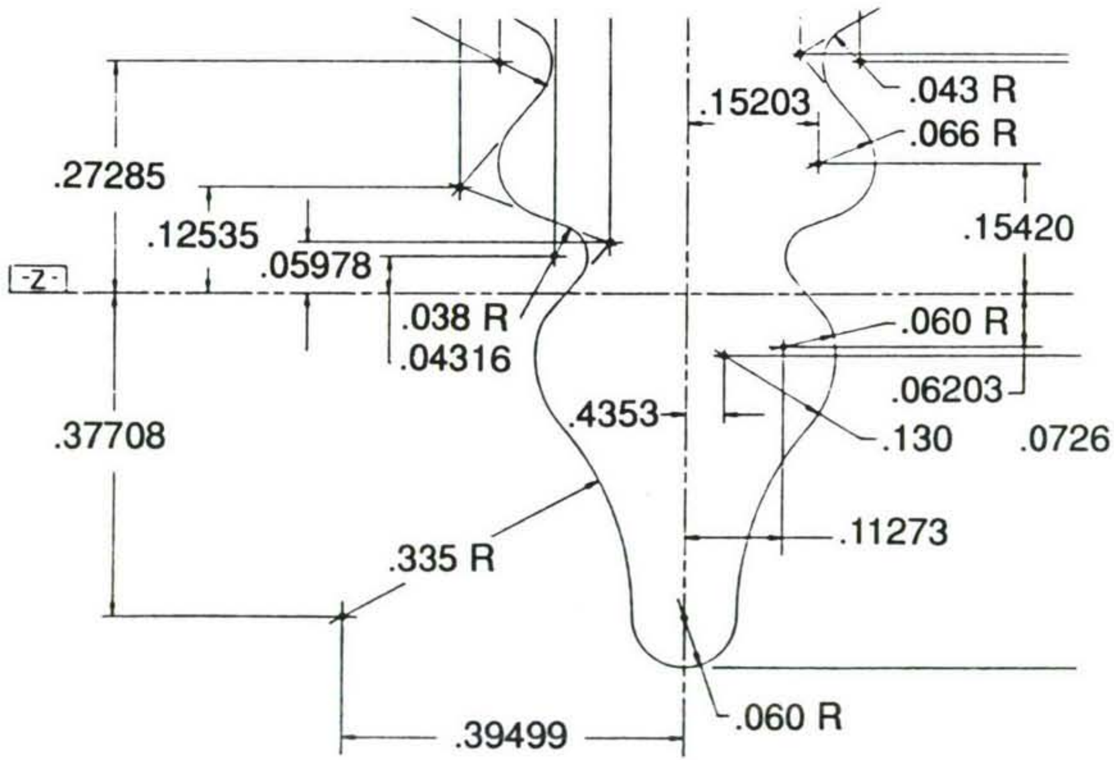
Dovetail Inspections

- The geometry associated with axial dovetails has also historically been a problem.
- Irregular contours and dimensional tolerances have prevented scanning around the slots.



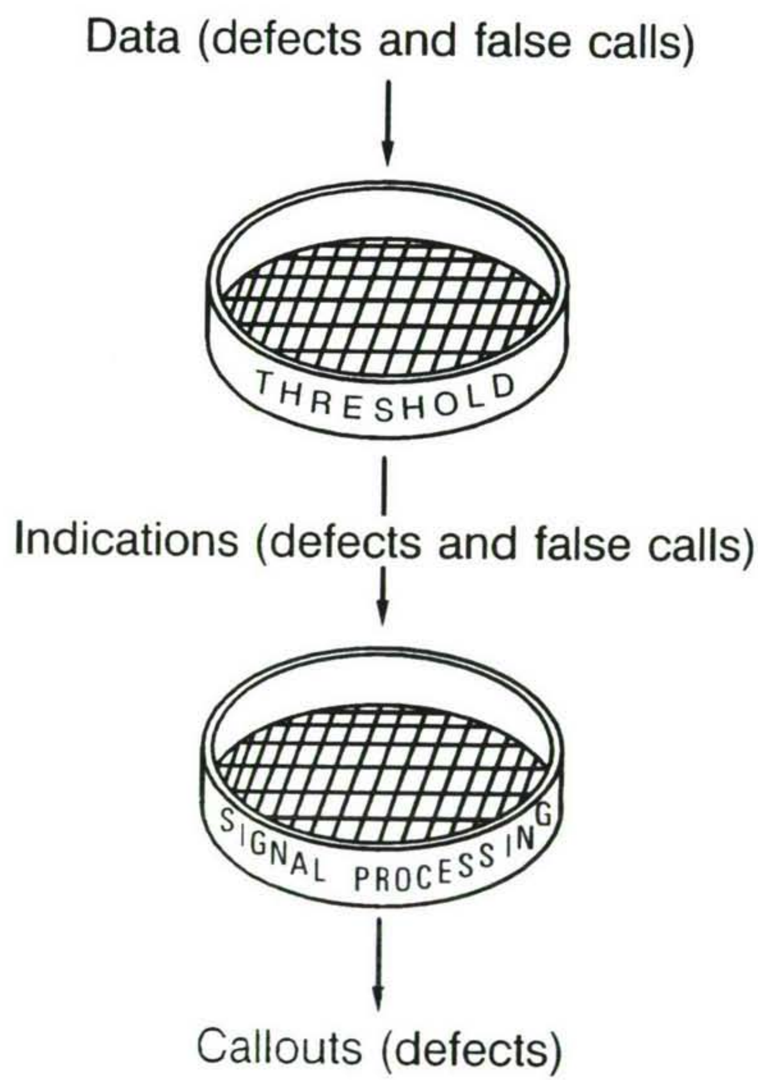
S-47941-100589

Dovetail Inspections

Scanning axially through the dovetail presents its own difficulties:

- Coil positioning requires fixturing for each line to be scanned, or very precise probe positioning/manipulating
- Forward and aft ends of the dovetails give large geometry signals -- overwhelming any flaw signals

Process Goal



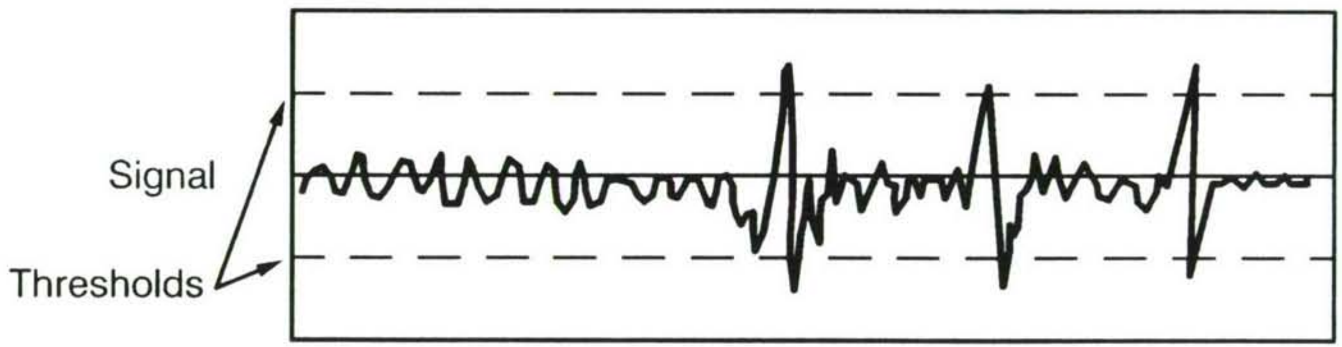
T2191.05 891101

Signal Processing

- The signal processing “sieve” relies on recognition of differences between defect signals and geometry signals
- This recognition looks for signal patterns in:
  - 1-dimension (waveform analysis)
  - 2-dimensions (image analysis)

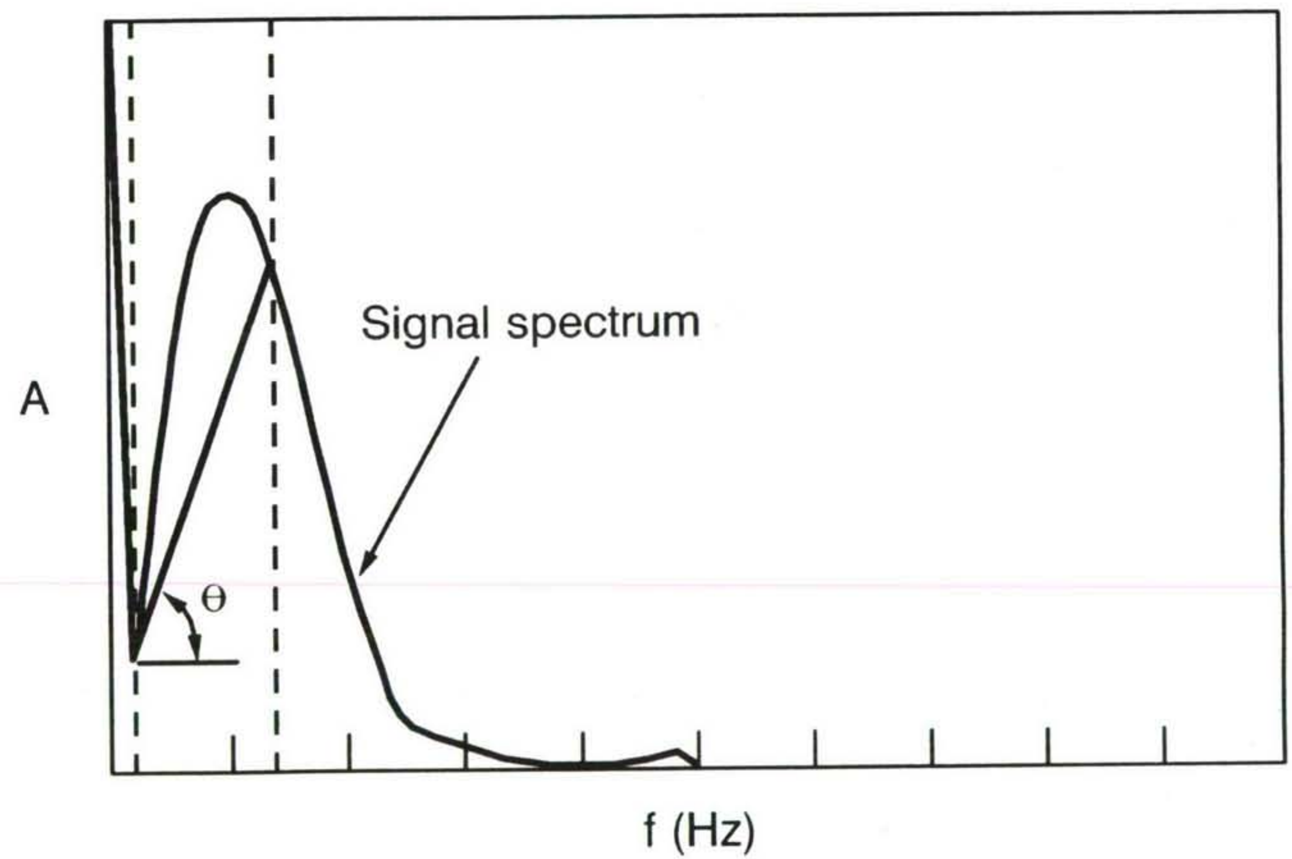
Waveform Analysis

- Examines each separate indication above the standard millivolt threshold:



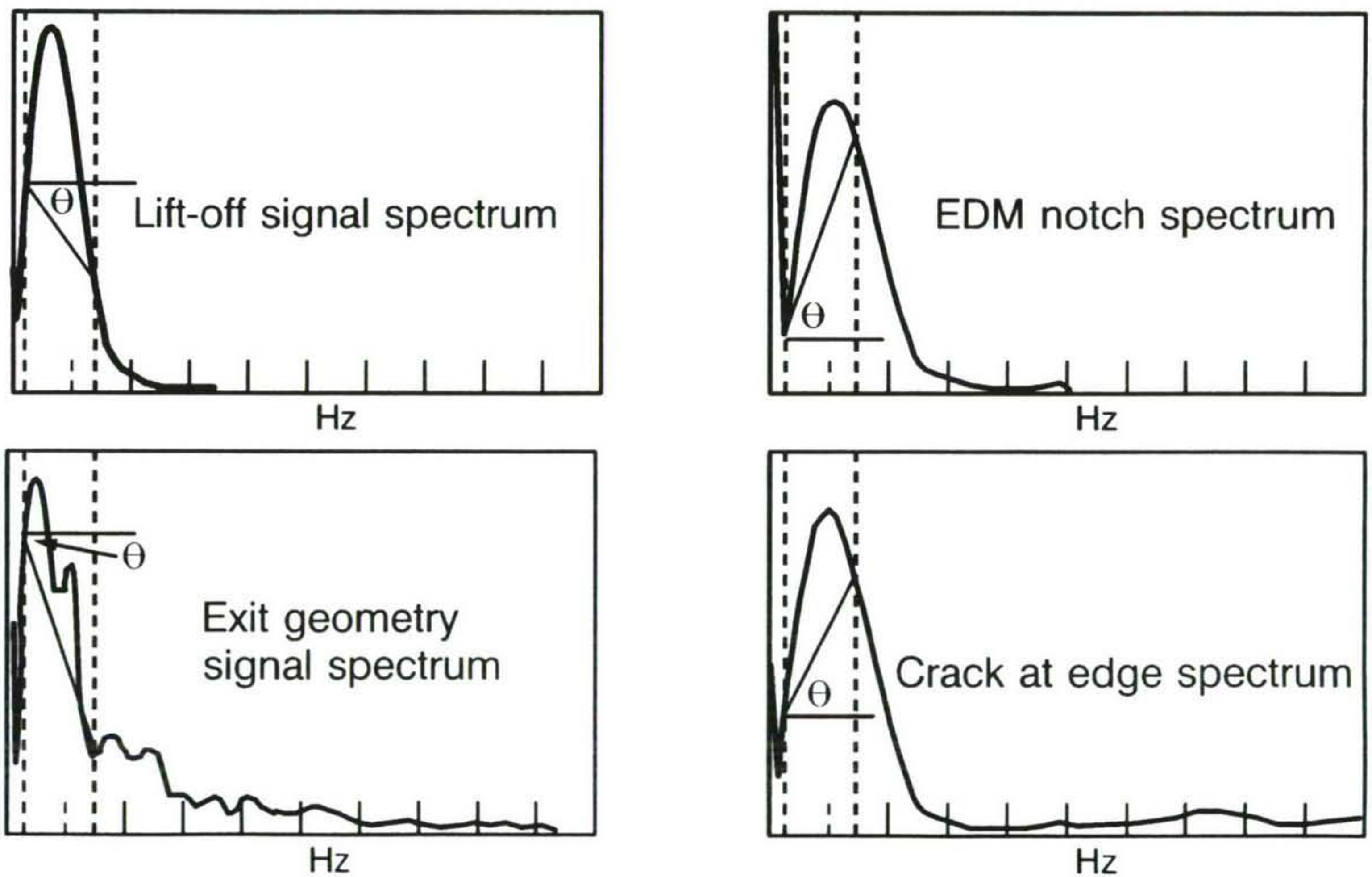
- Each waveform is individually considered in the frequency domain

This Waveform Analysis is Based on the Bandwidth of the Signal Frequency Spectrum



Bandwidth angle of a frequency spectrum

Discriminate Geometry Signals from Discrete Signals (Entry and Exit Geometry from Defect)

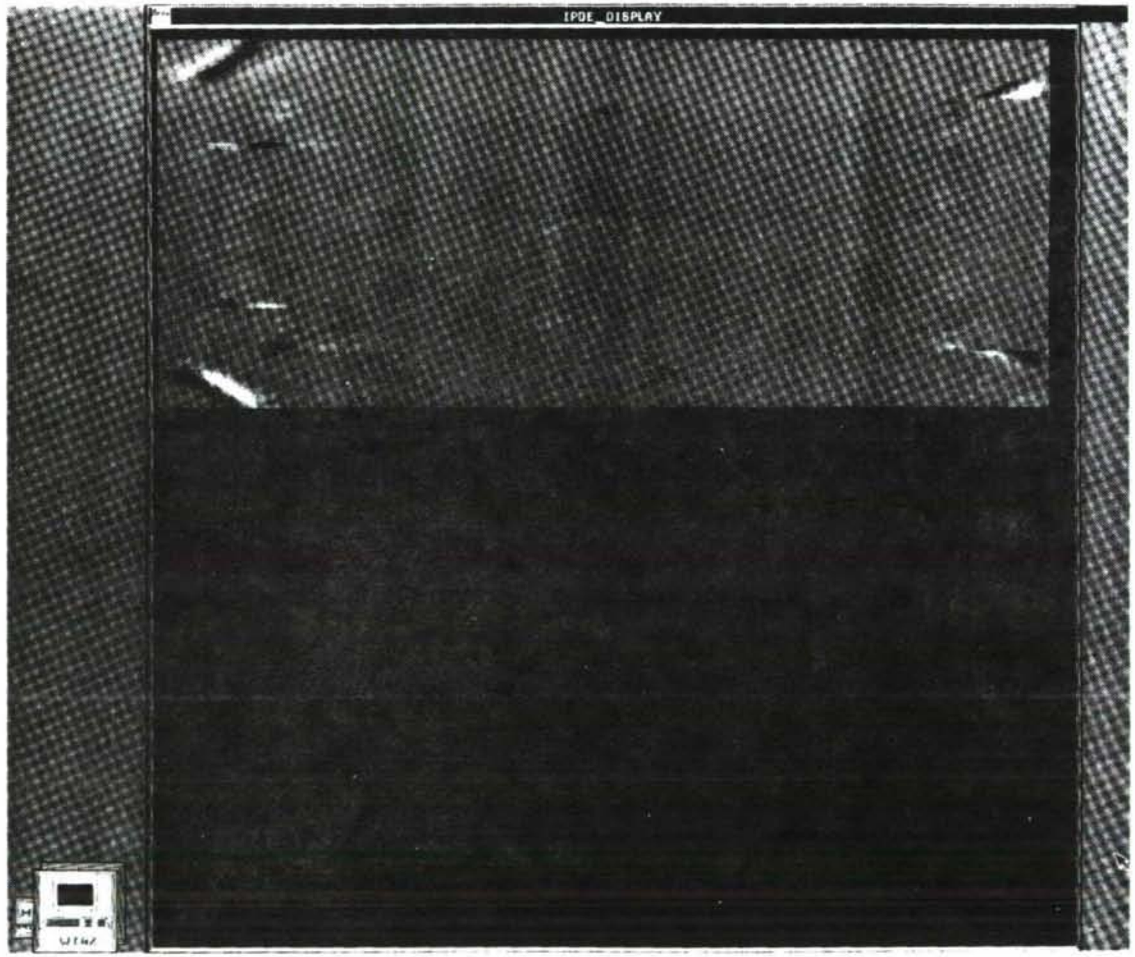


T2191.09 891101

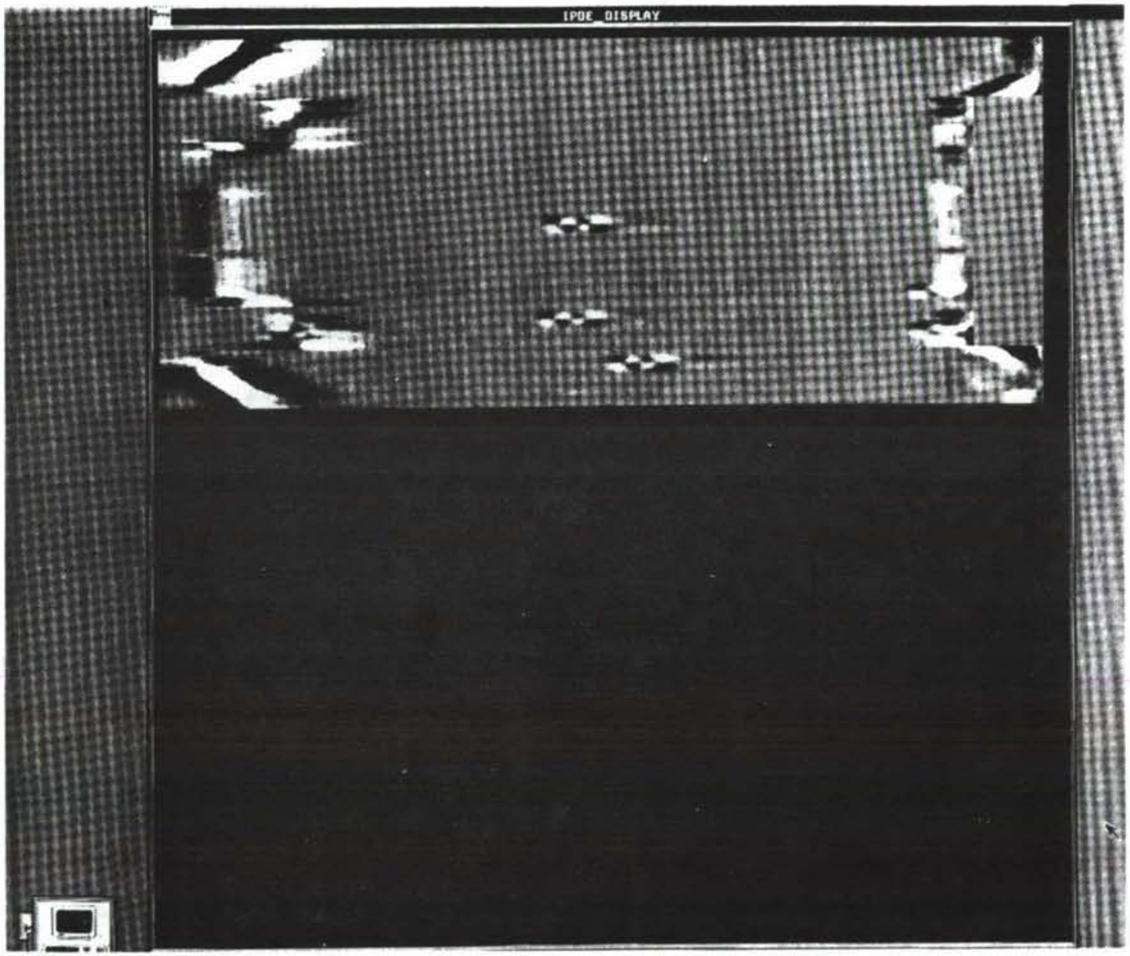
Image Analysis

- For dovetail inspections, 2-D imaging is used. In this, the geometry signals must be recognized and censored to allow flaw detection
- The following images illustrate steps in the inspection of a dovetail slot with several EDM notches
  - The axis of the slot runs left-to-right in the images
  - Top-to-bottom represents the slot circumference

Raw Image

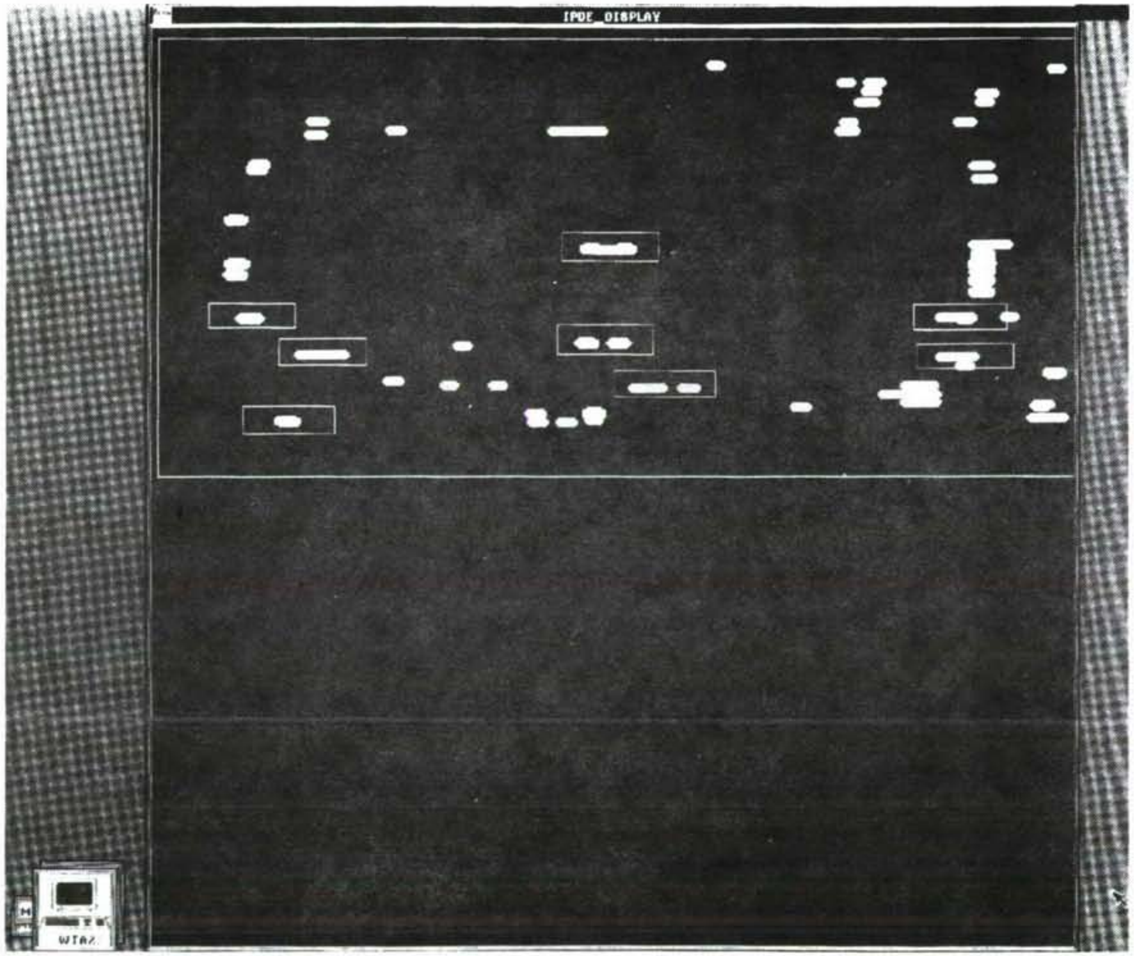


First Level Processed Image



S-47945-100589

Threshold Image—Indicating Callouts



S-47946-100589

Process Evaluation

- Evaluation follows the pattern outlined in the proposed "Mil Std for the NDE System Reliability Assessment"
  - Testing on LCF cracked specimens
  - "a" vs. "a-hat" data analysis
  - System variables evaluated
- Formal production demonstrations witnessed by USAF

Summary

- Convential Eddy Current techniques cannot extract flaw signals from geometry in some complex features
- GE has developed signal processing techniques to allow these inspections:
  - Waveform analysis
  - Image pattern recognitions
- These procedures have been demonstrated and are being implemented in GE Production

T2191.16 - 891101

Diffracto-Sight application to inspection of Composite and Metal Aircraft Structures

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David L.Simpson

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National Aeronautical Establishment
National Research Council, Canada

1989 USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE
5-7 December 1989
San Antonio, Texas

ABSTRACT

A new surface inspection technique, the Diffracto-Sight (D-Sight) method will be described. The technique was used to locate indentations associated with low energy impact damage. In graphite/epoxy specimens good correlation was observed between internal impact damage as shown on ultrasonic C-scan images and indentations detected with the D-Sight method. Test specimens are currently mounted on an aircraft to observe the influence of in-service surface degradation on technique resolution. Results of an inspection of composite aircraft structure previously exposed to regular service conditions will be shown. The method has the potential for inexpensive, rapid and objective detection of low energy impact damage over large areas of composite aircraft structure. Application of this method, because of its inherent reliability, could result in the increasing of the design allowable strain levels for some composite components.

In metal aircraft structures, the technique was shown to be effective in demonstrating which holes were subjected to the process of cold working. The result of the inspection can be easily stored on video tape, as printer output or digitized. Since often large numbers of holes have to be cold worked, this capability is significant advantage of the technique.

The Shadow Moire technique has been proposed for similar applications, however D-Sight offers the advantages of a large field of view, a resolution independent of the color of the surface inspected, simpler interpretation of images and ease of adaptation of image processing techniques resulting in operator independent objective inspections.

1 Introduction

Diffracto-Sight or D-Sight (developed by Diffracto Limited of Windsor, Ontario) is a patented [1] method of visualizing surface distortions, depressions or protrusions, as small as 10 micrometers. It is a real-time technique with particular application for the rapid inspection of large surfaces. Computer based image processing techniques have been applied to D-Sight images. An image from a previous inspection can be directly compared to current results for quick and operator independent identification of areas where changes in surface features have occurred.

Advanced composite materials, especially graphite reinforced epoxies, have high specific strength and stiffness properties and very good fatigue resistance. Unfortunately, these materials are sensitive to low energy impact damage from such common occurrences as hailstones, stones thrown off the runway or tools dropped by maintenance personnel. These impacts, commonly referred to as Barely Visible Impact Damage or BVID, can result in barely visible surface damage but significant internal damage.

This sensitivity to impact damage is reflected in the approach taken by both designers and certification authorities. Composite aircraft structures are expected to carry ultimate load in the presence of BVID type damage. The damage threshold is related to current in-service inspection capabilities and directly influences design allowable strain levels for structures fabricated from composite materials. Regular in-service inspections of aircraft with ultrasonic scanning devices are not practical if only due to cost and time required for such inspections. Current inspection practices rely on visual detection of significant impact damage.

Komorowski [2] suggested the application of a D-Sight technique to the inspection of composite structures for BVID. Previous papers [2][3][4], reported the results of experiments in the application of D-Sight to the detection of impact damage. In [3] the significance of D-Sight to composite aircraft structure inspections was explored. Because of D-Sight's inherent reliability, it could be possible to increase the design allowable strain levels for some composite components through relaxation of the visible damage criteria.

Recently Simpson suggested the use of D-Sight to detect cold worked holes. The process of cold working of holes extends the fatigue life of components by delaying crack initiation [5]. The process is being used on aging aircraft as well as on new aircraft in production. Often large numbers of holes have to be cold worked and D-Sight could be used as a quality assurance tool not available previously. The D-Sight technique is highly reliable in this application and, because it is capable of inspecting a number of holes simultaneously, is economic.

At the previous ASIP conference Scotese and Huang [6] presented a low velocity impact damage detector based on the Shadow Moire method. Results of experiments with Shadow Moire from an internal Structures and Materials Laboratory program are reported in [7].

This paper gives a brief overview of D-Sight system followed by discussions of its applicability to impact damage detection in composites and the inspection of cold worked holes. A comparison is then made between D-Sight and Shadow Moire techniques.

2 D-Sight

The optical set-up for D-Sight, described in References [8] and [9], consists of a light source, a retro-reflective screen and the object being inspected (Figure 1). The surface being inspected must be reflective, however rough surfaces can be made reflective by wetting with a fluid. The D-Sight effect can be explained using geometric optical principles. If a flat surface with an indent is inspected, then the light striking the indent is deflected. It then strikes the retro-reflective screen at a point removed from the light rays reflected from the area surrounding the indent. The retro-reflective screen attempts to return all these rays to the initial reflection point on the inspected surface. However the screen, which consists of numerous glass beads, returns a cone of light to the surface, not a single ray. This imperfection of the retro-reflective screen creates the D-Sight effect. By back-lighting the defect, the technique increases the light intensity on one side of the indent and reduces it on the opposite side. The process can be viewed as a slope detecting technique with positive surface slopes looking dark and negative slopes looking bright relative to the background. Both flat and moderately curved (in any orientation) surfaces can be inspected using this method.

The D-Sight technique was initially developed to inspect car body panels and metal working dies and Diffracto has several commercial versions of D-Sight in production. The most complex systems include video cameras and computer based image processors. D-Sight images can be stored for future reference as video images on tape, digitally on computer storage media, or as video printer output.

3 Impact Damage Detection

3.1 Laboratory Testing

Details of D-Sight evaluations using impacted specimens were reported in [4]. Table 1 summarizes the test results. It should be noted that two graphite/epoxy material systems were used. The AS4/3501-6 is an older system and typical of the first graphite/epoxies used in large production runs (F-18, AV-8B fighters). The IM6/F584 is a more recent system with tougher resin.

Before impacting, the specimens were inspected with D-Sight at Diffracto for pre-impact surface characterization. These results were recorded using a video camera and image processor. A commercial solution of oil and kerosene was used to enhance the reflectivity of the specimen surface. This may not be an acceptable solution for some applications, however, Diffracto has recently demonstrated the suitability of a combination of water and glycerine for reflectivity enhancement.

The results of the D-Sight inspections of the impacted specimens along with pre-impact D-Sight images are shown in Figure 2 (for panel 297). To quantify D-Sight indent indication, the diameters of the dark and light spots on the D-Sight panel image were measured. Since the panels were compressed in the images to approximately 67% of their full width the measurements were non-dimensionalized. This was done by dividing all measurements by the width of the panel in the image and multiplying by 100. This eliminated the scaling effect and measures thus produced were called relative D-Sight units. All D-Sight results in this crudely quantified form are included in Table 1.

The zoom lens of the video camera was adjusted such that the (305x178mm (12x7in)) specimens occupied the full field of view. This made quantification of the D-Sight images more precise. Figure 3 shows a D-Sight image of two specimens with impact indents of 70 and 55 micrometers deep respectively in the center of an area approximately 0.9 m wide and 2.1 m long (3x7 ft). As can be seen, both indents are easy to locate. Without D-Sight these impact sites would be invisible to the unaided eye.

All 200 series specimens had the CF-18 paint finish. However, the 295-B specimen was impacted from the unpainted side. The D-Sight image of this specimen and specimen 294-C are shown in Figure 4. As can be seen coloration of the surface does not affect impact indent visibility.

Figures 5, 6 and 7 show plots of the D-Sight impact damage indication versus the C-scan measured damage for both material systems for 8, 24 and 48 ply specimens respectively. These plots show that there is a correlation between internal damage detected by C-scan and external damage detected by D-Sight. The more brittle AS4/3501-6 system experiences more internal damage relative to external surface deformation measured in D-Sight 'units'. For both systems, D-Sight indications were present for all internal damage greater than 15 mm as measured from C-Scan inspections.

The D-Sight based technique of impact damage detection relies on the impacting object leaving a permanent indent on the impacted surface. From the plot in Figure 8 it can be seen that at the low impact energy levels resulting in barely visible impact damage there is a fair correlation between indent depth and impact energy level. This observation is not significantly influenced by specimen material system or thickness. It should be noted that the thicker laminates exhibit a higher threshold for the onset of surface indents.

3.2 Implications

The implications of using D-Sight for in-service detection of impact damage in aircraft composite structures can be evaluated against the draft USAF damage tolerance design requirements identified by Whitehead [10]:

Scratches:
- Assume the presence of a surface scratch 100mm (4.0in) long and 0.4mm (0.02in) deep

Delamination:
- Assume the presence of an inter ply delamination that has an area equivalent to a 50.8mm (2.0in) diameter circle with dimensions most critical to its location

Impact Damage:
- Assume the presence of damage caused by the impact of a 25.4mm (1.0in) diameter hemispherical impactor with 136 J (100 ft*lbf) of kinetic energy or that kinetic energy required to cause a dent 2.54mm (0.10in) deep, whichever is least.

The impact indent assumption is well above the D-Sight threshold of 10 micrometers. The D-Sight detected damages are plotted as indent depth versus C-scan measured delamination in Figure 9 and versus impact energy in Figure 10. The damage assumptions from draft USAF damage tolerance design requirements are also shown in both figures for reference.

Recently Levin [11] has subjected impacted graphite/epoxy sandwich specimens to fighter spectrum fatigue loading. He observed that after only few hours of loading indent visibility was reduced by 20 to 30%. The result was alarming in that the critical impact damage had become non-visible. These indents between 1 and 2 mm deep would still be easily detectable with D-Sight.

3.3 Flight testing.

As part of a preliminary investigation, some of the impacted specimens are being flown mounted behind the nose landing gear of a T-33 aircraft operated by the NAE Flight Research Laboratory. After 56 hours of operation which involved 150 takeoff and landing cycles over a 10 month period, there is no indication that the aircraft service environment will affect the ability of D-Sight to detect impact damage.

3.4 Composite Material Aircraft Structure Inspection.

D-Sight inspections of a CF-18 horizontal stabilator and trailing edge flap were carried out. The components had seen extensive service prior to being inspected. The D-Sight images are shown in Figures 11 and 12. The appearance of the CF-18 composite structure was essentially identical to the laboratory specimens. While impact damage was absent from the inspected components, some indents, not previously observed were found. Ultrasonic inspection determined that these were manufacturing defects.

3.5 Conclusions and Recommendations for Impact Damage Detection.

Based on this preliminary investigations, the following conclusions have been reached:

- D-Sight appears to have sufficient resolution to detect post-impact surface perturbations indicative of significant sub-surface damage;

- The D-Sight method has the potential for the rapid, inexpensive and reliable inspection of large aircraft external composite surfaces;

- Widespread application of D-Sight could lead to higher design strain allowables for some composite structures and it may significantly influence future certification requirements for composites.

- Full correlation studies between the D-Sight detection capabilities and the resultant damage must be undertaken to quantify any application limits. These studies must address the issues of material systems, lay-up, component surface coating, fatigue loading and normal surface degradation. Probability of detection data must be determined before the full potential of the method can be realized.

4 Cold Worked Holes Detection.

The process of cold working of holes extends life of aging and new aircraft. A popular technique called Split Sleeve Cold Expansion has been used for nearly 20 years. A typical improvement in fatigue life of 3:1 is claimed by Fatigue Technology Inc.(FTI) [12].

The hole is cold worked by inserting and subsequently withdrawing a tapered mandrel. The mandrel is fitted with a pre-lubricated split sleeve. As the mandrel is drawn through the sleeve, radial plastic expansion of the hole occurs creating a permanent zone of compressive stresses at the hole circumference. These residual stresses lower the cyclic tensile stresses and provide enhanced fatigue resistance.

The cold working process deforms the metal surface around the hole edge. The 'ridge' formed around the hole edge in aluminum plates is approximately 0.2 mm high. D-Sight is capable of detecting this 'ridge' thus confirming the hole has been cold worked. With development, it may also be possible to categorize the degree of cold working against a reference standard.

4.1 Laboratory Testing.

A specimen was manufactured from 7075-T6 aluminum alloy to the dimensions shown in Figure 13. All holes, except one row were cold worked using the FTI Split Sleeve Cold Expansion process.

Results of the D-Sight inspection of the specimen are shown in Figures 14 and 15. In Figure 14, only 4 holes are shown. The two cold worked holes on the left show up as dark ellipses due to the angle at which they are observed. The deformed surface around the hole shows up as distinct halo around the dark ellipse. In Figure 15 the whole specimen D-Sight image is shown. The row of non cold worked holes is indicated by the box.

The results are very encouraging. For the two sizes of holes tested the technique was equally sensitive.

4.2 Conclusions and Recommendations for Cold Worked Holes Detection.

The D-Sight technique can be used to confirm the cold working of holes.

Further evaluation should be carried out using different materials, hole sizes, degrees of cold working and various cold working processes.

For sites where application of D-Sight may be difficult, a silicon rubber or similar replica of the surface with cold worked holes could be taken and inspected with D-Sight.

5 Comparison with Shadow Moire.

The Moire fringes are produced when two sets of lines are superimposed to produce optical interference. In the Shadow Moire method, the shadow of a grating on inspected surface is observed through the same grating. A typical set up is shown schematically in Figure 16.

Typical Shadow Moire images of an impacted panel and a specimen with cold worked holes are shown in Figures 17 and 18.

Interpretation of the fringes is generally straight forward - the fringes are contour lines of the inspected surface. Each line is a location of points equally distant from the grating used to produce the effect.

Sensitivity of the Shadow Moire method depends on the angle of illumination, the angle of observation and the density of the grating. However, for higher density gratings (10 or 40 lines per mm) diffraction becomes a serious problem. The grating has to be placed on the surface being inspected. Scotese and Huang [6] reported 0.76 mm separation between surface and gratings.

The size of the grating limits the area that can be inspected. For curved surfaces and for the higher density gratings needed for impact damage detection, the inspection area is limited by the maximum allowable separation of the grating and the surface. Scotese and Huang [6] limited their device to 125x100 mm gratings. Much larger areas of a slightly curved surface can be inspected with D-Sight using one image. As shown in Figure 3, a 0.055 mm indent is detectable in an inspection area of 2100x900 mm.

The Shadow Moire fringe pattern depends on the exact positioning of the grating relative to the surface. If direct comparison of repeat inspections using high density gratings is required, great care must be taken to reproduce a fringe pattern observed previously. The D-Sight system is inherently very sensitive (approximately comparable to a Shadow Moire inspection with 40 lines per mm grating) and very tolerant to changes in set up geometry. Comparison of the cold worked holes specimen images (Figures 15 and 18) are good example of this problem. All cold worked holes produced an essentially similar pattern in the D-Sight image. The appearance of the cold worked hole in the Shadow Moire image depended on the relative position of the hole and the grating. Some cold worked holes could have been taken for unprocessed holes in the Shadow Moire image.

Both methods generally require simple surface preparation. The Shadow Moire technique requires a lightly colored surface to give good fringe contrast. Scotese and Huang [6] used white spray powder. D-Sight needs highlighting fluid to make the inspected surface reflective.

From the D-Sight image, it is possible to determine whether the surface perturbation is concave or convex. The Shadow Moire technique requires an additional operation to be performed to obtain this information.

The D-Sight technique produces an image in which changes in light intensity can be related to surface perturbations. This type of image can be easily analyzed by standard image processing algorithms. In order to adopt image processing to Shadow Moire special software for interpreting interference fringes has to be developed.

Shadow Moire produces results which can be easily quantified. The technique can determine the shape of the surface inspected. A more rigorous D-Sight theory is required before similar measurement with this technique will be possible.

5.1 Conclusion: D-Sight - Shadow Moire Comparison.

The two techniques complement each other. D-Sight seems to have advantages as a surface perturbation detection method. D-Sight inspection results are easy to reproduce for repeat inspections. The Shadow Moire technique is more adequate for measuring the surface shape.

6 Summary

The D-Sight technique has been shown to be successful in detecting non visible impact damage in composite materials. The technique was used on actual aircraft components. The D-Sight technique could also be used to confirm the cold working of holes in metal aircraft components. The results indicate that further development and application of this method should be pursued.

7 Acknowledgments

This work was carried out under NAE Project 07336 Task 4269, D-Sight Application to Aircraft Inspection.

The authors would like to thank the following people:

Dr. Walter Pastorius of Diffracto Ltd. for performing most of the D-Sight inspections.

Mr. T. Chapman of National Aeronautical Establishment (NAE) for performing C-Scanning tests.

The NAE Flight Research Laboratory personnel for flying of specimens on their aircraft.

The personnel of Department of National Defence and Bombardier-Canadair for making the CF-18 inspection possible.

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[12] Fatigue Technology Incorporated, Focus on Fatigue, Vol.1, Winter, 1989.

Number of plies	Specimen	Impact energy [J]	c-Scan damage [mm]	D-Sight indication	Profiler [mm]
48	294-A	10.85	9.2	3.3	0.018
48	294-B	13.56	48.6	8.8	>0.1
48	294-C	10.85	51.4	8.4	0.111
48	295-A	9.49	7.3	2.5	0.014
48	295-B	13.56	44.4	8.4	0.119
48	295-C	10.85	18.3	7.0	0.094
24	296-A	4.07	8.3	0	n.v.
24	296-B	4.07	8.3	0	n.v.
24	296-C	5.42	29.3	8.5	0.051
24	297-A	10.85	33.9	9.5	0.104
24	297-B	8.13	30.3	7.6	0.089
24	297-C	4.75	25.7	6.5	0.07
8	298-A	5.42	22.9	6.8	0.063
8	298-B	0.68	7.3	1.8	0.016
8	298-C	6.78	26.6	6.0	0.07
8	299-A	1.36	12.9	3.3	0.028
8	299-B	4.07	18.6	5.1	0.055
8	299-C	4.07	24.3	7.2	0.055
48	357-A	10.85	32.4	5.5	0.104
48	357-B	16.27	40.5	5.5	0.108
48	357-C	21.69	56.6	7.1	0.235
24	356-A	2.71	15.8	0	0
24	356-B	5.42	23.7	3.2	0.093
24	356-C	8.13	35.6	4.0	0.133
8	355-A	2.71	22.5	3.9	0.05
8	355-B	1.36	14.8	1.6	0.024
8	355-C	4.07	35.6	5.6	0.067

All specimens 305x178 mm.
The 200 series specimens were made from IM6/F584 graphite epoxy.
The 300 series specimens were made from AS4/3501-6 graphite epoxy.
The lay up: $[0/45/90/-45]_s$.
The impactor: 25.4mm (1in) diameter spherical.

Table 1. Impact and D-Sight test data.

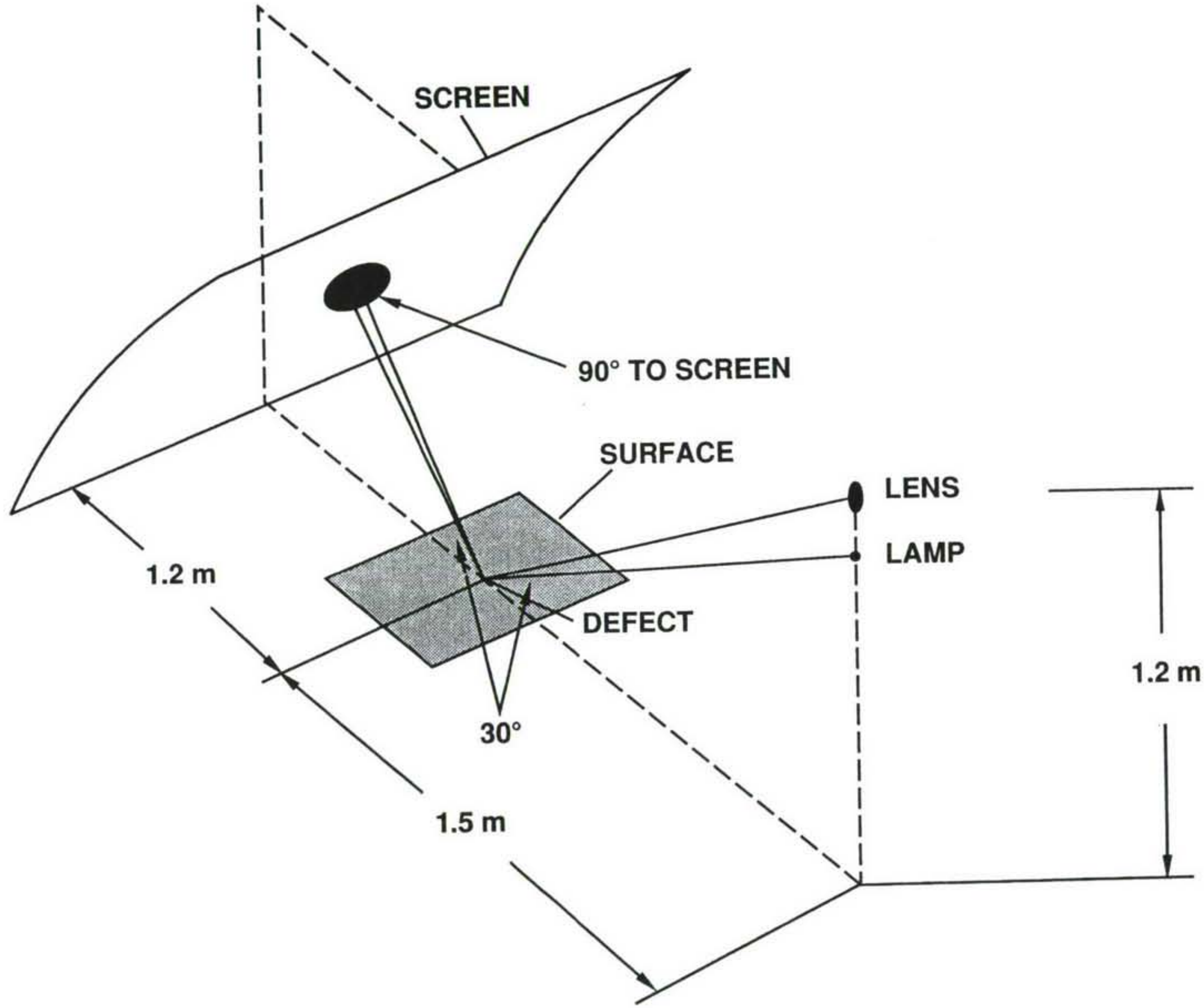

Figure 1. Diffracto-Sight Set-Up

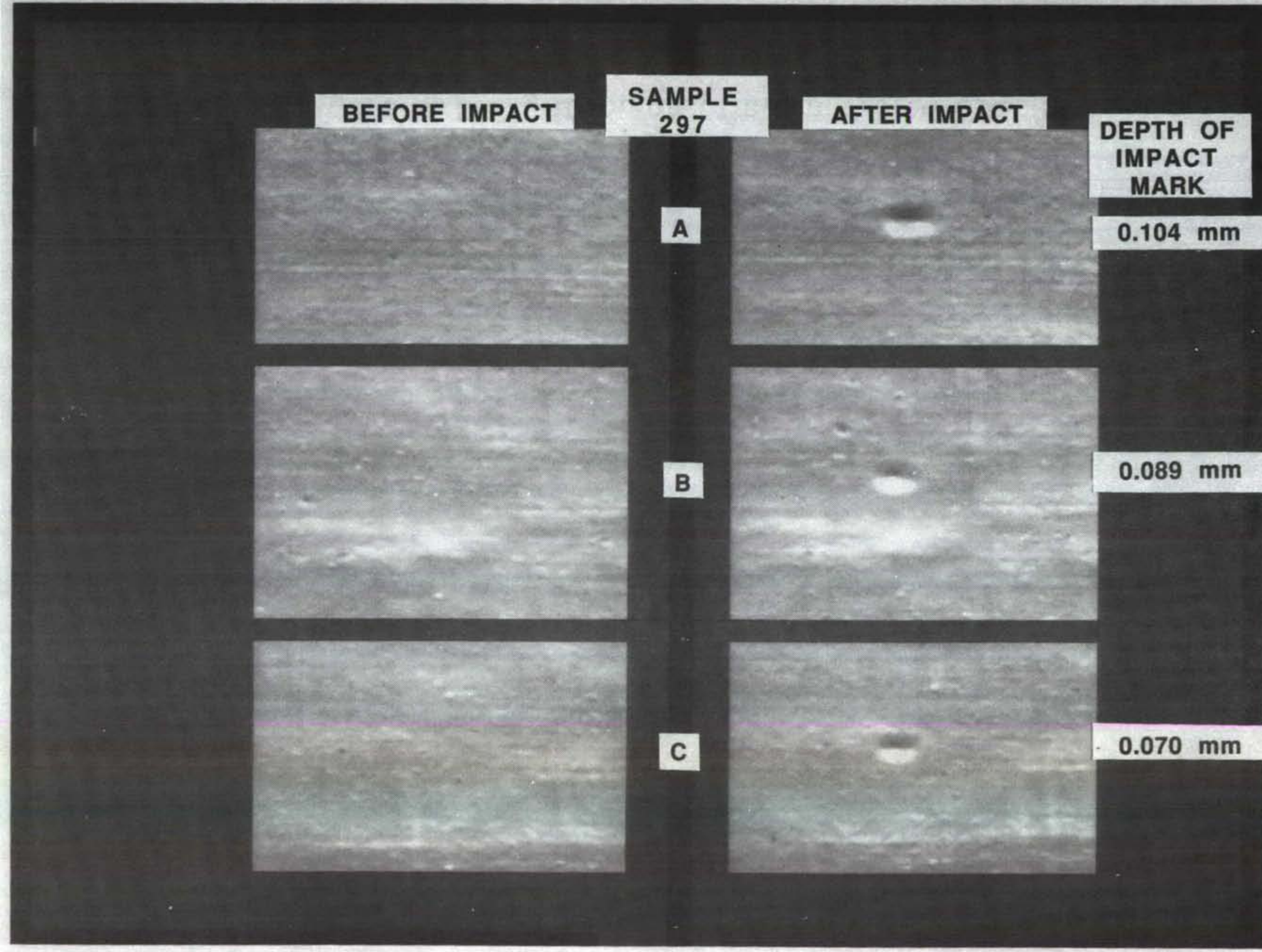

Figure 2. D-Sight images of specimens 297 - A, B, C before and after impact.

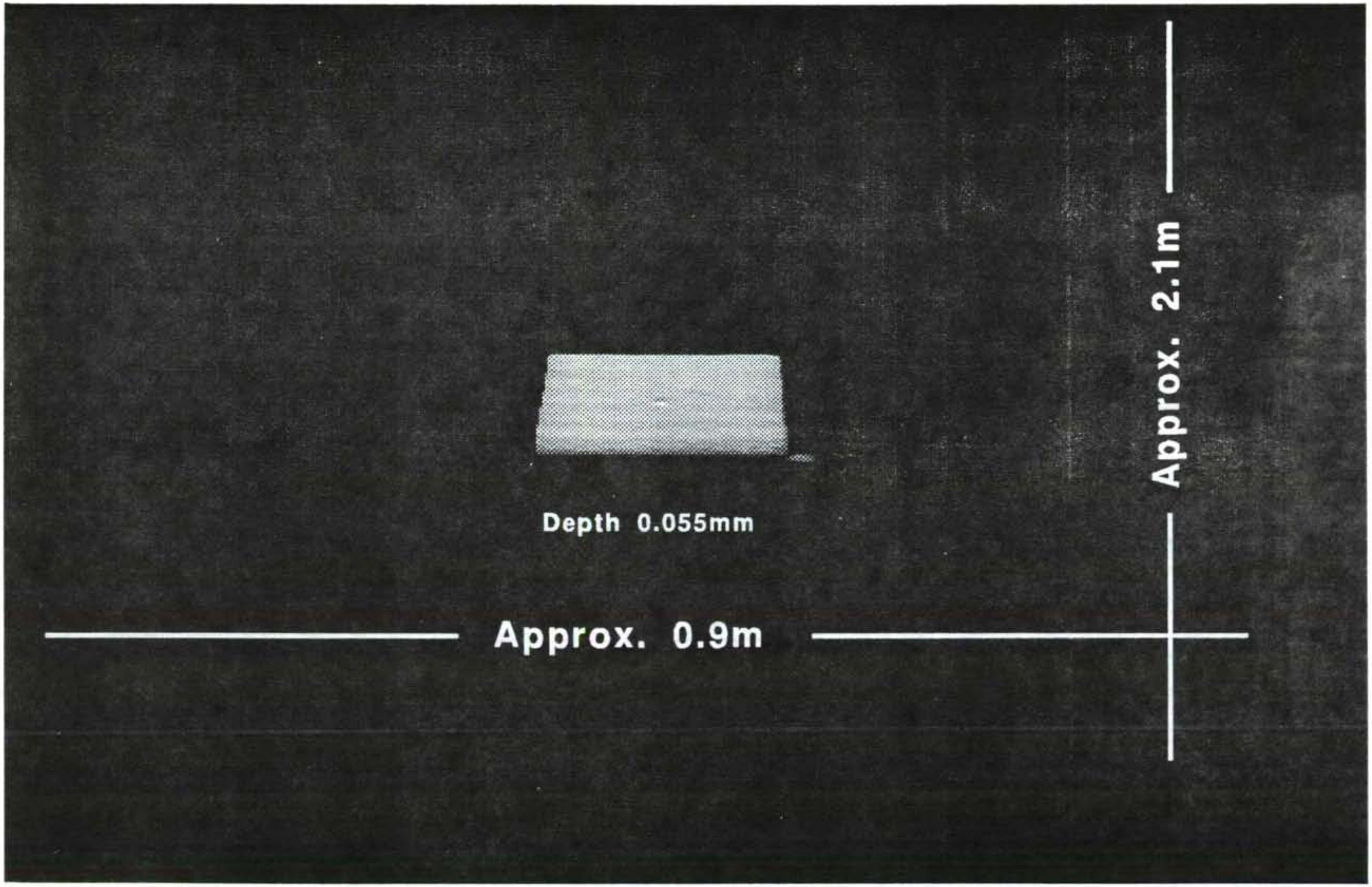

Figure 3. D-Sight image of an impacted specimen.
Field of view 0.9 m wide by 2.1 m deep.

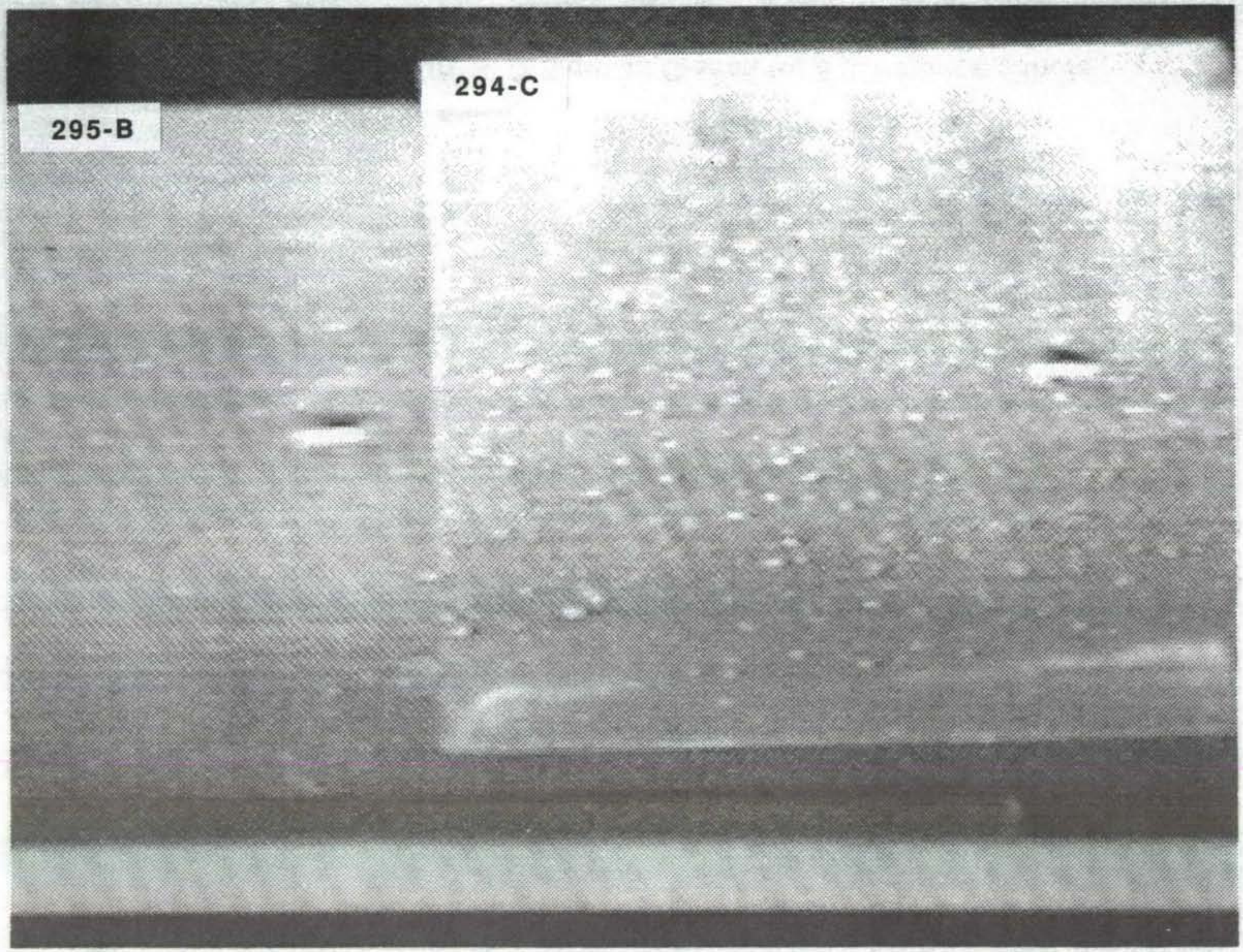

Figure 4. D-Sight image. Specimen 295-B not painted, 294-C painted with CF-18 paint finish.

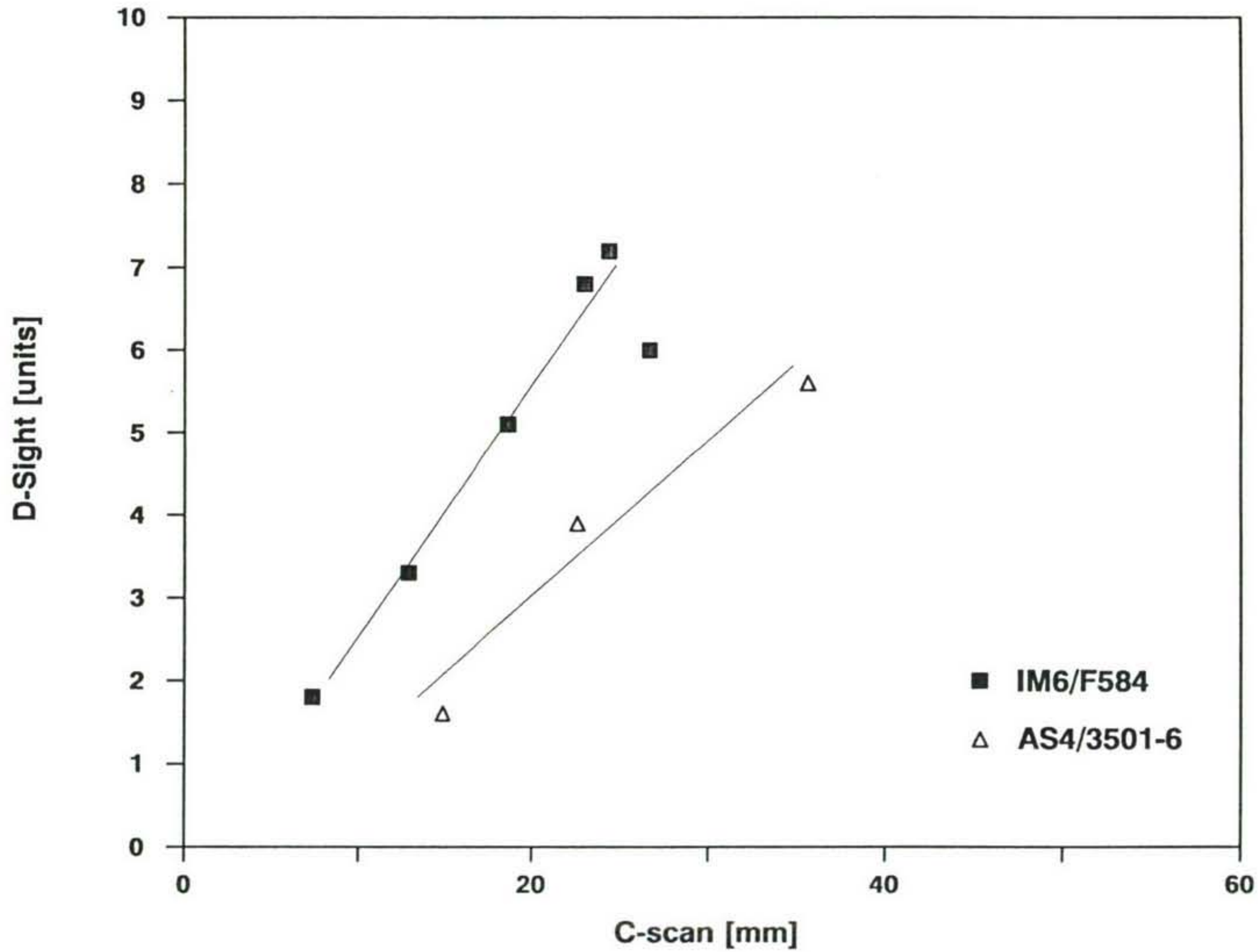

Figure 5. D-Sight vs C-scan for 8 plies thick panels

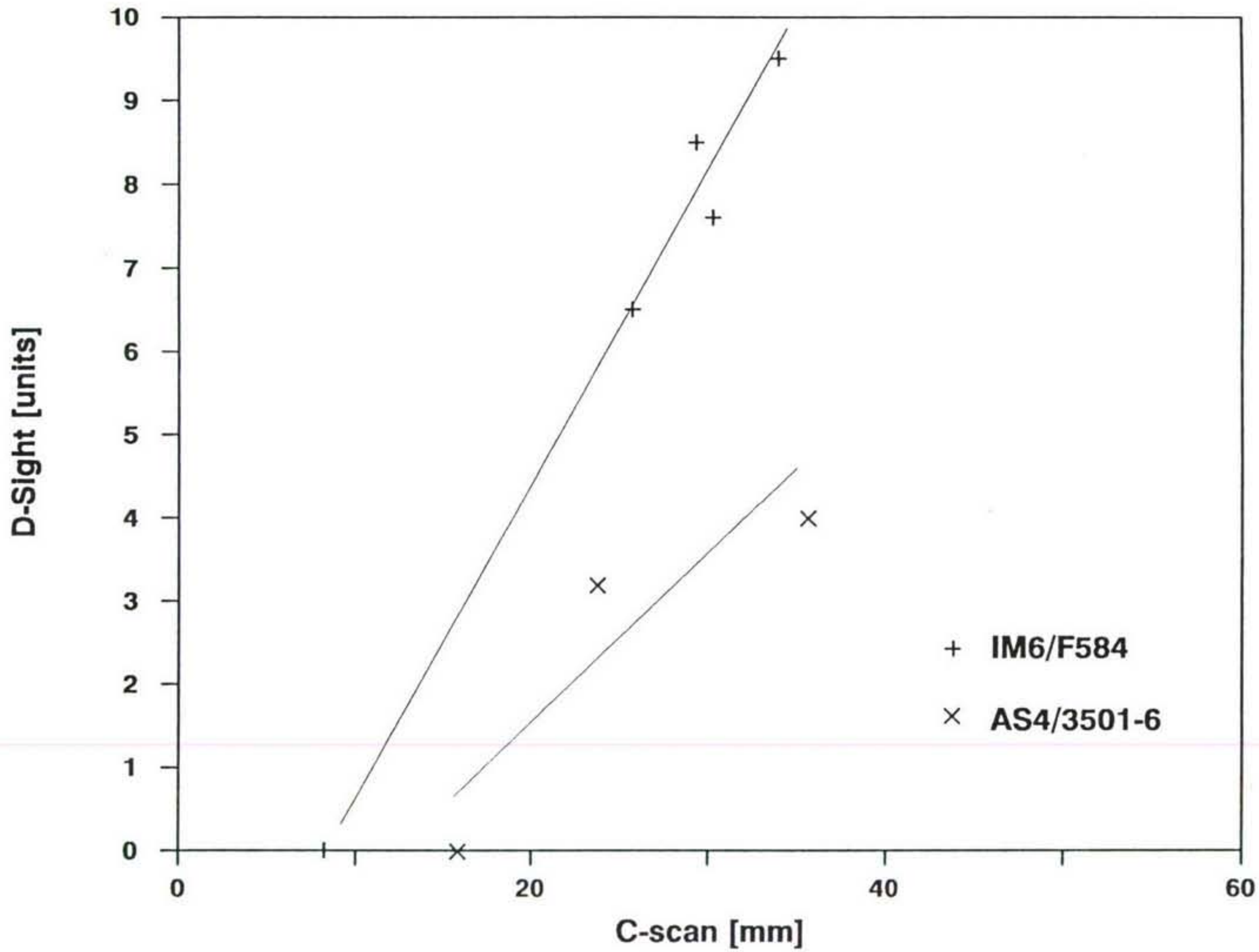

Figure 6. D-Sight vs C-scan for 24 plies thick panels

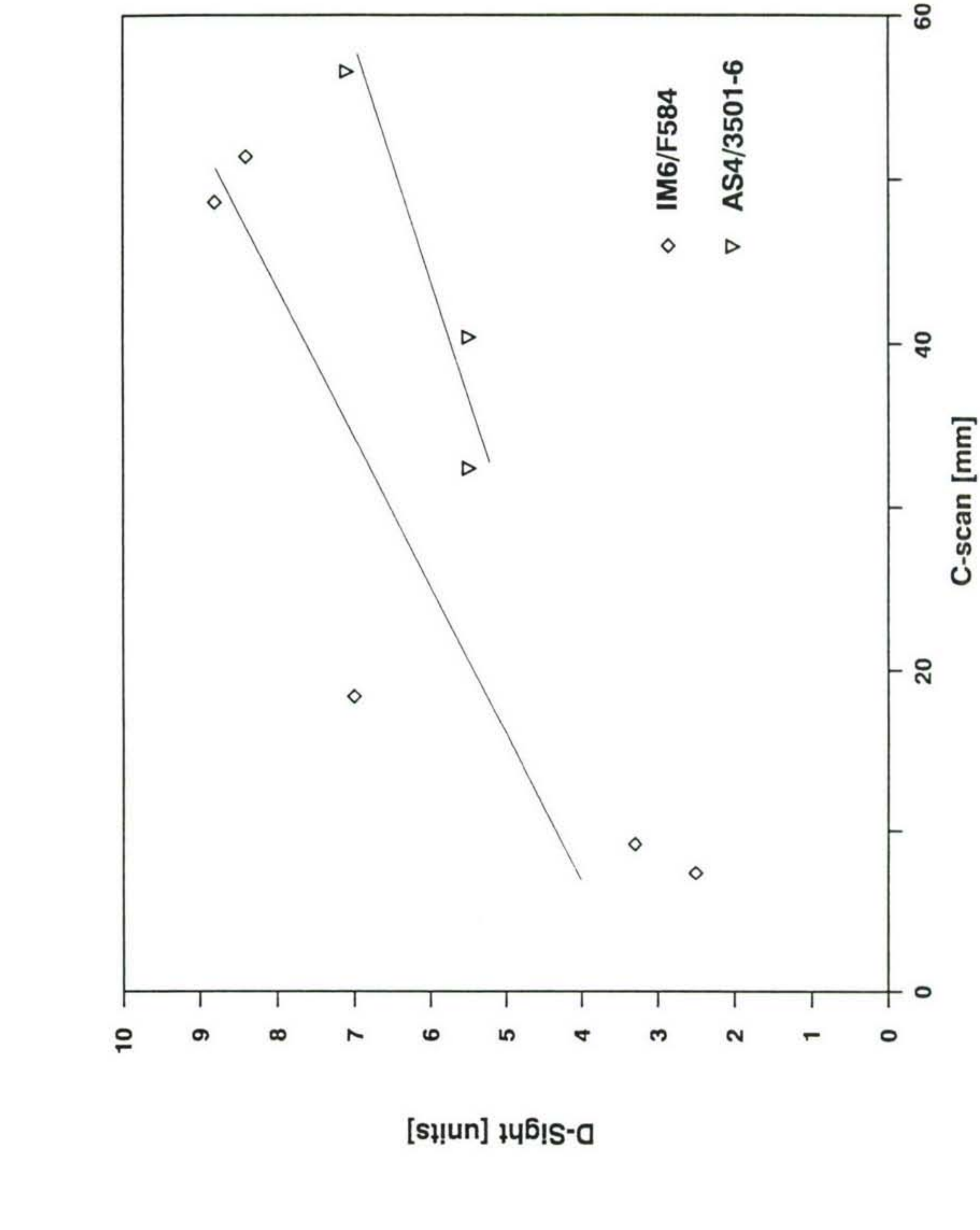

Figure 7. D-Sight vs C-scan for 48 plies thick panels

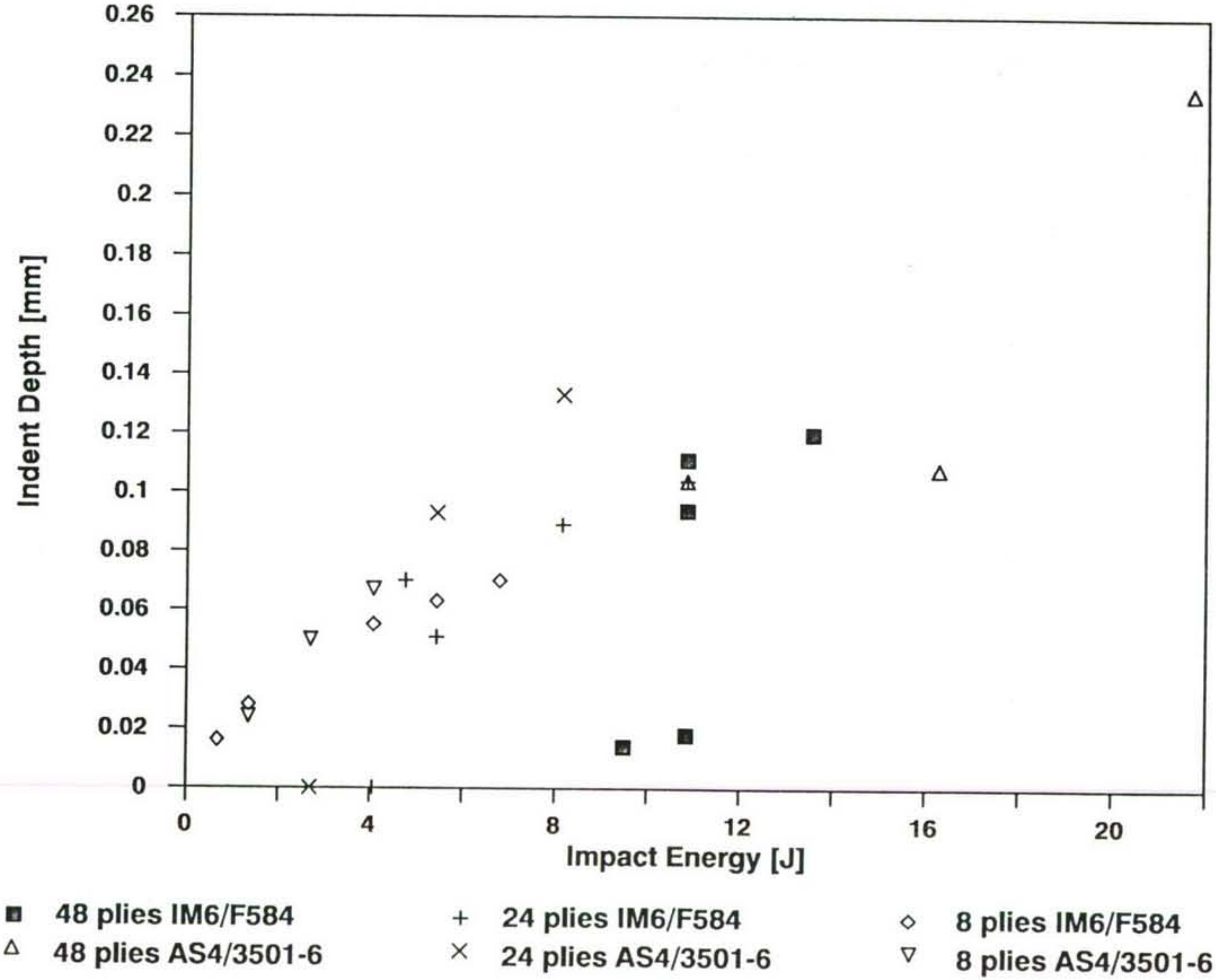

Figure 8. Indent depth vs impact energy

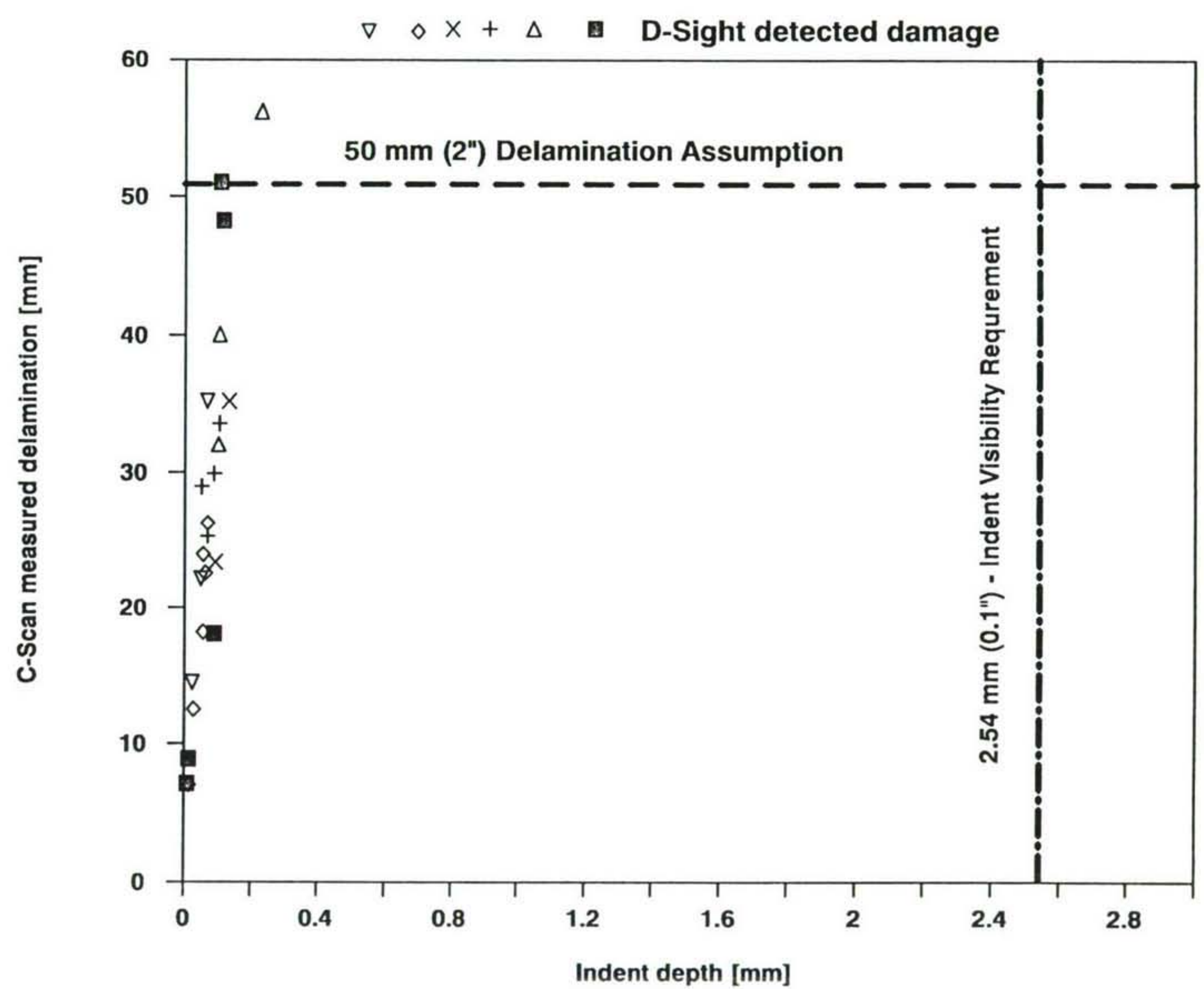

Figure 9. D-Sight and damage assumptions in draft USAF Damage Tolerance Design Requirements [10]

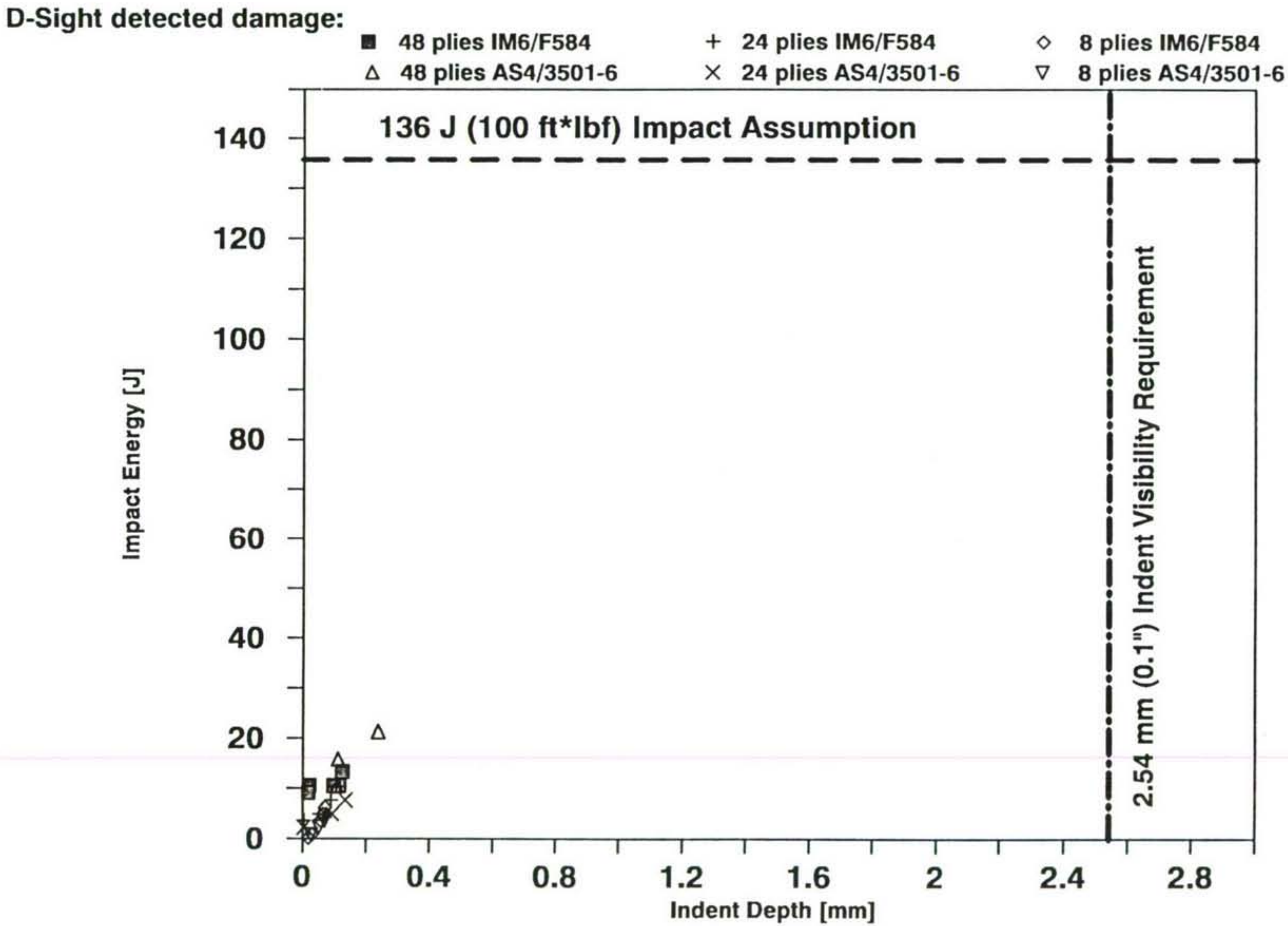

Figure 10. Impact energy vs indent depth and damage assumptions in draft USAF Damage Tolerance Design Requirements [10]

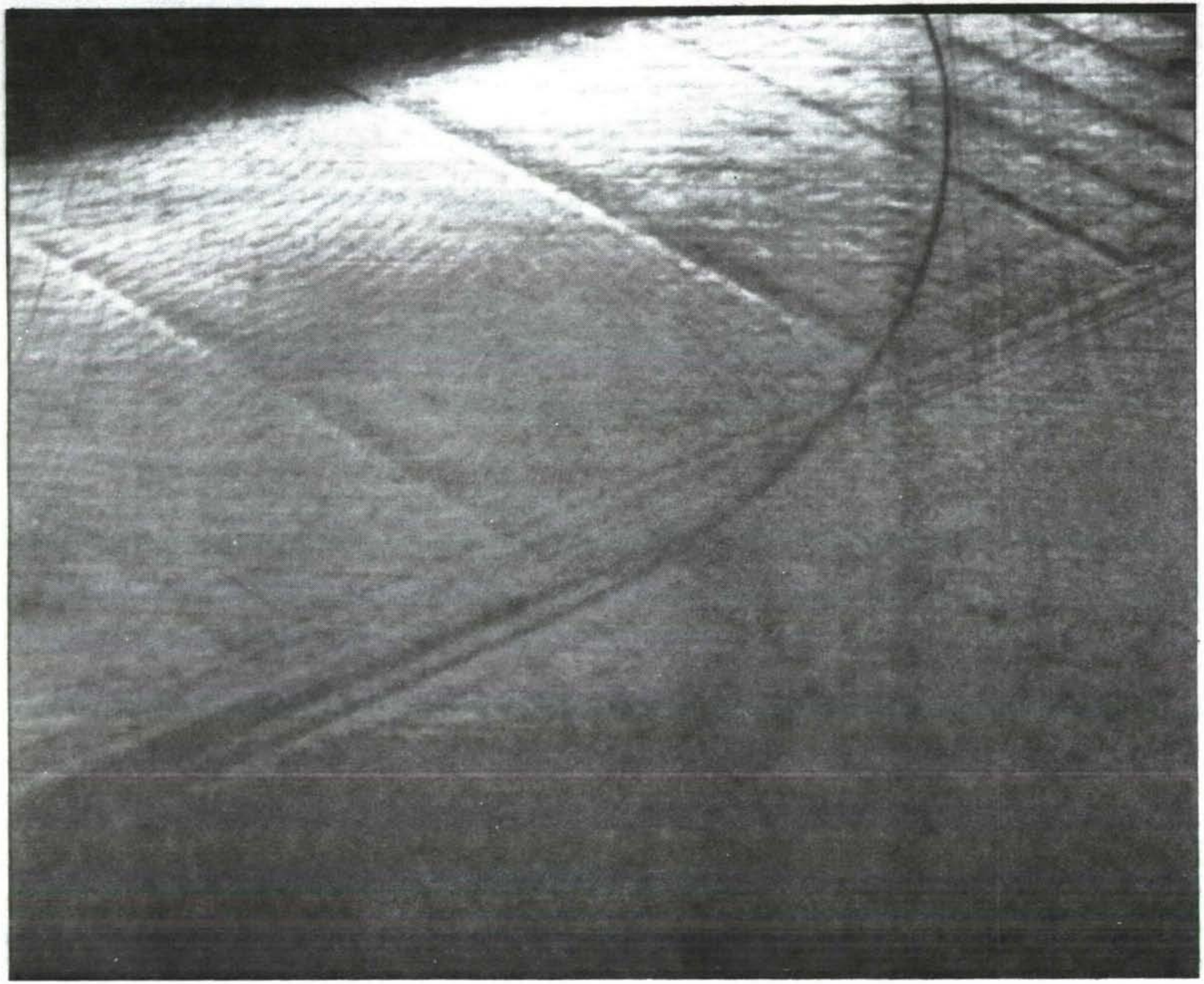

Figure 11. D-Sight image. CF-18 horizontal stabilator detail.

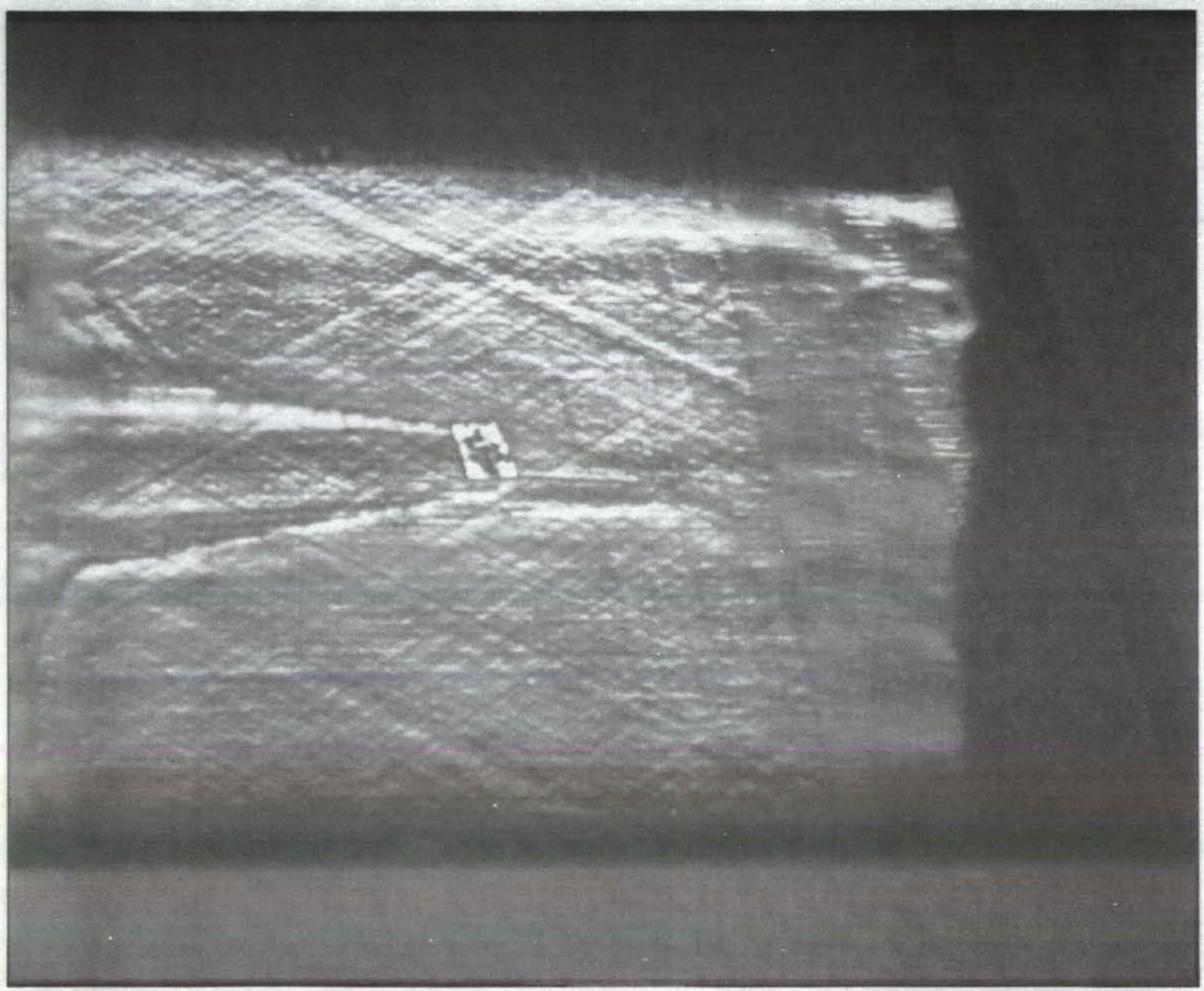

Figure 12. D-Sight image. CF-18 trailing edge flap detail.

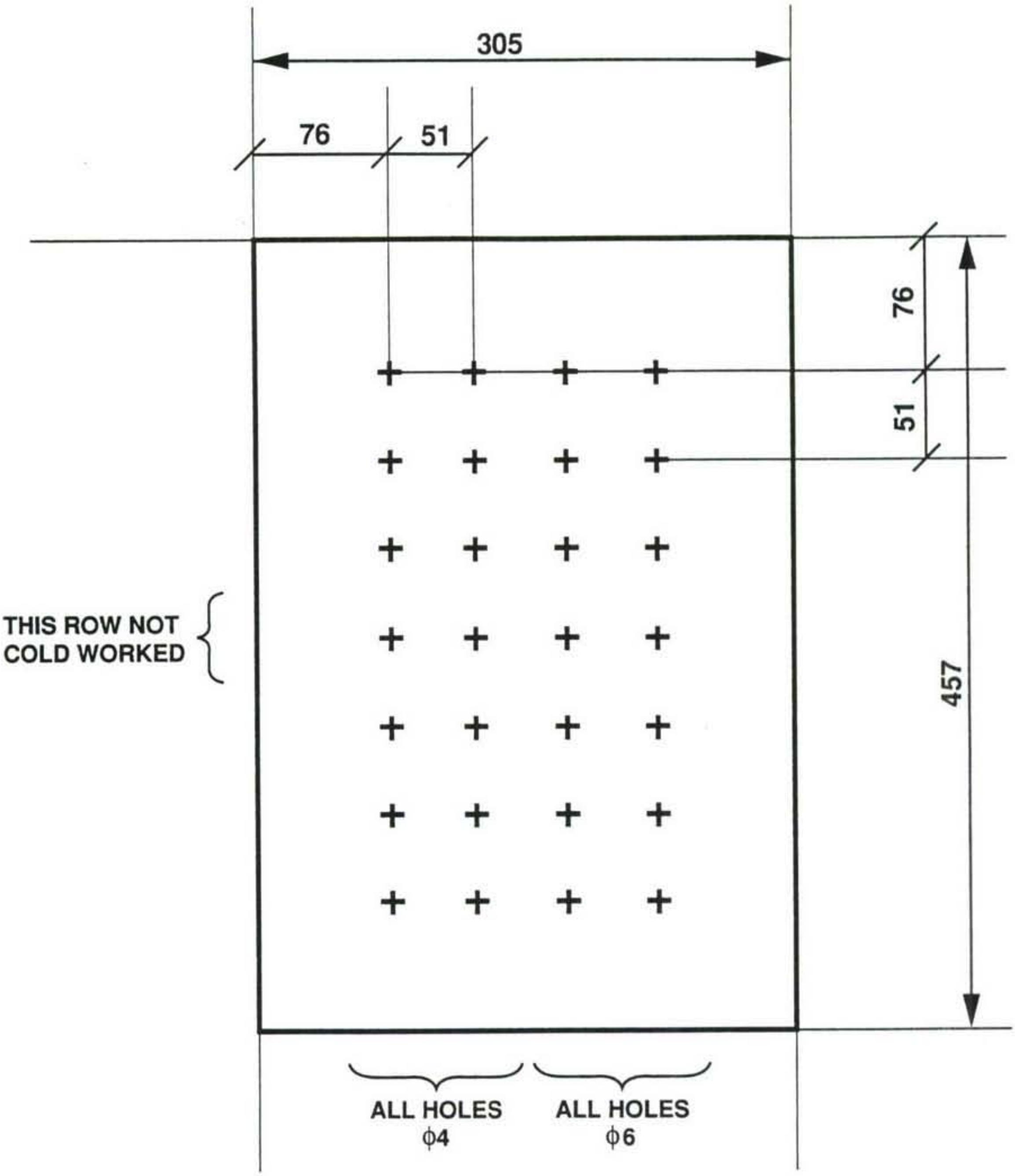

FIG. 13: 707576 SPECIMEN WITH COLD WORKED HOLES. PLATE THIKNESS 9.5mm

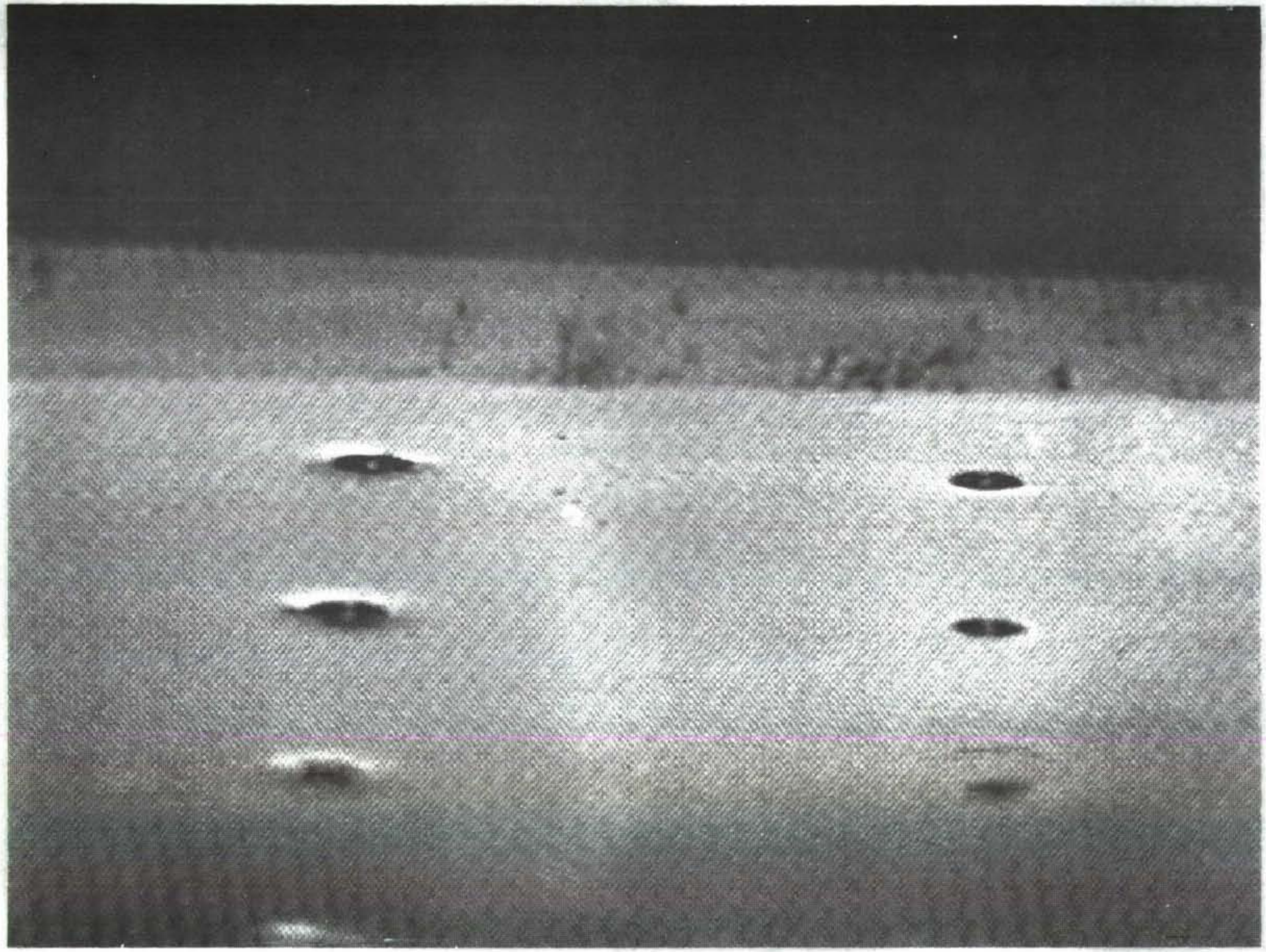

Figure 14. D-sight image. Note the distinct halo around cold worked holes on the left.

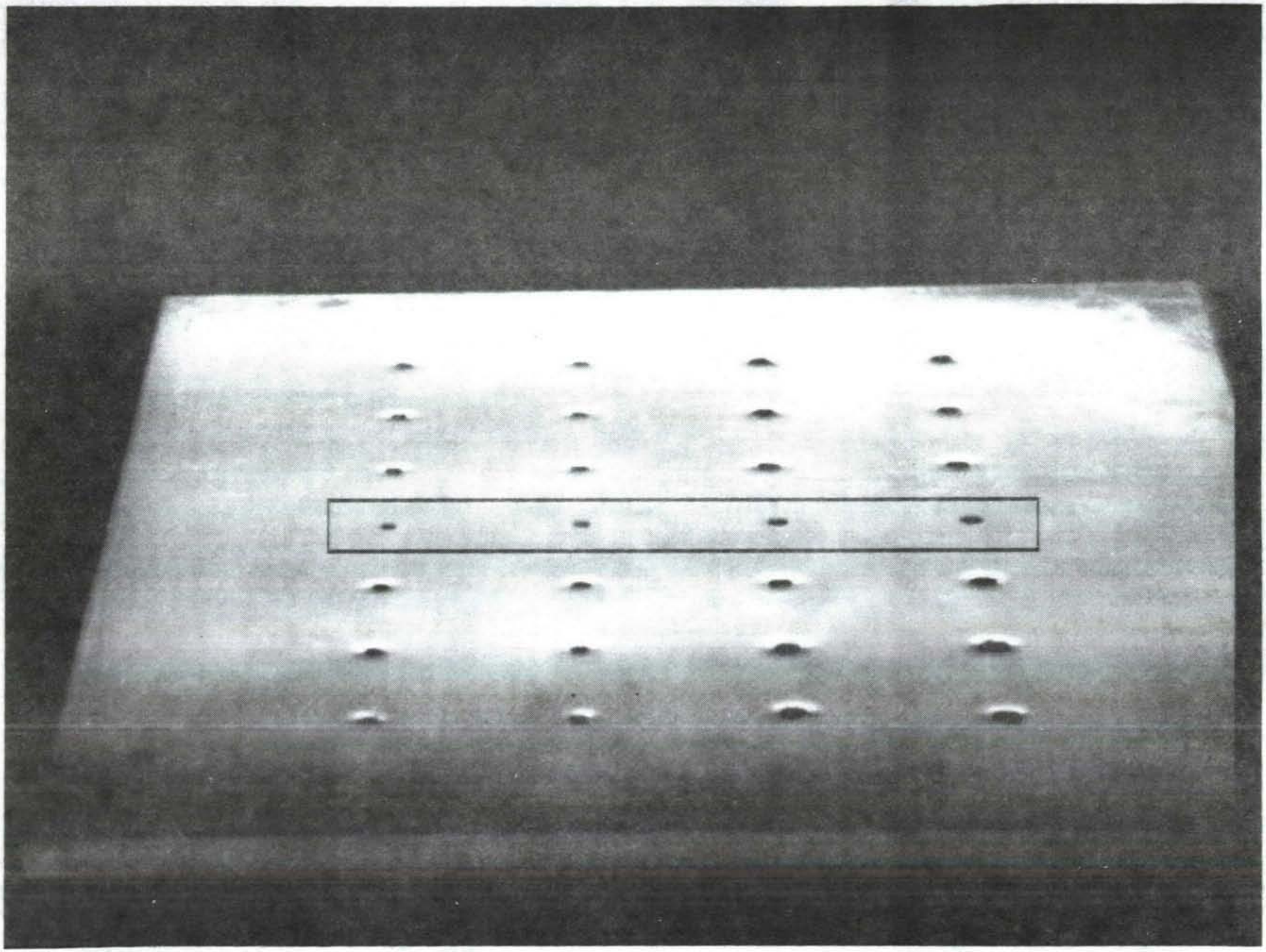

Figure 15. D-Sight image. 7075T6 specimen with cold worked holes.

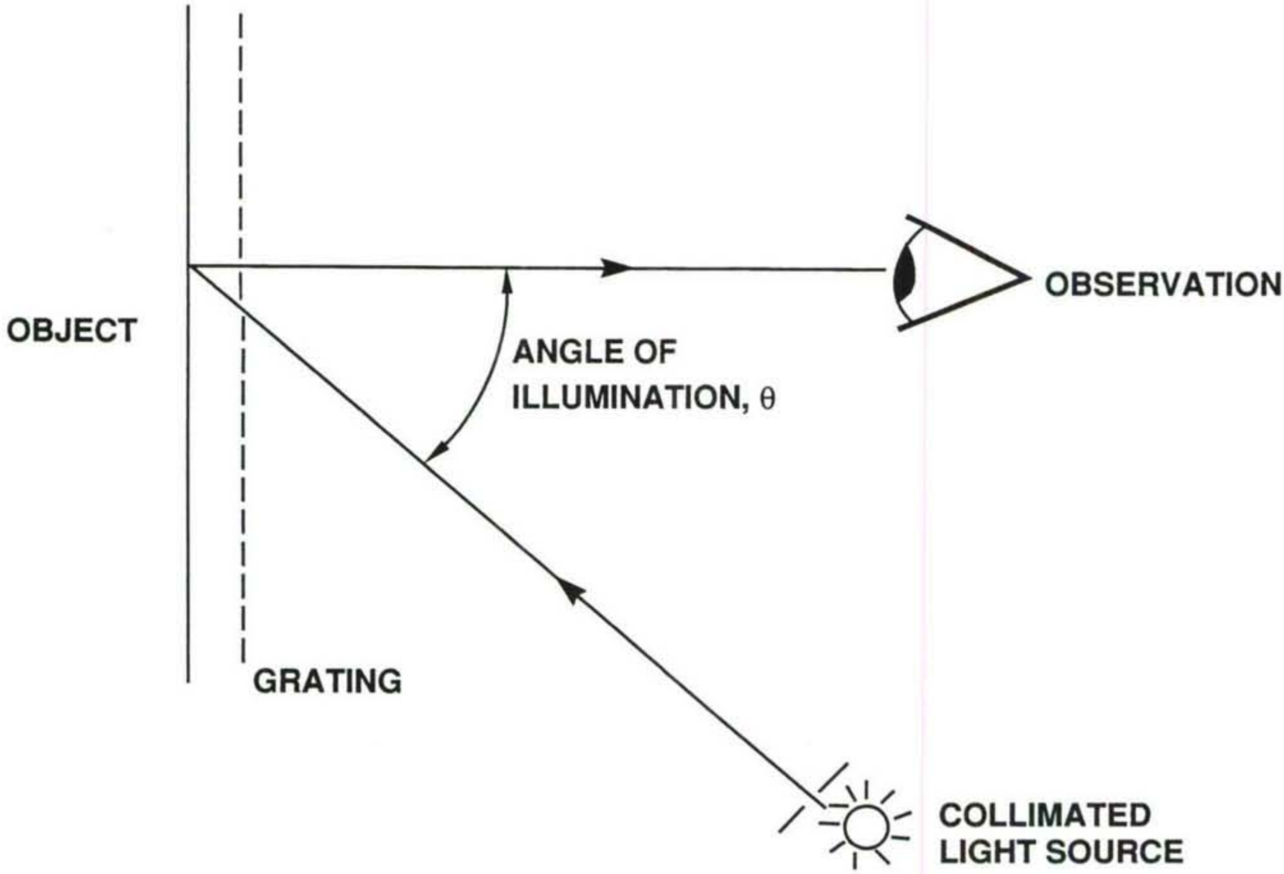

Figure 16. Set-up geometry for Shadow Moire inspection of an object

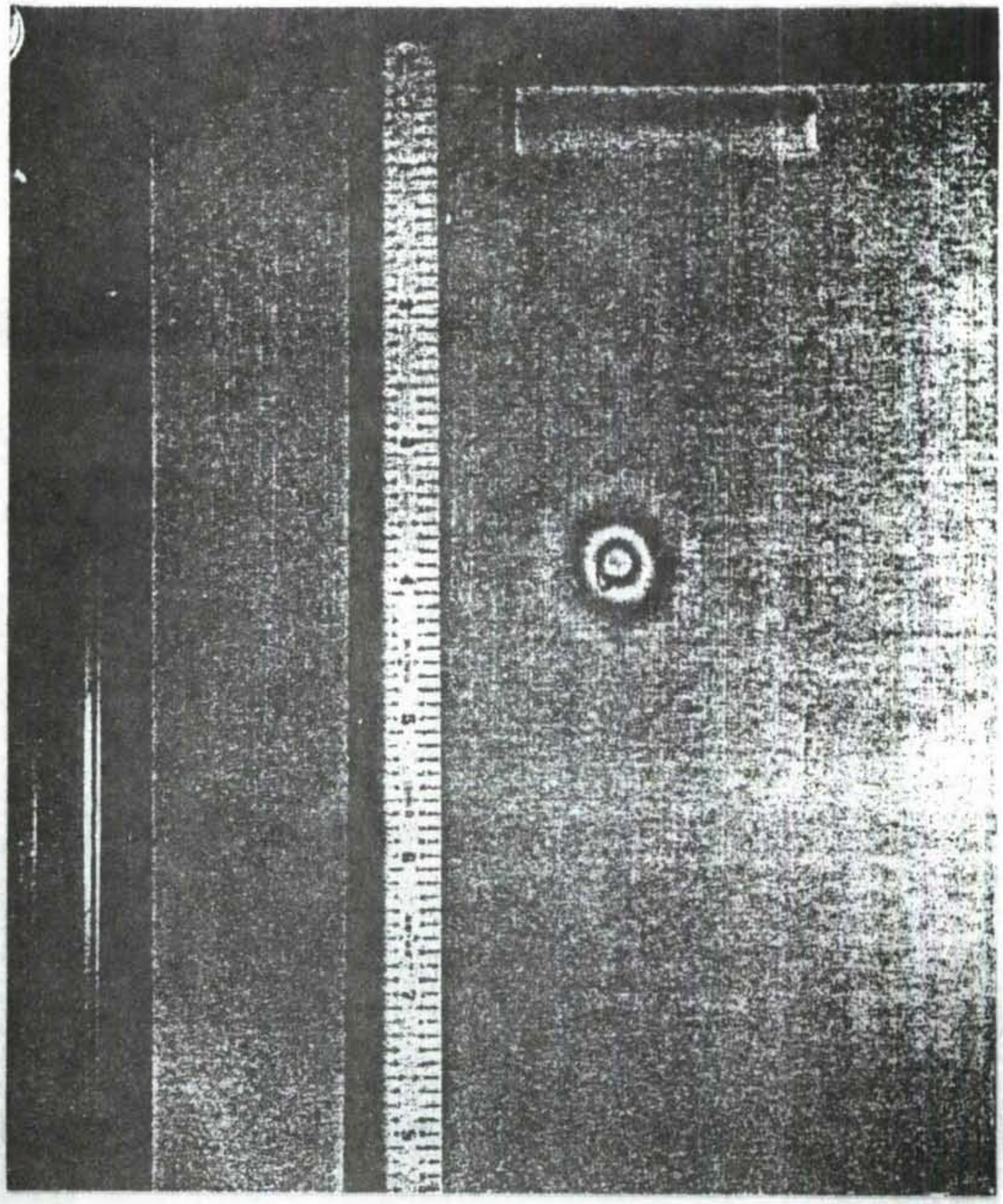

Figure 17. Shadow Moire image on panel 365B following 27J impact with a 25 mm diameter steel impactor. θ = 45°; 7.87 lines per mm.

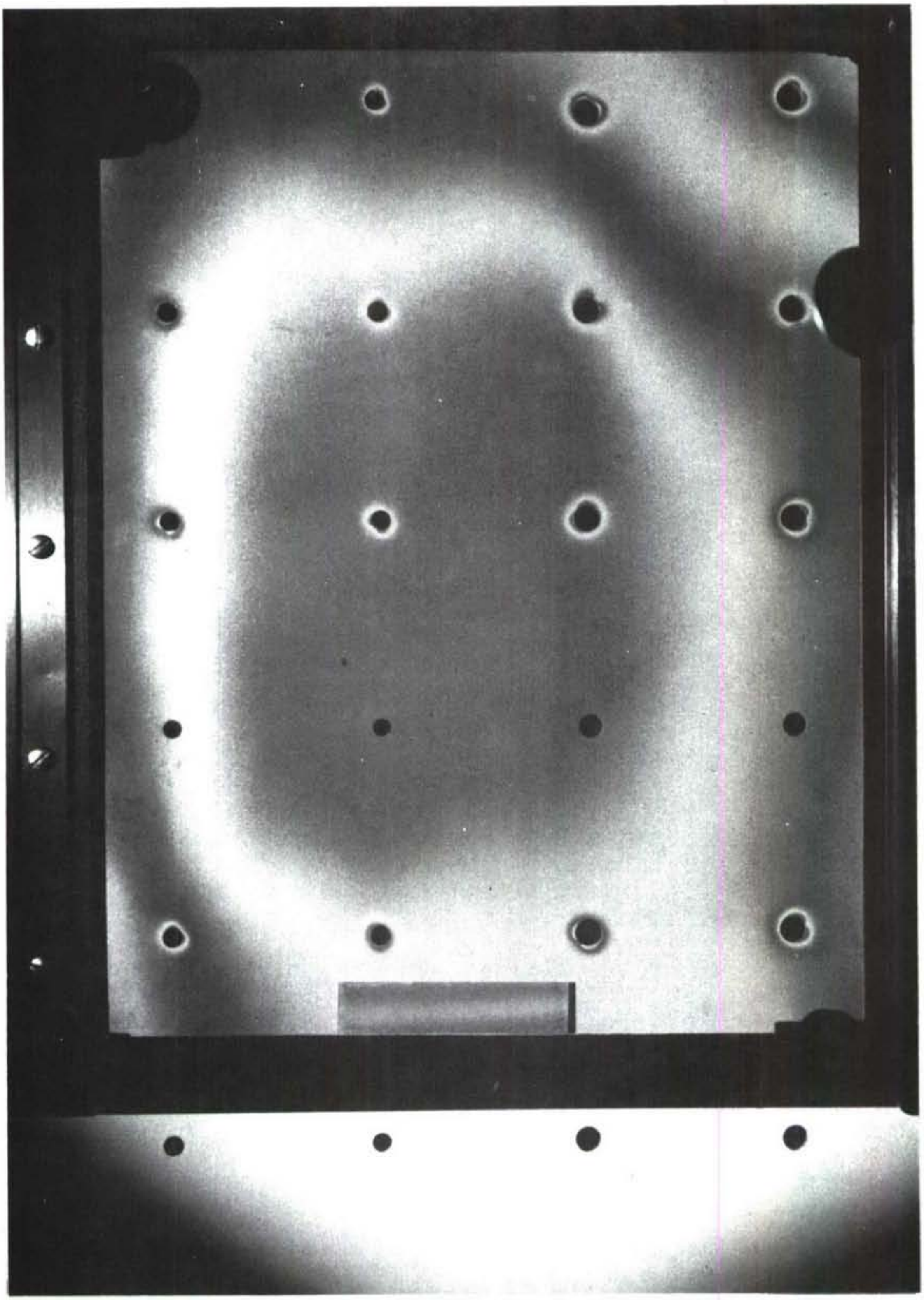

Figure 18. Shadow Moire image of 7075T6 specimen with cold worked holes. θ = 45°; 7.87 lines per mm.

"New Magneto-Optic/Eddy-Current Inspection Methods For Aging Aircraft"

by

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I. Introduction

Present nondestructive testing techniques used on aging aircraft are in great need of improvement, especially those used for the detection of cracks and corrosion around rivets in lap joints and other critical areas [1-7]. The predominant existing techniques, visual inspection and eddy current inspection, are limited in their ability to provide rapid and accurate inspections of the large surface area of an airplane.

Visual inspection techniques have been responsible for detecting about 80% of the cracks [1,2]. However, this method is limited in its ability to detect the small, tight fatigue cracks emanating from rivets that are of immediate concern. Only after these cracks have linked and become large are they generally found by visual means. One test showed that a crack 3.5 inches long had about a 50 percent probability of being detected visually [3].

Eddy current techniques are used extensively to inspect small areas around individual rivets in riveted lap joints. These techniques are time consuming and tedious, requiring skilled operators to interpret sometimes ambiguous results [1, 5-7]. Painted areas cause difficulty due to lift-off of the probe coil, especially as the paint becomes thicker from additional coats. The paint may also cover the rivets so well that they are often difficult or impossible to locate.

In a typical conventional eddy current inspection of the type used on a fuselage lap splice, a small probe coil is traced around an individual rivet by hand using a template as a guide. If care is not taken in centering the template around the rivet, which can be very difficult to do when the rivet is covered by paint, erroneous flaw indications can easily result. The correct selection of probe coils, template hole sizes, and test frequencies for such inspections is very important. Newer sliding-type eddy current probes are being developed, but these still suffer from lift-off problems. They may also miss cracks which are normal to the line of rivets.

Most conventional eddy current instruments display variations in the complex impedance seen by the probe coil (corrected for lift-off) caused by defects such as cracks, zones of metal fatigue or corrosion. These signals are not always easy to interpret since they look nothing like the flaws themselves and since different types of flaws can sometimes give nearly identical signals. Simple units having only meter indications (as opposed to impedance plane displays) give even less detailed information on the characteristics of the flaw.

This paper describes a new technology being developed by Physical Research, Inc., magneto-optic/eddy current imaging. We are developing instruments based on this technology as improved tools for inspection problems encountered in aging aircraft. The technology represents a new approach to the problem of detecting cracks and interpreting data in eddy current inspections.

Faster, simpler and more reliable inspection techniques are clearly needed to aid existing visual and eddy current methods. An acceptable, practical technique should have the speed and ease of interpretation of a visual inspection and reliability that meets or exceeds that of conventional eddy current inspections. The new magneto-optic/eddy current technology appears to be capable of meeting these criteria by offering a nearly ideal solution to these inspection problems.

Our approach is based on unconventional eddy current excitation, and exploits a new method of extracting the desired flaw information. A magneto-optic sensor is used to form images of flaws <u>directly</u> and in <u>real time</u>, which allows the device to cover relatively large areas in a rapid manner compared with conventional eddy current techniques. This new technology is also inherently less sensitive to lift-off than conventional eddy current methods and requires no lift-off corrections. In addition, since the output is in the form of a direct visual image, the entire examination can be recorded on low cost video tape.

Besides the advantage of very rapid inspections (6 to 9 rivets on a 737 lap joint can be viewed at one time) the magneto-optic/eddy current device output readily lends itself to analysis using image processing algorithms to provide an automated interpretation of flaws if desired.

II. Results

The new magneto-optic/eddy current device, pictured in Figure 1, is compact, hand-held and light weight. At present, the prototype unit consists of a special power supply/controller which is connected to an optical head through a power cable. Images can be viewed directly, or optionally, the output can be readily converted to standard television format for viewing and recording.

The device allows the inspector to visualize the magnetic fields caused by the disruption of induced eddy currents due to rivets or flaws. Detection is accomplished by a sensor exhibiting the Faraday magneto-optic effect. Images appear directly on the three inch diameter crystal. Because the device employs an inductively coupled current source, paint or other coatings need not be removed for inspections. Moreover, a single imaging sensor may be used for a wide range of different inspection frequencies, from zero (DC) to over 100KHz. This means that only one sensor is needed for all inspections of flat or slightly curved surfaces of the type encountered on a typical aircraft outer skin.

The images reproduced in the accompanying illustrations, Figures 2-5, are of fatigue cracks in aluminum alloys typical of those used in the aircraft industry. The crack specimens were provided by Fatigue Technology, Inc. and Douglas Aircraft, Inc.

In recent tests, the device has also been able to produce excellent images of cracks in riveted lap joint sections. The device, which can be fitted with 25 feet or more of cable, has been tested on several commercial aircraft. Rivets on aircraft are very rapidly and easily imaged.

Test inspections have indicated that crack images can be seen in some cases through nonconductive material which is far thicker than any conceivable paint layer (up to 0.1 inch at frequencies near 20 KHz). Moreover, images can even be made through conductive paint. At lower frequencies we have also been able to image corrosion-like targets through 0.050 inches of aluminum skin. Very tight fatigue cracks as small as 0.03 inches long have already been observed on the samples available to us, and the technology has yet to be pushed to its limits. It may be possible to detect even smaller cracks. Present indications are that cracks beneath rivet heads might be detectable using the right combination of current and frequency.

III. Summary and Conclusions

In summary, we believe that the new magneto-optic/eddy current technology is ready to be put to use in a number of different areas including inspection of aging aircraft and production quality assurance. The advantages include output in the form of direct, real time images; ease of use; rapid areal coverage; good sensitivity to large or small cracks; and low cost, complete

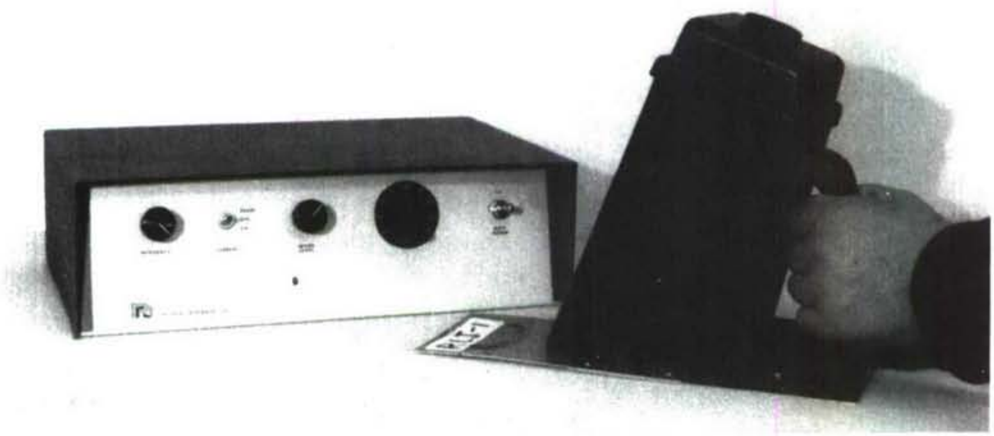

Figure 1. A picture of a prototype Magneto-Optic/Eddy Current imager.

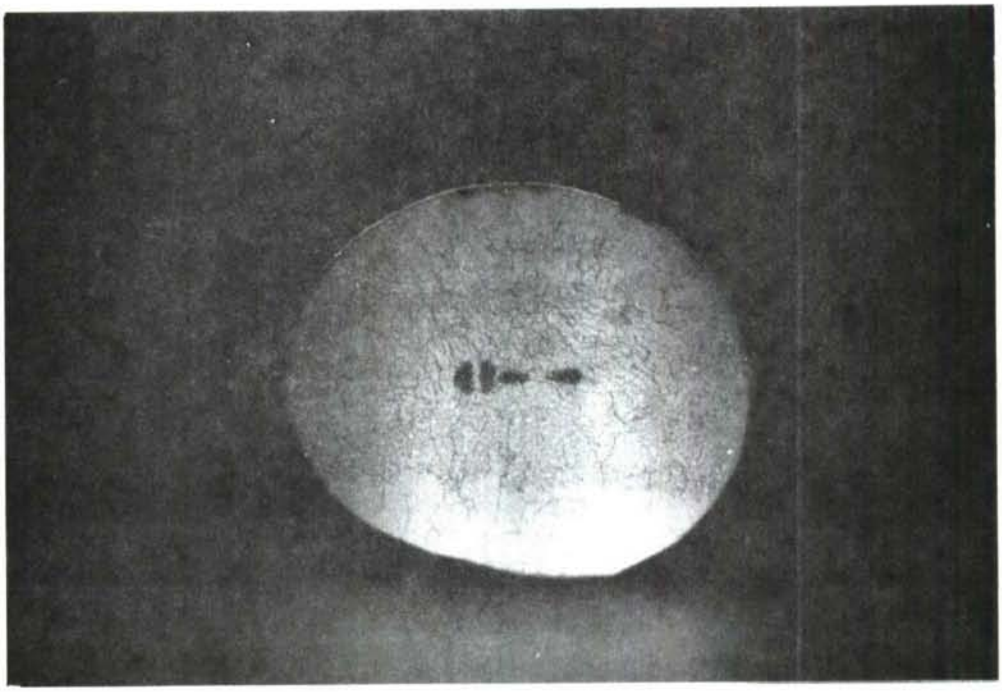

Figure 2. An image of a 0.50 inch long fatigue crack emanating from a 0.19 inch diameter hole in a 0.045 inch thick aluminum fatigue test specimen.

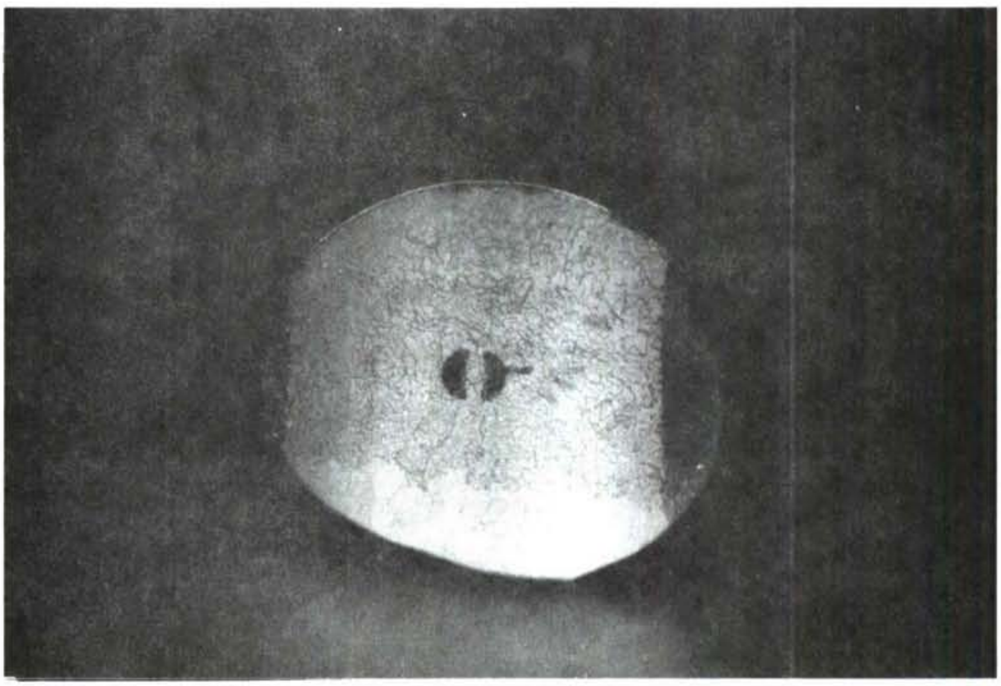

Figure 3. An image of a 0.18 inch long fatigue crack emanating from the edge of an 0.45 inch diameter aluminum rivet in an 0.3 inch thick aluminum fatigue test specimen.

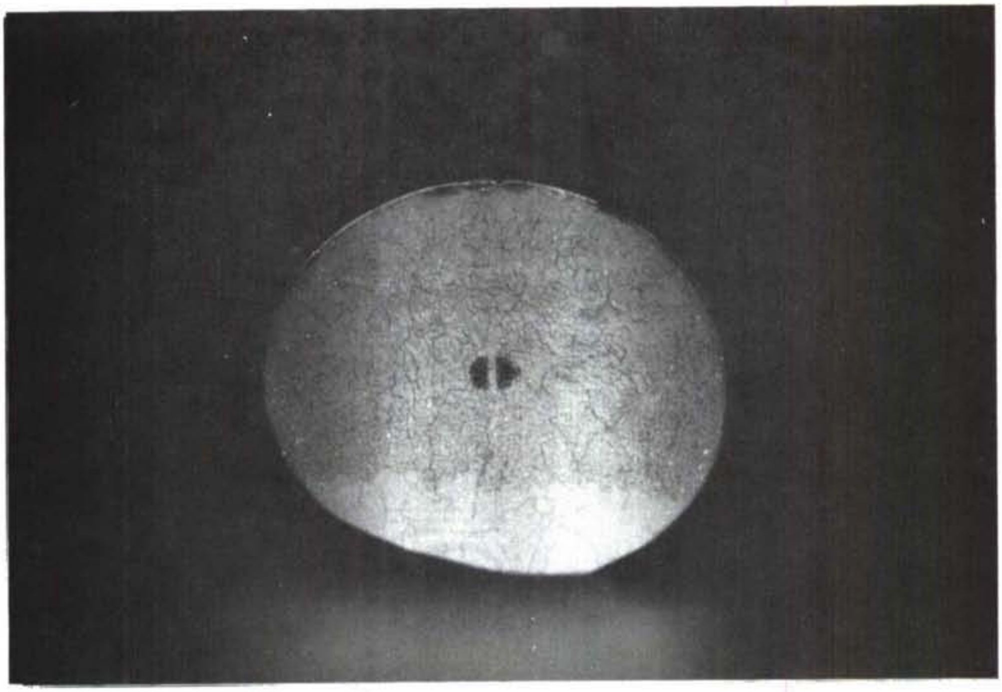

Figure 4. An image of a 0.05 inch long fatigue crack emanating from a 0.19 inch diameter hole in a sheet of aluminum 0.045 inches thick.

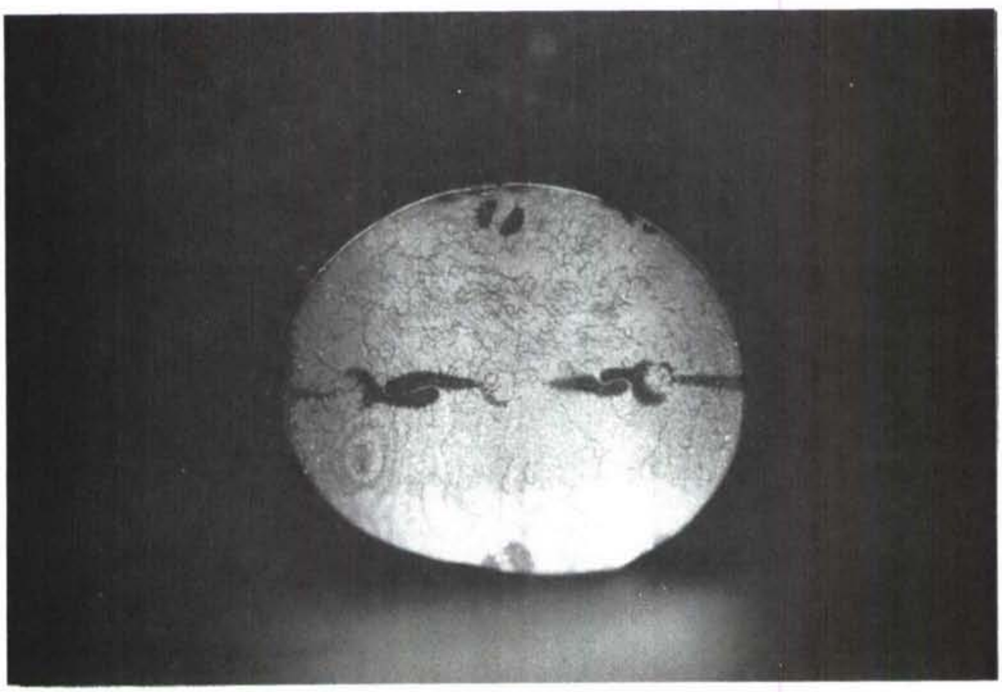

Figure 5. An image of a lap splice fatigue crack showing multiple overlapping, disconnected cracks.

documentation of inspections. Prototype units will be available in the immediate future (December 1989) for evaluation and qualification by airframe manufacturers and airlines. We anticipate production units being available during the first quarter of 1990.

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HIGH CYCLE FATIGUE ENDURANCE LIMIT TESTING OF TITANIUM MATERIALS

CAPTAIN VANCE JOHNSON
PROPULSION SPO
WRIGHT-PATTERSON AFB, OH

WORDS TO BE USED IN THE PRESENTATION

"HIGH CYCLE FATIGUE ENDURANCE LIMIT TEST OF TITANIUM MATERIAL"

Captain Vance Johnson, Aeromechanical Engineer, ASD/YZEE

TITLE PAGE Self explanatory

CHART 1 (Overview) This chart gives an overview of the six major subtopics to be presented. First, I present a brief background of what caused us to conduct the high cycle fatigue (HCF) test of titanium material. Secondly, current testing requirements and procedures throughout the United States will be mentioned. Thirdly, the results of a HCF test conducted this year (1989) is the main crux of the briefing. Fourth, I will indicate what future testing needs to be conducted and where we go from here. Fifth, I will state what possible implications this testing may have in terms of changing the requirements of the Engine Structural Integrity Program (ENSIP).

CHART 2 (Background Information) I have always found it easier to follow a briefing and become interested in the topic if I can relate to a "real world" problem and understand what drives people to do the things they do. Therefore, I offer brief background information in this subtopic to obtain my audience's interest and understanding of why we conducted a high cycle fatigue test of titanium material. This chart will be used to mention that we are experiencing similar problems with titanium structures throughout several engine lines and different prime contractors (in a generic sense, no names mentioned). Also mentioned will be the fact that these problem areas are so difficult to pinpoint due to the fact that it does not show itself to be a chronic problem with every part or in all engines. In fact, of the thousands of parts flying around out there, only a small number of parts have shown cracking in very few engines. These few number of failures makes solving the problem very frustrating and no less important because critical failure of these components will bring airplanes down.

CHART 3 (Compressor Blade and Diskpost Model) This chart is a finite element analysis model of a particular compressor blade and diskpost that shows the location where cracks occur. (No engine line or prime contractor name will be mentioned). The cracking occurs just above the pressure face in the blade dovetail and diskpost region where the radii begins. The finite element analysis on this model and another similar HCF problem area from a different prime contractor, both predict steady state stresses to be quite high. However, I will reiterate the fact that only few parts fail, therefore high steady state stresses is not the simple answer to these problems and that something else must be a contributing factor in these failures.

The purpose of this chart is to get the audience tuned into a real world problem so they can relate back to it as the briefing progresses. These finite element pictures were cut out of the

analysis section that predicted stresses at every element node. The stress data itself will not be shown because it does not add to the purpose of the briefing and although the data is not proprietary, I feel it certainly should be considered "sensitive" and kept between the U.S. Air Force and the General Electric Company.

CHART 4 (Campbell Diagram) This chart shows a typical Campbell diagram from an aeromechanical test of the previously shown blade conducted at Arnold Engineering Development Center (AEDC), Tullahoma, TN. The point of this chart is to show that no high vibratory stresses can be demonstrated and that these parts should not be failing and whatever is causing these problems is a "tough nut to crack"!

I do not feel this data to be considered very "sensitive" because it does not show anything significant and that is my point for using this chart. (Point of clarification for your understanding - this data was produced by AEDC under an Engineering Assistance to Production Services (EAPS) contract between them and General Electric Company. However, EAPS funds are provided to G.E. by the U.S. Air Force to be used for solving production type problems. Therefore, General Electric Co. has no proprietary claim to this data.)

CHART 5 (Current Materials Testing Requirements) This is the first chart of the second major subtopic. The purpose of this chart is to state that current material testing standards in this country is to conduct fatigue tests to ten million (10^7) cycles for ferrous (iron) alloys and halt testing for nonferrous alloys at thirty million ($3X10^7$) fatigue cycles. Runout is a term used to indicate that a particular fatigue test specimen did not demonstrate a crack within the standard 10^7 or $3X10^7$ fatigue cycles and the test was halted for that specimen.

CHART 6 (S/N Curve for Ferrous Materials) The stress versus number of fatigue cycles (S/N) curve in this chart is copied from Mil Handbook Five and shows that the slope of the curve goes asymptomatic by 10^7 fatigue cycles. This indicates that current testing practice for steels is adequate. A point will be injected that ferrous materials have been around a long time and their material characteristics well understood.

CHART 7 (S/N Curve of Titanium Alloy) This curve, also copied from Mil Handbook Five, shows that the slope is still negative at 10^7 fatigue cycles with only one data point indicating a runout has occurred. This indicates that the endurance limit for titanium materials is beyond 10^7 and maybe the characteristics of these alloys are not that well understood and current testing standards not adequate.

CHART 8 (TI Material) The S/N curve not going asymptomatic in the previous chart concerned us and caused us to ask several of our prime engine contractors (GE, P&W, Garrett, Rolls Royce, Allison) if they had any experience with titanium material beyond ten million fatigue cycles. This chart shows the response we received from a prime

contractor (Pratt & Whitney, but not to be mentioned) which really caught our attention. This S/N curve shows a negative slope even at one hundred million (10^8) fatigue cycles. This supported our theory that just maybe the answer to the unexplainable field problems is simply due to the endurance limit of titanium continuing to drop until it crosses with the low vibratory stress levels we are measuring but at much greater number of fatigue cycles than current testing procedures obtain. These last two charts really should give the audience an understanding of why we conducted a high cycle fatigue endurance limit test. However, I will not pull any wool over my audience's eyes, so I will mention the fact that this particular prime contractor was conducting fatigue experiments of TI material combined with fretting characteristics and this curve was plotted based on those results. Therefore, this curve should not be directly compared with the previous curve, but the fact remains that this curve for titanium material is not asymptomatic.

CHART 9 (High Cycle Fatigue Results of a TI 8-1-1 Test Beyond 10^7 Cycles) This chart introduces the third subtopic. The main points of the test we conducted are listed on this chart so people will be able to understand in the future when they are flipping through the briefing on their own. I probably will not mention the main points at this time and just use this chart to introduce the subtopic of a TI 8-1-1 test that we conducted this year.

CHART 10 (Fatigue Test of Titanium Engine Blade) This chart basically shows how the test was conducted. Again for the audience's understanding, I show a sketch of a blade with strain gages used for conducting a strain survey to know where the highest stresses are on the airfoil. I also show a hologram picture of the blade being vibrated in its' natural 3-stripe mode at approximately 7000 Hz frequency where the test was actually conducted.

CHART 11 (High Cycle (>>>10^7) Fatigue Results) This chart is the whole crux of the briefing. The actual data from the test is plotted on 7 cycle semi-log graph paper. A 10% reduction in fatigue endurance strength is shown beyond 10^7 cycles where normal testing standards would not have predicted. Ten percent reduction may not be the total answer to our problem but a contributing factor. A couple of underlying subtleties will be mentioned. First, this blade shaking in the 3-stripe mode cannot exactly be related in conventional material testing criteria of a certain R factor or A ratio, but the test was conducted in a fashion to get results without taking years of testing in more conventional manners. Secondly, even though we were conducting the test in a very high frequency mode, it still took six weeks of test time on just one specimen to get the data point to jump from one billion (10^9) cycles out to ten billion (10^{10}) cycles. This test would have taken years in a conventional manner using Material Test Systems (MTS) machines testing at a rate of a couple hundred hertz frequency.

CHART 12 (Future Endurance Limit Testing) This chart again just has the purpose of introducing the next major subtopic. Preliminary results are in, so where do we go from here in terms of testing to explore this endurance limit reduction in titanium alloys? I

summarize the future testing on this chart (mean load testing and alternating mean loads) but probably will not mention it at this time. I put these two bullets on this chart to benefit understanding for those who flip through the briefing charts at a future date.

CHART 13 (Goodman Diagram) This chart is used to show the interaction of steady state and vibratory stresses on design variables. Conventionally, a Goodman Diagram is used to show infinite life of a part at 10^7 fatigue cycles. We are questioning this design philosophy in terms of what effect 10^9 - 10^{10} fatigue cycles has on this diagram. Also, we do not feel anyone has a good handle on the lower right hand corner of the diagram (high steady state and low vibratory stresses) where low cycle fatigue (LCF) interacts with high cycle fatigue (HCF). The little diagram in the upper right corner will be used to show a future test load spectrum where a high cycle vibratory stress will be superimposed on top of a steady mean stress.

CHART 14 (F-16 Mission Profile) This chart shows the actual data from F-16 flight loads data recorder during a wild weasel mission in Germany. I'll use this chart to describe how the Air Force is actually flying their jets in terms of engine parameters and what effect these kinds of results should have on future material characterization tests. This data was generated by YZEE on a mission usage survey during October, 1988 and is unclassified.

CHART 15 (HCF/LCF Materials Test Load Spectrum) This chart ties together the F-16 mission profile by showing a future material test load spectrum we have in mind for alternating the mean stress (LCF) while having a superimposed high cycle vibratory stress. This type material testing should represent actual field usage as well as help give a better understanding of the HCF/LCF interaction; i.e. lower right hand corner of Goodman Diagram.

CHART 16 (Possible Test Apparatus for Combination LCF/HCF Material Testing) This chart shows a rough sketch of a vibration machine at the Oklahoma City ALC materials test laboratory. We have made initial contacts with these folks to help investigate this LCF/HCF phenomenon and this machine is one possible idea we had for conducting a titanium material test in a relatively short period of time. Essentially, we would rig up a hydraulic cylinder to cycle the mean loads while using the vibration machine's shaker table to apply the high cycle vibratory loads.

CHART 17 (Possible Future ENSIP Requirements) This chart not only introduces the fifth major subtopic but is the main chart used in describing how all this testing could affect future requirements in the ENSIP specification (MIL-STD-1783). Also, it touches on the fact that the advanced tactical fighter (ATF) SPO has already changed the specifications for their engine material characterization studies based on the results we obtained from our TI 8-1-1 test conducted this year.

CHART 18 (ENSIP Specification Tasks) This chart should be familiar to almost everyone in the audience since the ASIP and ENSIP programs are structured very much alike. I use this chart to tie all this testing

back into the specification requirements and show how it affects TASK II in the materials and processes design data charcterization studies.

CHART 19 (Summary) This is a summary chart to touch on the main points and conclude the briefing.

OVERVIEW

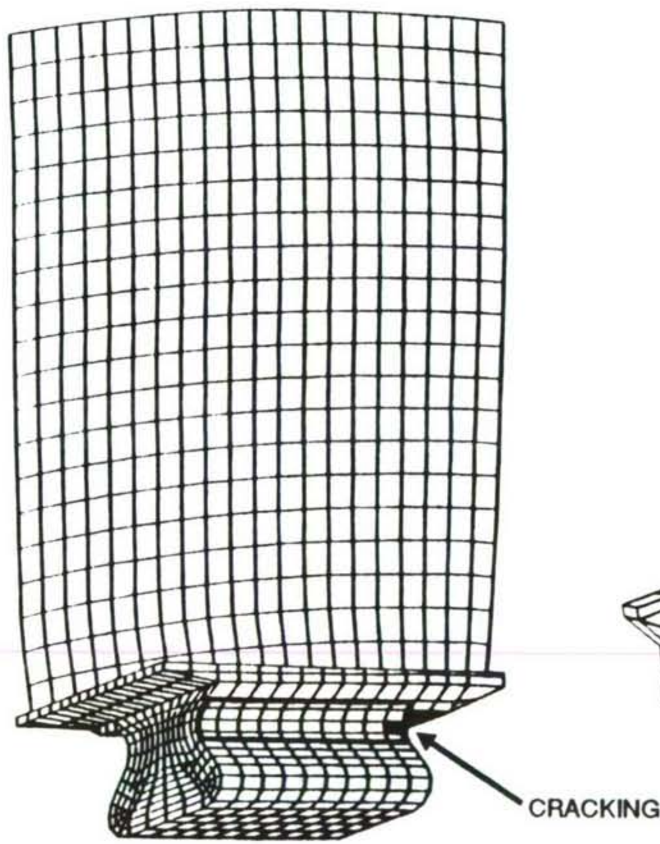
- BACKGROUND INFORMATION
- CURRENT TESTING REQUIREMENTS
- HIGH CYCLE FATIGUE RESULTS OF A TI 8-1-1 TEST BEYOND 10^7 CYCLES
- FUTURE ENDURANCE LIMIT TESTING
- POSSIBLE FUTURE ENSIP REQUIREMENT
- SUMMARY

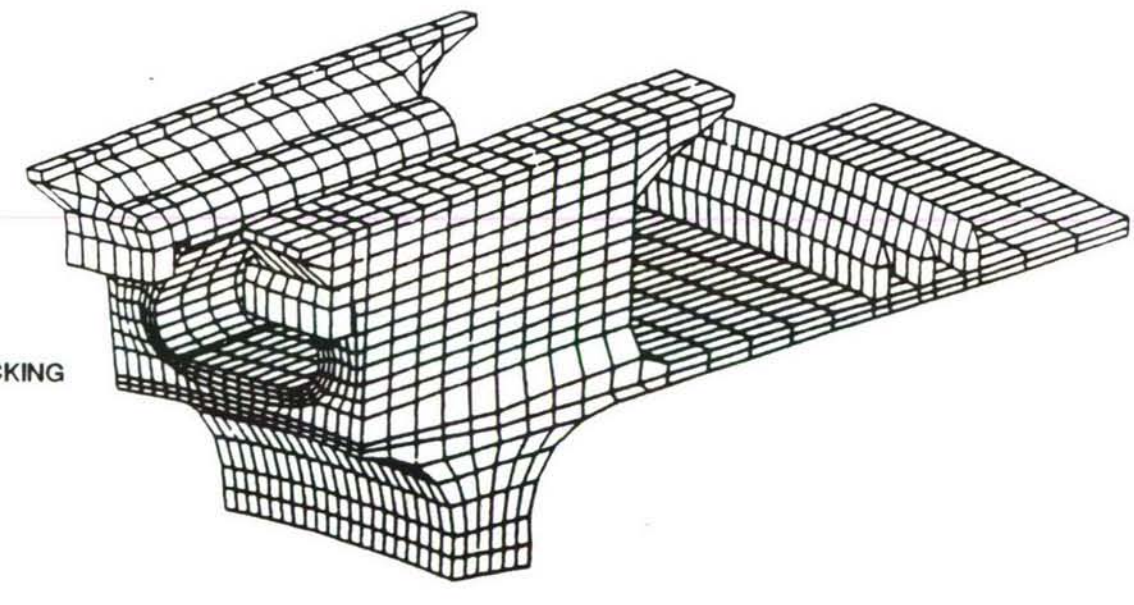
BACKGROUND INFORMATION

- EXPERIENCING UNEXPLAINABLE FATIGUE CRACKS IN TITANIUM PARTS THROUGHOUT SEVERAL ENGINE LINES
    - HIGH PRESSURE COMPRESSOR BLADES AND DISKS
    - FAN BLADES AND DISKS
- PROBLEM DOES NOT OCCUR IN EVERY PART OR IN ALL ENGINES

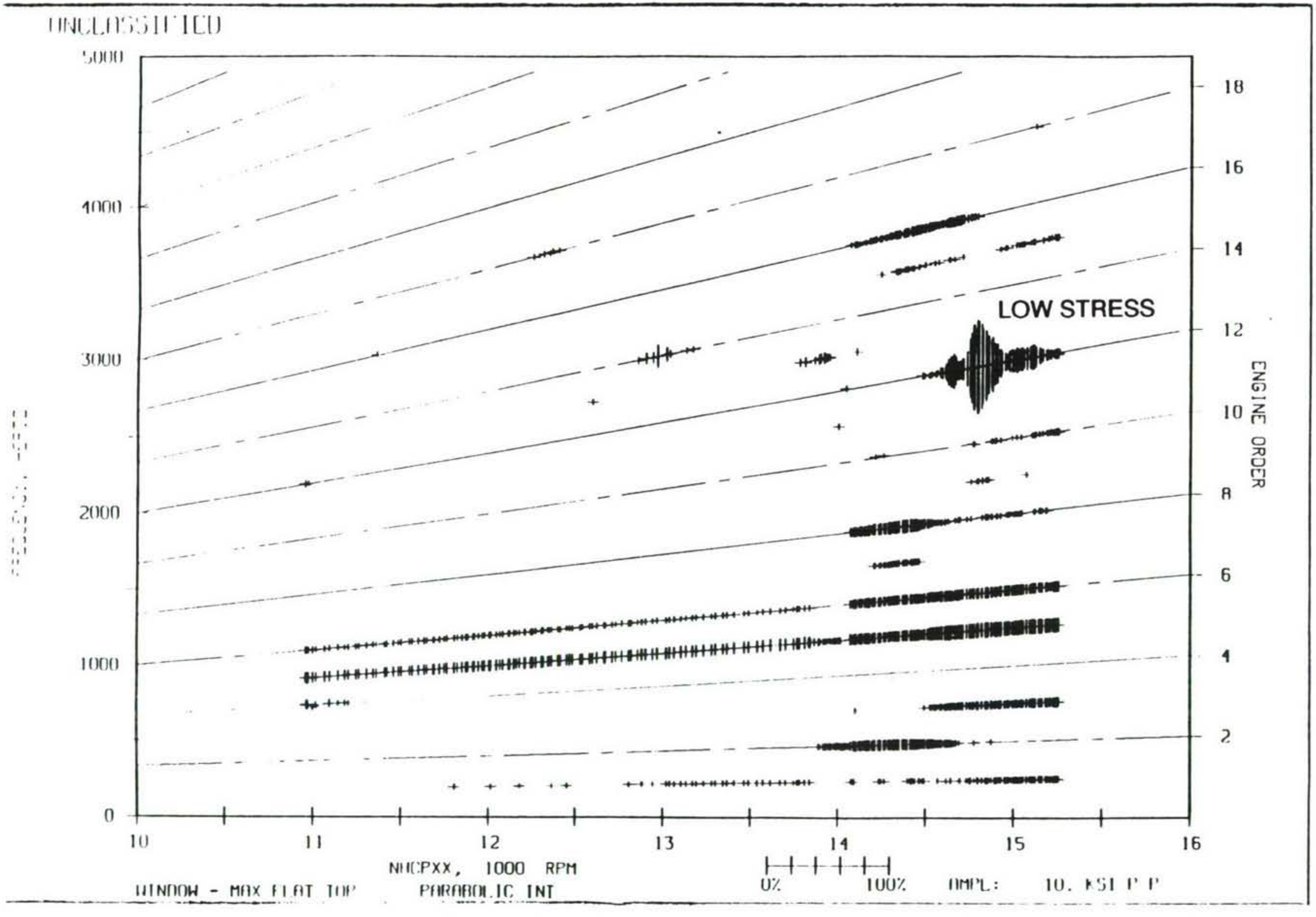
JET ENGINE HIGH PRESSURE COMPRESSOR BLADE AND DISK MODEL

- BLADE DOVETAIL EXPERIENCING SPORADIC CRACKING
- DISKPOST CRACKS AT MATING LOCATIONS ALSO EXPERIENCED





- TYPICAL CAMPBELL DIAGRAM FROM AEROMECHANICAL TESTS
  - BLADE DOVETAIL REGION STRAIN GAGE
  - ALL MEASURED STRESSES VERY LOW



CURRENT MATERIAL TESTING REQUIREMENTS

- FERROUS ALLOYS
  - RUNOUT AT TEN MILLION (10^7) CYCLES

- NONFERROUS ALLOYS
  - RUNOUT AT THIRTY MILLION (3×10^7) CYCLES

- TYPICAL S/N CURVE FOR FERROUS MATERIALS
- SLOPE OF CURVE ASYMPOTOMATIC AT TEN MILLION (10^7) FATIGUE CYCLES

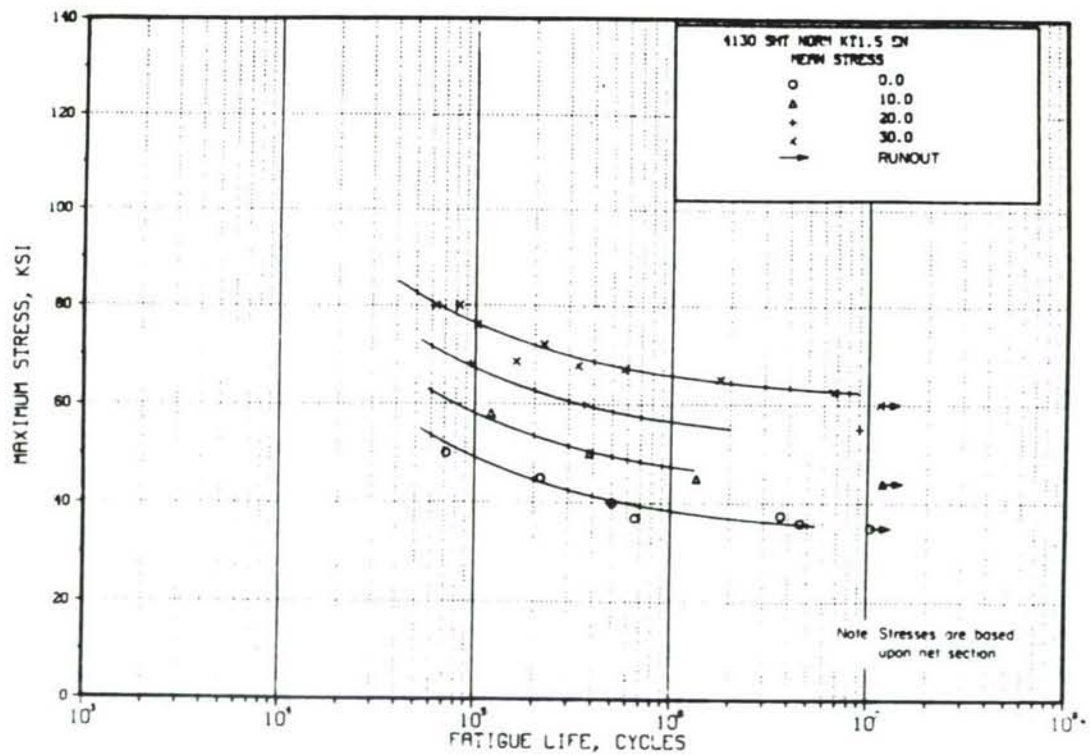


FIGURE 2.3.1.1.8(n). *Typical S/N curve diagram for notched (K_t = 1.5) fatigue behavior of 4130 alloy steel sheet, normalized (longitudinal).*

- TYPICAL S/N CURVE OF TITANIUM ALLOY
- CURVE HAS NEGATIVE SLOPE AT TEN MILLION (10^7) FATIGUE CYCLES

MIL-HDBK-5E
1 June 1987

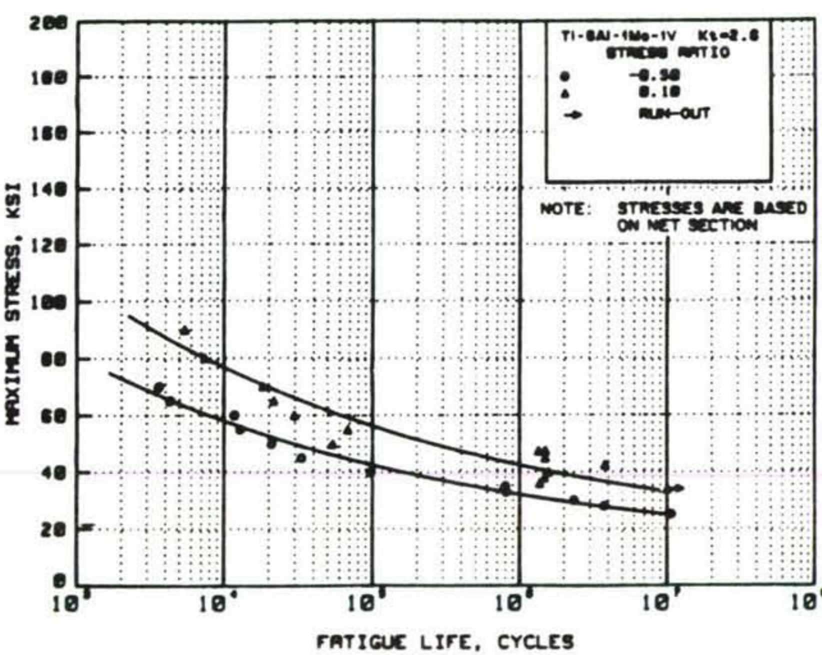
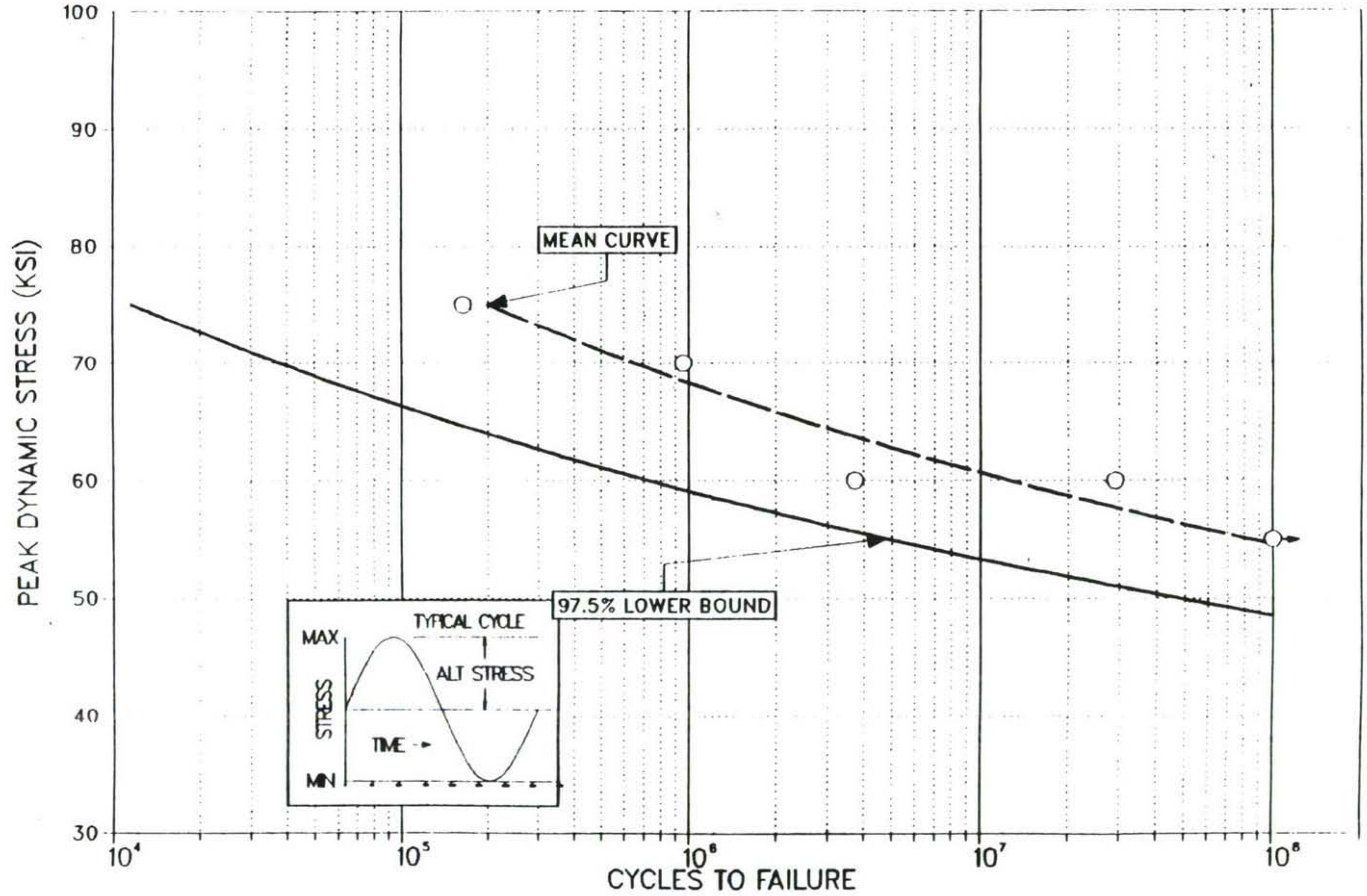


FIGURE 5.3.2.2.8(b). *Best fit S/N curves for notched, K_t = 2.6, duplex annealed Ti-8Al-1Mo-1V sheet at room temperature, long transverse direction.*

TI MATERIAL



PEAK DYNAMIC STRESS (KSI)
100
90
80
70
60
50
40
30
10^4
10^5
10^6
10^7
10^8
CYCLES TO FAILURE
MEAN CURVE
97.5% LOWER BOUND
TYPICAL CYCLE
MAX
ALT STRESS
STRESS
TIME
MIN

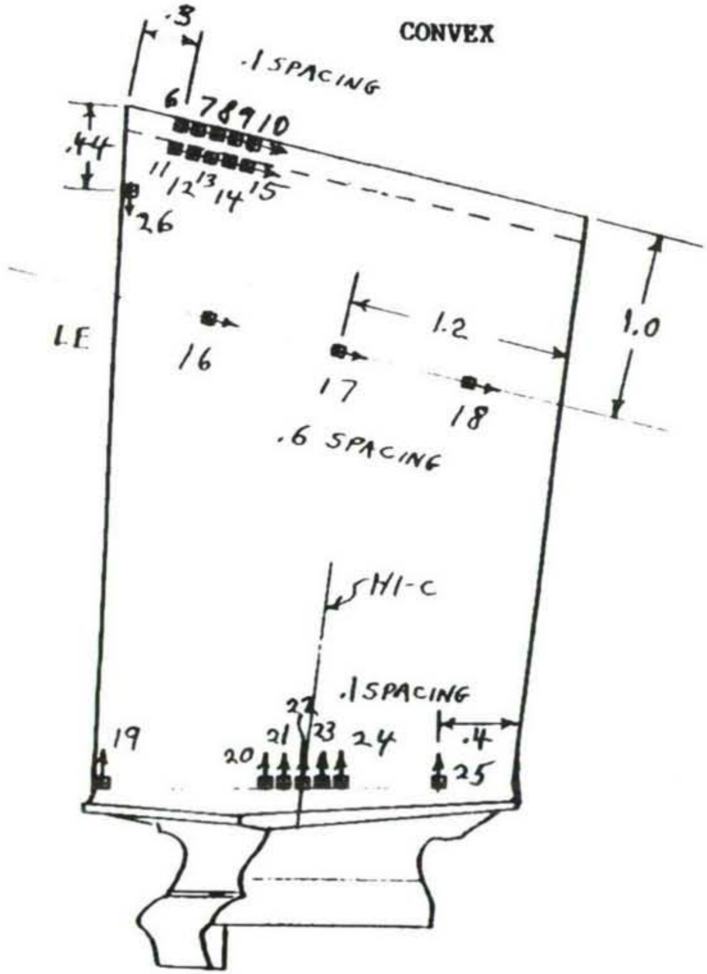
HIGH CYCLE FATIGUE RESULTS OF A TI 8-1-1 TEST BEYOND 10^7 CYCLES

- TEST CONDUCTED ON A TITANIUM ENGINE BLADE IN THREE STRIPE MODE

- RESULTS INDICATE A TEN PERCENT DECREASE IN FATIGUE STRENGTH BETWEEN 10^7 AND 10^{10} CYCLES

- FATIGUE TEST OF TITANIUM ENGINE BLADE
  - STRAIN SURVEY CONDUCTED ON AIRFOIL
  - FATIGUE TEST CONDUCTED BY STIMULATING BLADE AIRFOIL IN NATURAL 3-STRIPE MODE

CONVEX



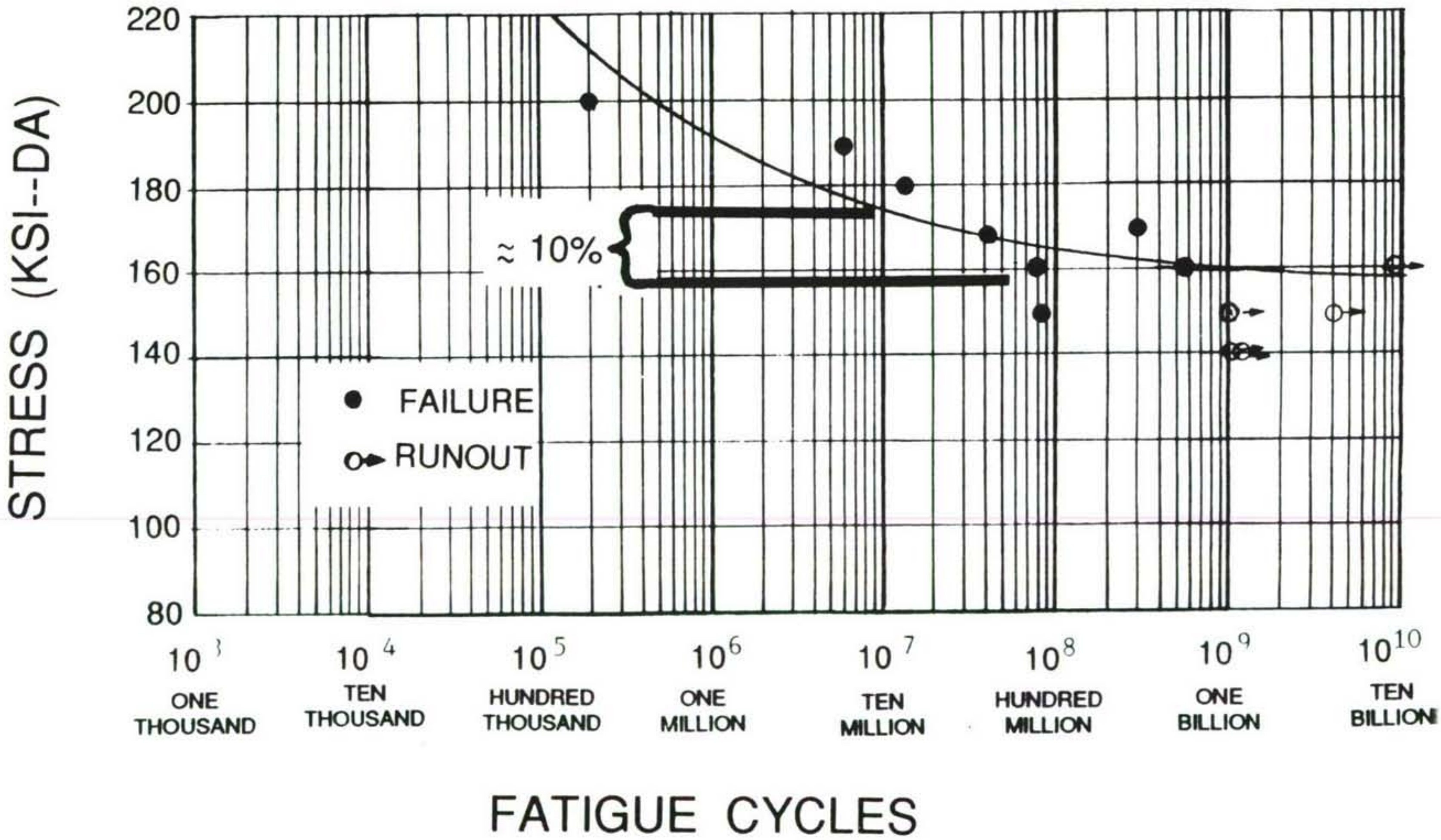
CONCAVE

6875 Hz

TITANIUM MATERIAL

- TI 8-1-1 Compressor Blade Vibrated In Natural Three Stripe Mode At ~7000 Hz

- Approximately 10% Decrease In Endurance Capability Beyond 10^7 Fatigue Cycles

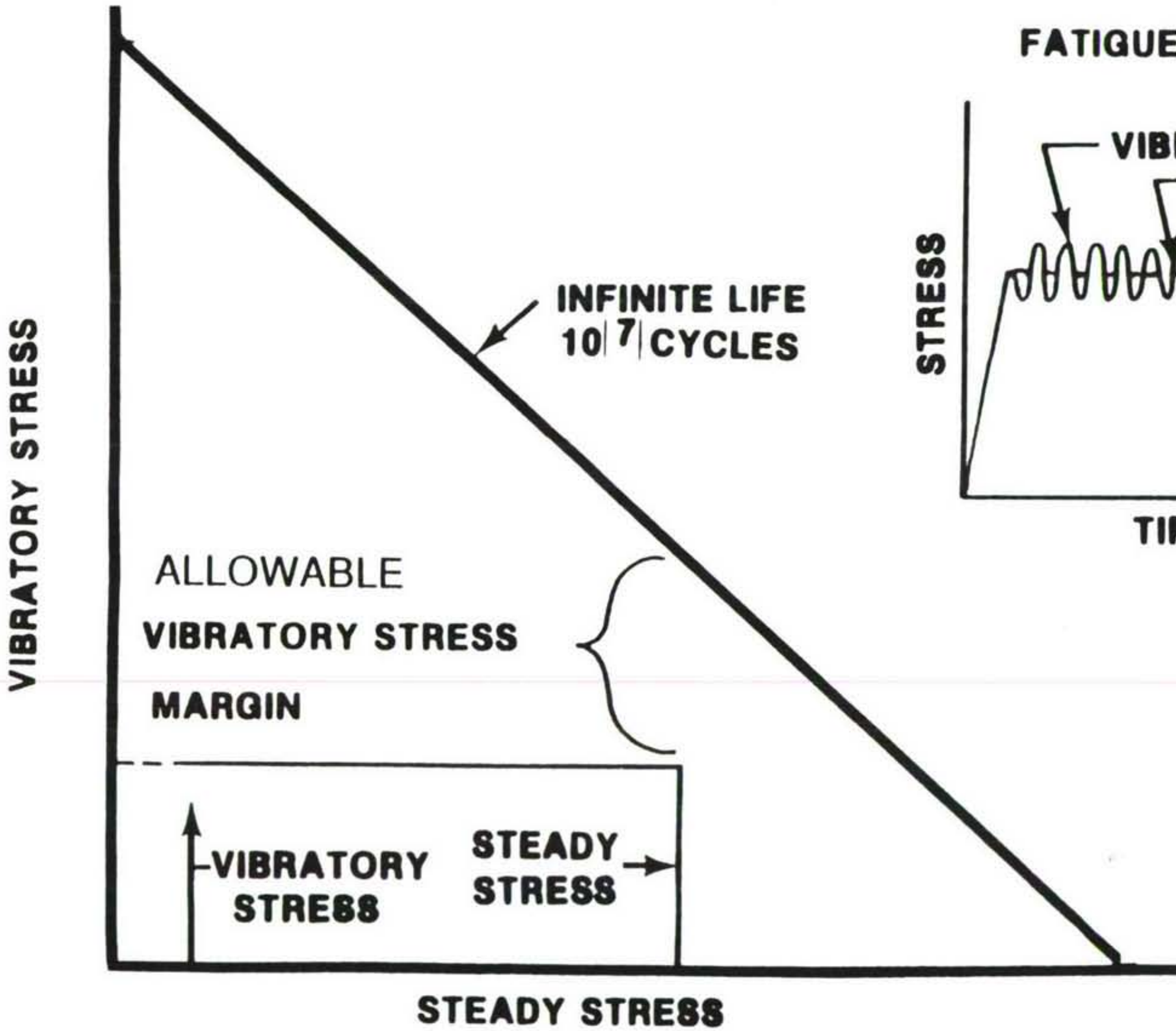


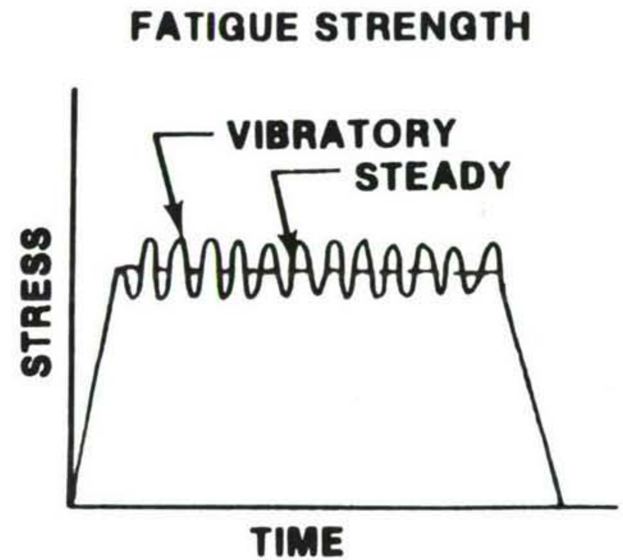
FUTURE ENDURANCE LIMIT TESTING

- MEAN LOAD WITH SUPERIMPOSED VIBRATORY LOAD

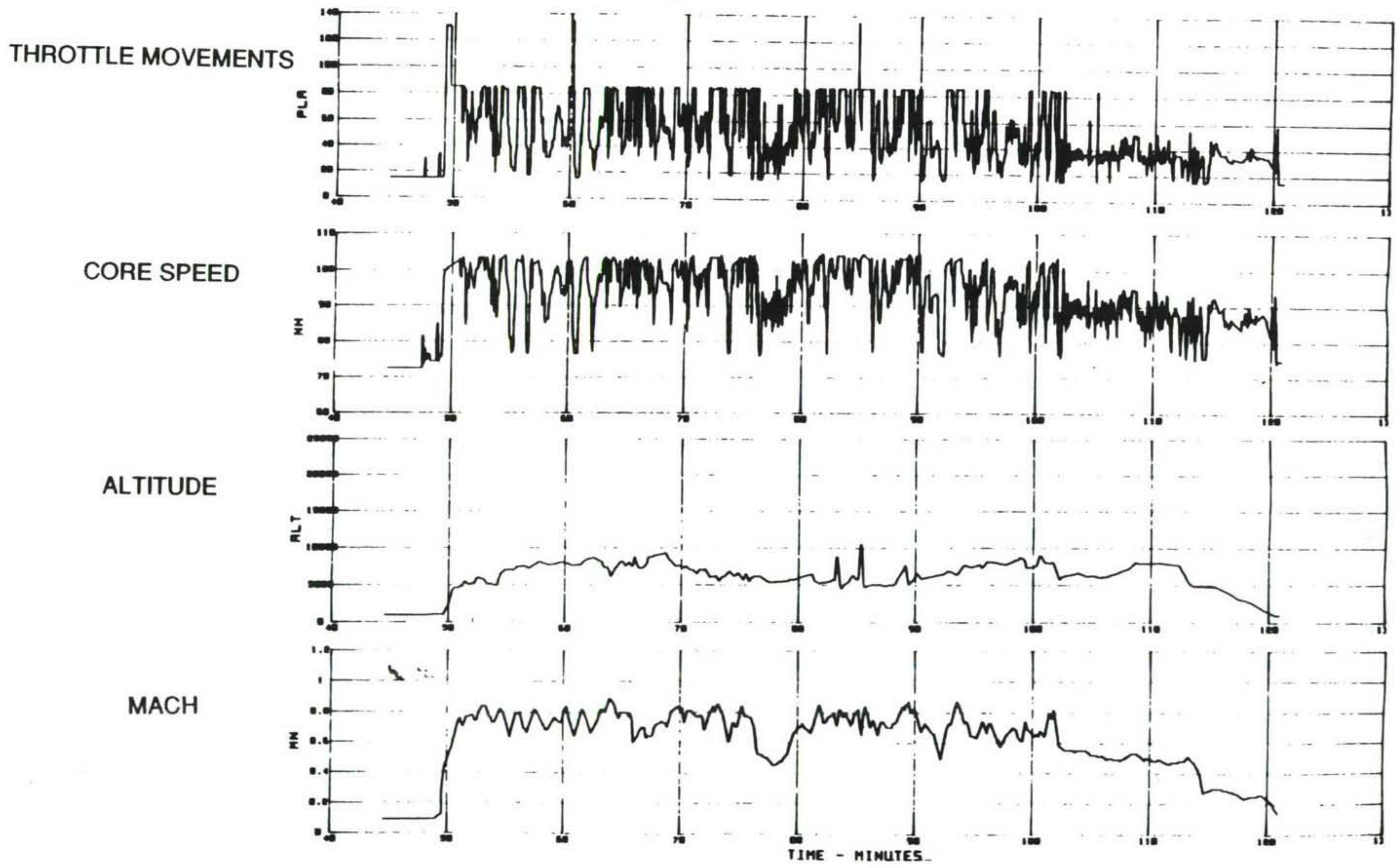
- ALTERNATING MEAN LOAD (LOW CYCLE FATIGUE) COMBINED WITH SUPERIMPOSED VIBRATORY LOAD (HIGH CYCLE FATIGUE)

DESIGN VARIABLES
GOODMAN DIAGRAM

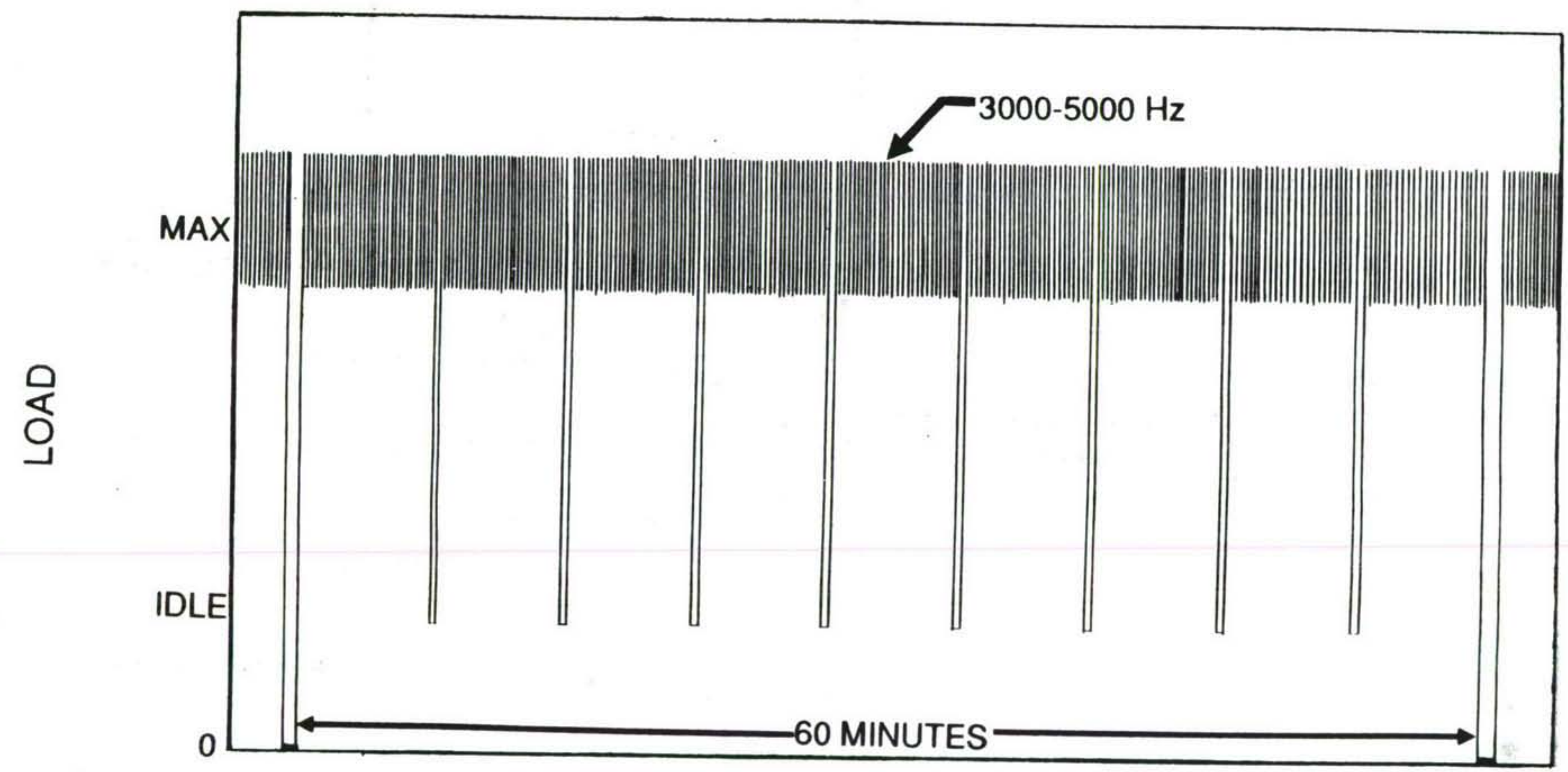




- F-16 MISSION PROFILE
  - MANY THROTTLE CYCLES
  - CORE SPEED EXCURSIONS WHICH ALTERNATE THE CENTRIFUGAL MEAN LOADING ON BLADES/DISKS
  - FLYING FAST ON DECK (HIGH PRESSURE AND TEMPERATURE)

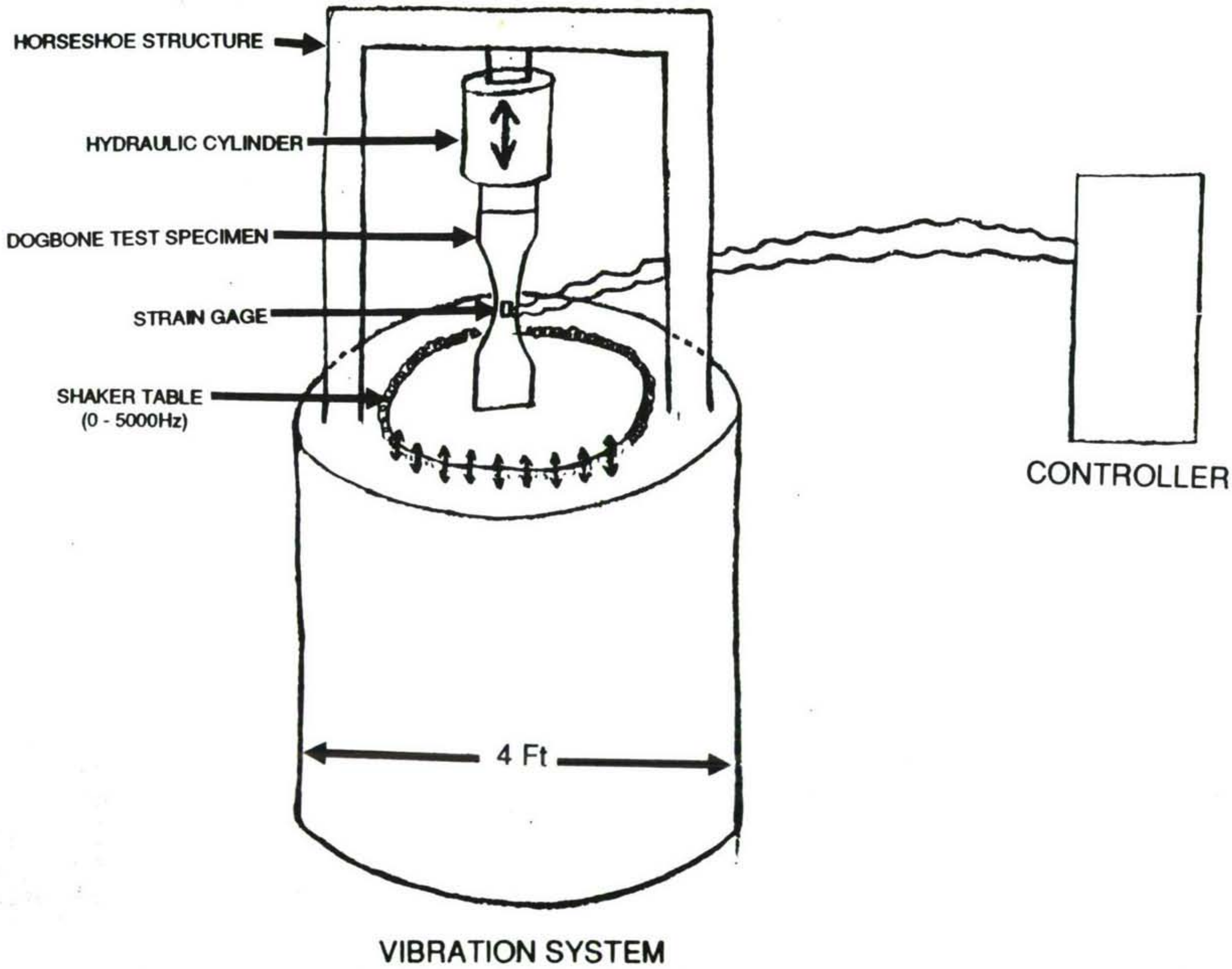


- HCF/LCF MATERIALS TEST LOAD SPECTRUM
  - ALTERNATING MEAN LOADS (LOW CYCLE FATIGUE)
  - SUPERIMPOSED HIGH CYCLE VIBRATION LOADING



HCF/LCF FATIGUE CYCLES

•ONE POSSIBLE TEST APPARATUS FOR COMBINATION LCF/HCF MATERIAL TESTING



POSSIBLE FUTURE ENSIP REQUIREMENTS

- RESULTS OF ENDURANCE TESTS MAY JUSTIFY CHANGING MIL-STD-1783
  - TASK II OF ENSIP SPECIFICATION
- ADVANCED TACTICAL FIGHTER SPO IS REQUIRING 10^9 CYCLE MATERIALS TESTING IN ENGINE SPECIFICATION
  - STRUCTURAL COMPONENTS VIBRATING AT HIGH FREQUENCIES
  - CAN GENERATE 10^9 CYCLES IN RELATIVELY SHORT TIME SPAN

ENSIP SPECIFICATION TASKS

TASK I	TASK II	TASK III	TASK IV	TASK V
DESIGN INFORMATION	DESIGN ANAL. COMPNT. & MAT. CHARAC.	COMPONENT & CORE ENG. TESTING	GROUND & FLIGHT ENG. TESTS	PROD. QUAL. CONTROL & ENG. LIFE MGT.
• ENSIP MASTER PLAN • DESIGN SERV. LIFE & USAGE REQUIREMENTS • DESIGN CRITERIA	• DESIGN DUTY CYCLE • MAT'LS AND PROCESSES DESIGN DATA CHARACTERIZED • STRUCTURAL/ THERMAL ANALYSIS • MFG. AND QUALITY CONTROL	• STRENGTH TESTING • DAMAGE TOLERANCE TESTS • DURABILITY TESTS • THERMAL SURVEY • VIBRATORY STRAIN & FLUTTER BOUNDARY SURVEY	• ENVIR. VERIF. TESTING • (AMT) TEST SPEC. DERIV. • DURABILITY TEST (AMT) • DAMAGE TOL. TESTS • FLIGHT TEST STRAIN SURVEY • UPDATED DURA. & DAM. TOL. CONTROL PLAN • PERFRM. DETERIOR. STRUC. IMPACT ASSESSMENT • CRITCL. PART UPDATE	• PROD. ENG. ANALYSIS • STRUC. SAFETY & DURAB. SUM. • ENG. STRUC. MAINT. PLAN • INDIV. ENG. TRACKING • LEAD THE FORCE PROG. (USAGE) • DURA. & DAM. TOL. CONTROL PLAN IMPL. • TECHNICAL ORDER UPDATE

ENSIP-9 De 107

SUMMARY

- TESTING DRIVEN BY UNEXPLAINABLE FATIGUE CRACKING PROBLEMS
- CURRENT TESTING HAS SHOWN POSSIBLE 10% REDUCTION IN ENDURANCE CAPABILITY BEYOND ACCEPTED STANDARD PRACTICE OF $3X10^7$ CYCLE RUNOUT
- FUTURE TESTS NEED TO ADDRESS LCF/HCF LOAD INTERACTION
- IMPLICATION OF TEST RESULTS MAY REQUIRE ENSIP SPECIFICATIONS (MIL-STD-1783) TO BE CHANGED REQUIRING 10^9 CYCLE MATERIAL CHARACTERIZATION

GA26604

"Aluminum-Lithium Technology Transition to Aerospace Structures - Lessons Learned"

R. J. Bucci, R. S. James, R. J. Rioja,
M. D. Goodyear & P. L. Mehr

Alcoa Laboratories

1989 USAF Structural Integrity Program Conference

1989 December 05-07
Hilton Palacio Del Rio Hotel
San Antonio, Texas

ALCOA

ALUMINUM-LITHIUM TECHNOLOGY TRANSITION TO AEROSPACE STRUCTURES - LESSONS LEARNED

by

R. J. Bucci, R. S. James, R. J. Rioja,
M. D. Goodyear, and P. L. Mehr

Alcoa Laboratories, Alcoa Center, PA 15069

at

1989 USAF Structural Integrity Program Conference
Hilton Palacio Del Rio Hotel, San Antonio, TX
December 5-7, 1989

Abstract

Much recent attention has been given to replacing present day aluminum alloys with low density, aluminum-lithium alloys in aerospace structures. More than a decade of R&D has gone into the development and characterization of current generation Al-Li products, and the list of candidate part applications is growing. It has become increasingly apparent, however, that Al-Li alloys, in many instances, cannot be substituted directly for current aluminum alloys without detailed analysis of their differing engineering characteristics. That is, implementation must consider the possibility of different material properties being critical to the design process. This thinking is applicable to any new material application which requires exploiting property trade offs to gain an advantage over the incumbent material.

The Al-Li implementation pace has been slower than first projected. This is the result of having to overcome a number of problems not envisioned in the alloy development stage. Addressing each new hurdle took time, however, the progress made and lessons learned have more favorably positioned Al-Li alloys for an efficient transfer to widespread use. This presentation highlights important lessons learned over the Al-Li development time frame, giving emphasis to areas where Al-Li experience differs from that of conventional aerospace alloys. The areas to be discussed are broadly categorized as follows:

- Lessons learned in formulating alloy development goals.
- Lessons learned in materials fabrication.
- Lessons learned in materials testing and extension to design.
- Lessons learned from customer evaluation trials.

GA26604.1

Table of Contents

- Al-Li technology is "alive and well." Al-Li products are being applied in a number of military and commercial programs.
  - Commercial product status
  - Review several applications
- Lessons have been learned during the implementation of Al-Li.
  - Review: hurdles
set backs
solutions → Tie in with USAF Technology Transition Criteria
- The future:
  - Promising - with added experience, more applications are expected.

Status of Al-Li Alloys

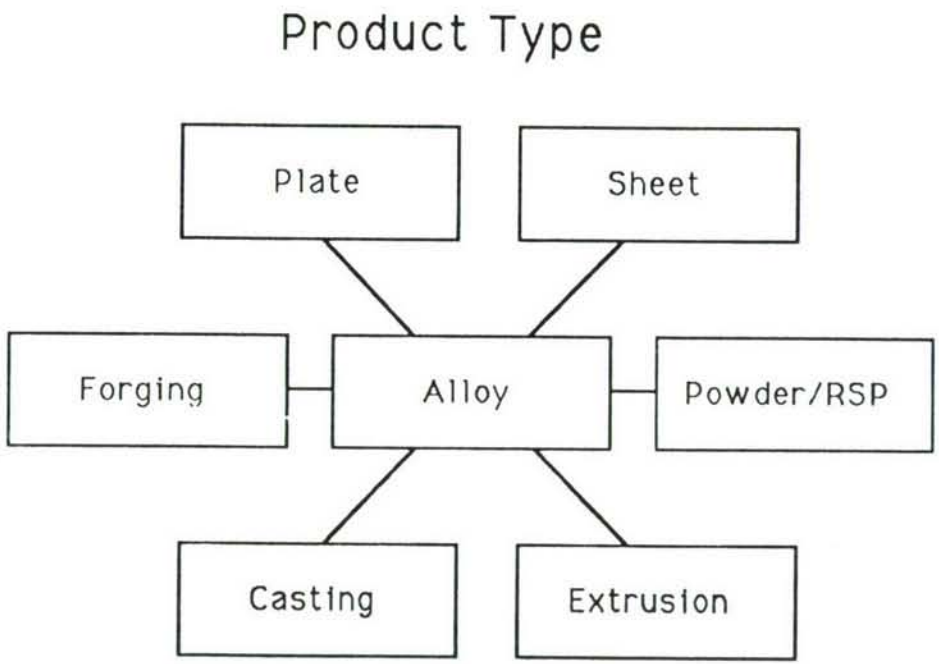
- They are not what we had hoped - a direct substitute for 7075, 2024 and other aluminum alloys.

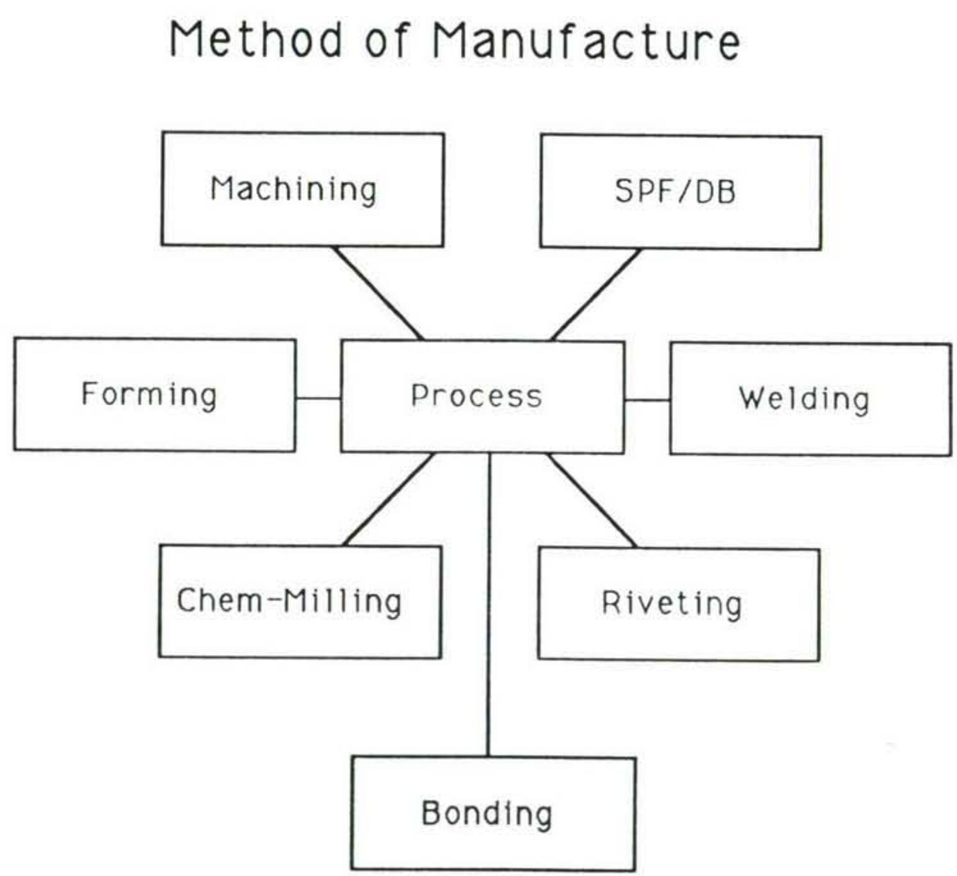
- They are:
  - lightweight (7-10% lower density)
  - commercially available
  - characterized with reproducible properties

- Recognizing attributes of the individual alloys has enabled Alcoa to work jointly with airframers in seeking out cost-effective applications and in the development of flying parts.

GA26604

Cost Considerations





GA26604.15

Al-Li Alloys - Alcoa Commercial Status

- Alloy 2090 - Committed to supply and service { sheet, plate, extrusion
- Alloy 2091 - Committed to supply and service - sheet.
- Alloy 8090 - Committed to qualifying and supplying plate as quickly as possible. Investigations underway for sheet, and a commercial position will be stated in 1990.

GA26604.12

Al-Li Alloy 2090
Alcoa Product Summary*

Product Form	Temper	Characteristics	Status
Sheet	0	Annealed, lowest strength, maximum formability	Commercial
Sheet	T3	Good formability. Will approach T8X properties after aging by user.	Commercial
Sheet	T31	Moderate formability. Can be aged to T8X properties by user.	Commercial
Sheet	T62	Solution heat treated and aged by user.	Commercial
Sheet	T83	Strength similar to 7075-T6.	Commercial
Sheet	T84	Strength/toughness similar to 7075-T76.	Commercial
Plate	T81	Strength similar to 7075-T651.	Commercial
Extrusion	-T86	Strength similar to 7075-T6511.	Commercial
Forging	-T6E203		Development

* Over 10 million lbs. ingot produced through 1989.

GA26604.13

Al-Li Alloy 2091
Alcoa Product Summary*

Product Form	Temper	Characteristics	Status
Sheet	0	Annealed, lowest strength.	Commercial
Sheet	T3	Heat treated and controlled stretch.	Commercial
Sheet	T4	Heat treated and flattened.	Commercial
Sheet	T84	T3 temper slightly aged, similar to 2024-T3, -T351 strength and toughness.	Commercial
Extrusion	- - - -		Not Available
Forging	- - - -		Not Available

* Over 4 million lbs. ingot produced through 1989.

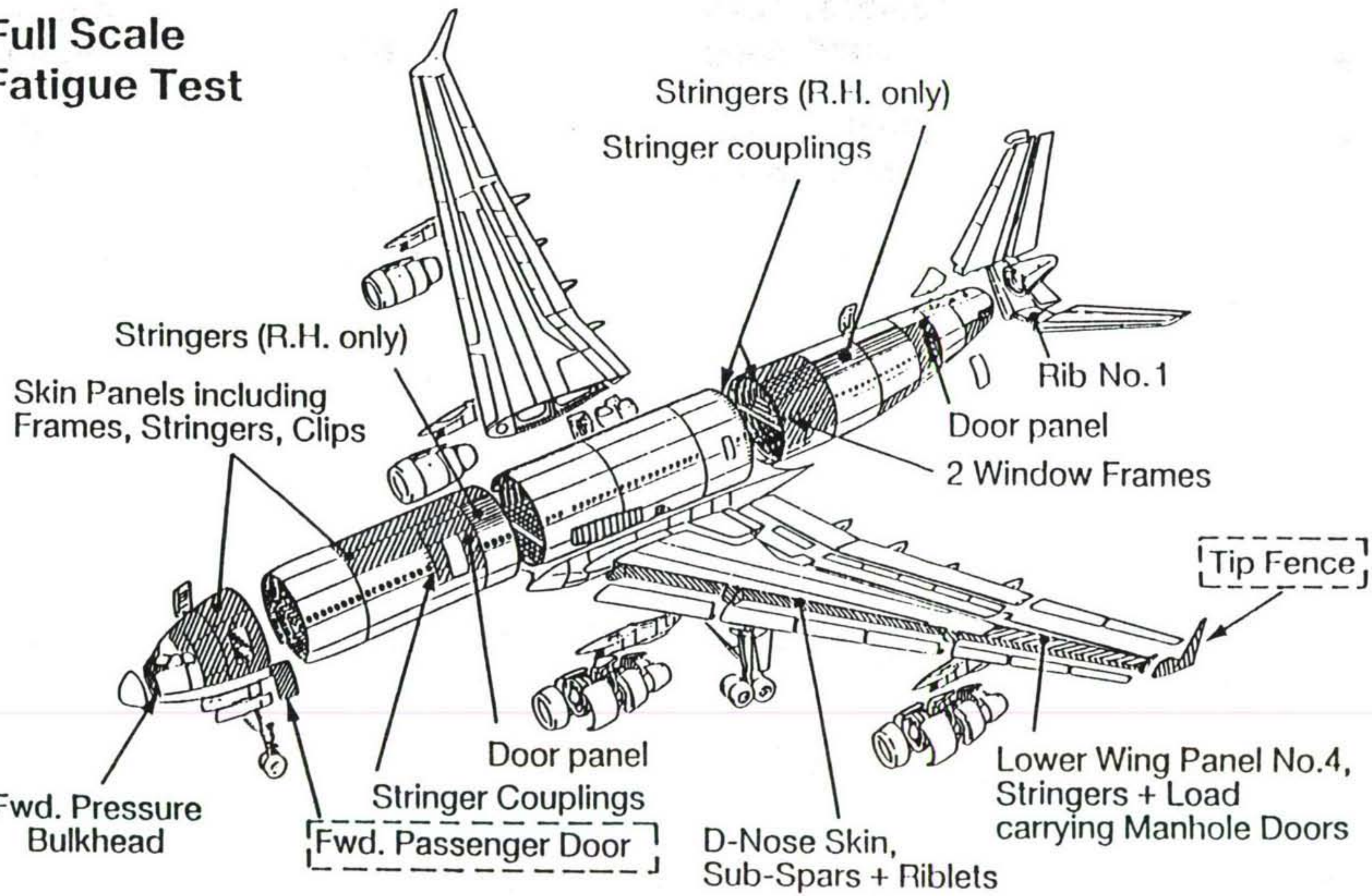
Al-Li Alloy 8090
Alcoa Product Summary*

Product Form	Temper	Characteristics	Status
Sheet	T8	Damage tolerant product similar to 2024-T3	Development
Plate	T7E102	Damage tolerant, 2024-T351 target.	Qualification
Plate	T8E57	Medium strength, 2124-T851 target.	Qualification
Extrusion	T8511	Damage tolerant product similar to 2024-T3511 strength and toughness	Development
Forging	T652	Medium strength and toughness, corrosion resistant.	Development
Billet	- -	Extrusion and forging starting stock.	Qualification

* Over 1 million lbs. ingot produced through 1989.

Application of AL-LI on A330 / A340

Full Scale Fatigue Test



Al-Li alloys are expected to be used extensively in Airbus A330/340 aircraft. About 1100 kg of weight can be saved in each airplane using alloys 2090, 8090, and 2091.

GA26604.3

Current Applications in Production

Application	Market	Alloy/Product	Program
Fixed leading edges	Commercial and Military	2090 sheet 8090 extr.	A330/340
Trailing edges	Commercial and Military	2090 sheet 8090 extr.	A330/340
Flooring	Commercial and Military	2090 sheet, extr.	
Exterior surfaces	Space, Military	2090 & 8090 sheet, plate	F/A18 X31A
Payload adapters	Space	8090 plate, extr. forgings	
Primary structure	Space	8090 plate, extr., forgings	

GA26604.16

Applications (Not in Production) Where Al-Li Alloys Proposed as the New Baseline

Application	Market	Alloy/Product	Program
Movable leading edge (slats)	Commercial Military	2090 sheet 8090 extr.	A330/340
Upper and lower wing skins	Commercial Military	8090 plate	A330/340 F/A18, F15
Fairings	Commercial	2090 sheet	A330/340
Stringers	Commercial	8090 extr. 2091 sheet	A330/340
Wing Spars	Commercial	8090 extr.	A330/340
Fuselage skin	Commercial Military	2091 sheet 8090 sheet	A330/340 F18 & others
Access doors/hatches	Commercial Military	2090 sheet 2091 sheet	F50 & others F/A18, F15 & others
Primary structure ribcage	Helicopter	8090 forg., sheet	EH101

GA26604.17

Other Applications Under Study

Application	Market	Alloy/Product
Bonded Upper Wings	Commercial	2090 sheet
Slats	Commercial	2090 sheet
Primary Structures	Commercial	8090 plate 2090 sheet
Nacelles	Commercial	2090 sheet
Fan casings	Commercial	8090 plate
Nose barrels, hatches/covers, ext. fuel tanks	Military	2090 sheet
Storage racks	Commercial	2090 sheet
Racking	Commercial Military	2090 extr. 8090 extr.

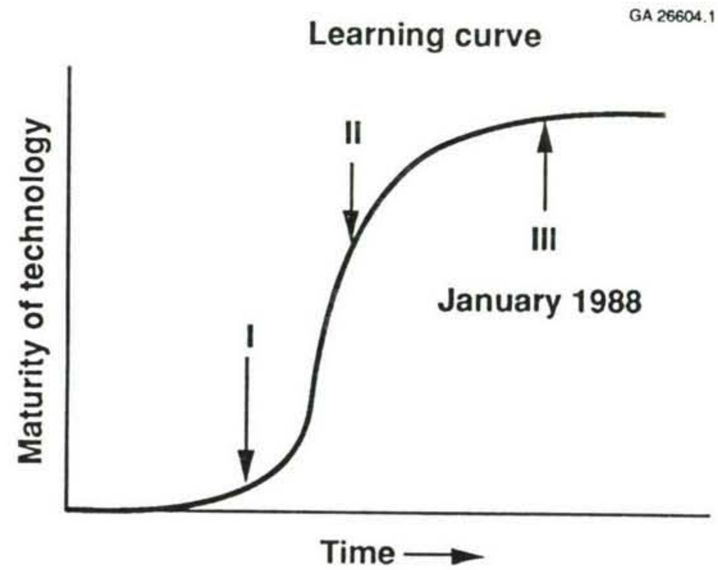
GA26604.4

Technology Transition Criteria
(Lincoln-USAF, 1987)

- Producibility
  - can properties be repeated in scaled up product?
- Stabilized Material and/or Process
  - are properties stable under manufacturing and operating envelopes?
  - are these envelopes known?
- Mechanical Property Characterization
  - is the material evaluation thorough - strengths & weaknesses?
  - are threats to structural integrity assessed?
- Predictability
  - can coupon data predict structural behavior?
  - are current test/analysis methods adequate?
- Supportability
  - can manufacturing or service damage be detected and repaired?

Alcoa Development of Alloy 2090 Plate and Sheet Products

New product development requires the climbing of a learning curve. At Alcoa the development of commercial quality 2090 products required the casting of over 10 million lbs. of ingot to attain the present property levels. A number of critical technologies have been optimized to improve both performance and property consistency. The figure on the right and tables below illustrate this.



Critical technologies improved prior to January 1988

Metal quality

- Improved composition control
- Eliminated harmful nonmetallic inclusions
- Reduced trace elements and hydrogen concentrations

Fabrication processes

- Established rolling practices
- Optimized grain microstructures
- Improved aging practices

GA26604.18

Alcoa Al-Li Alloy Vintages

Vintage I	1983 to 1986 November
Vintage II	to 1987 October

Designed Experiments

Vintage III	1988 January to date

GA26604.5

Development and Implementation of Al-Li Alloys (2090)

- Lessons learned
  - materials fabrication
  - materials testing
  - materials characterization and implications to design
  - determining correct property targets

Punch Line: Al-Li alloys in general have widespread potential uses, however, the material similarities and differences from standard Al alloys need to be quantified to ensure successful applications.

NSWC TR 89-106

Cooperative Test Program for the Evaluation of Engineering Properties of Al-Li Alloy 2090-T8X Sheet, Plate, and Extrusion Products

R. J. Bucci, R. C. Malcolm, E. L. Colvin,
S. J. Murtha, and R. S. James

Aluminum Company of America
Alcoa Laboratories
Alcoa Center, PA 15069

Final Report
(Period: 1984 September 14 to 1988 November 30)
Contract N60921-84-C-0078
Department of the Navy
Naval Surface Warfare Center
10901 New Hampshire Avenue
Silver Spring, MD 20903-5000

Distribution limited to U.S. Government agencies and their contractors only; Test and Evaluation. Statement applied September 1989. Other requests for this document must be referred to NSWC, Code R32.

1989 September 15

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12 PERSONAL AUTHOR(S): R. J. Bucci, R. C. Malcolm, E. L. Colvin, S. J. Murtha, R. S. James
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18 SUBJECT TERMS (Continue on reverse if necessary and identify by block number): Aluminum-Lithium, Alloy 2090, Sheet Plate, Extrusion, Test and Evaluation, Engineering Properties, Aerospace

19 ABSTRACT (Continue on reverse if necessary and identify by block number)

In response to demand for lighter and more cost-efficient aircraft, designers are looking to replace aluminum with aluminum-lithium in next generation airframe structures. One of the first new generation Al-Li alloys, Alcoa developed 2090, offers 8% lower density and 10% higher stiffness than 7075, a major high strength alloy used in aircraft. In a near-peak age (T8X) condition, alloy 2090 attains the highest strength of present commercial Al-Li alloys, matching the capability of 7075-T6.

New material acceptance requires critical assessments of performance and fit to manufacturing processes. This material distribution program was conceived with the goal of accelerating data collection on a promising Al-Li alloy, 2090, as input to preliminary design/manufacturing studies. Alcoa supplied samples of commercial sheet, plate, and extrusion products to 30 firms agreeing to share data under the auspices of a Navy cooperative test program. This report summarizes results of extensive testing considered for the material introduction. Comparison results on alloy 7075 are also reported.

Information established under this program revealed that Al-Li substitution for conventional aluminum alloys requires appreciation of their differing engineering characteristics. This thinking would apply to any new material application requiring the exploiting of property trade offs to gain an advantage over the incumbent material. The significance of test results and other Al-Li development issues, many not envisioned at the program outset, are discussed. Though the present work centered on alloy 2090, the many lessons learned over the program duration are applicable to Al-Li alloys in general, thus serving to position Al-Li alloys more favorably for widespread use.

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SECURITY CLASSIFICATION OF THIS PAGE
UNCLASSIFIED

The above report documents results of extensive alloy 2090 testing performed under a cooperative agreement involving 30 aerospace firms. The report final chapter discusses many important lessons learned during the Al-Li alloy 2090 development and evaluation stages, giving emphasis to areas where Al-Li and conventional aluminum alloy behaviors differ and the resulting consequences. Many of the experiences reported can be generalized to other Al-Li alloys.

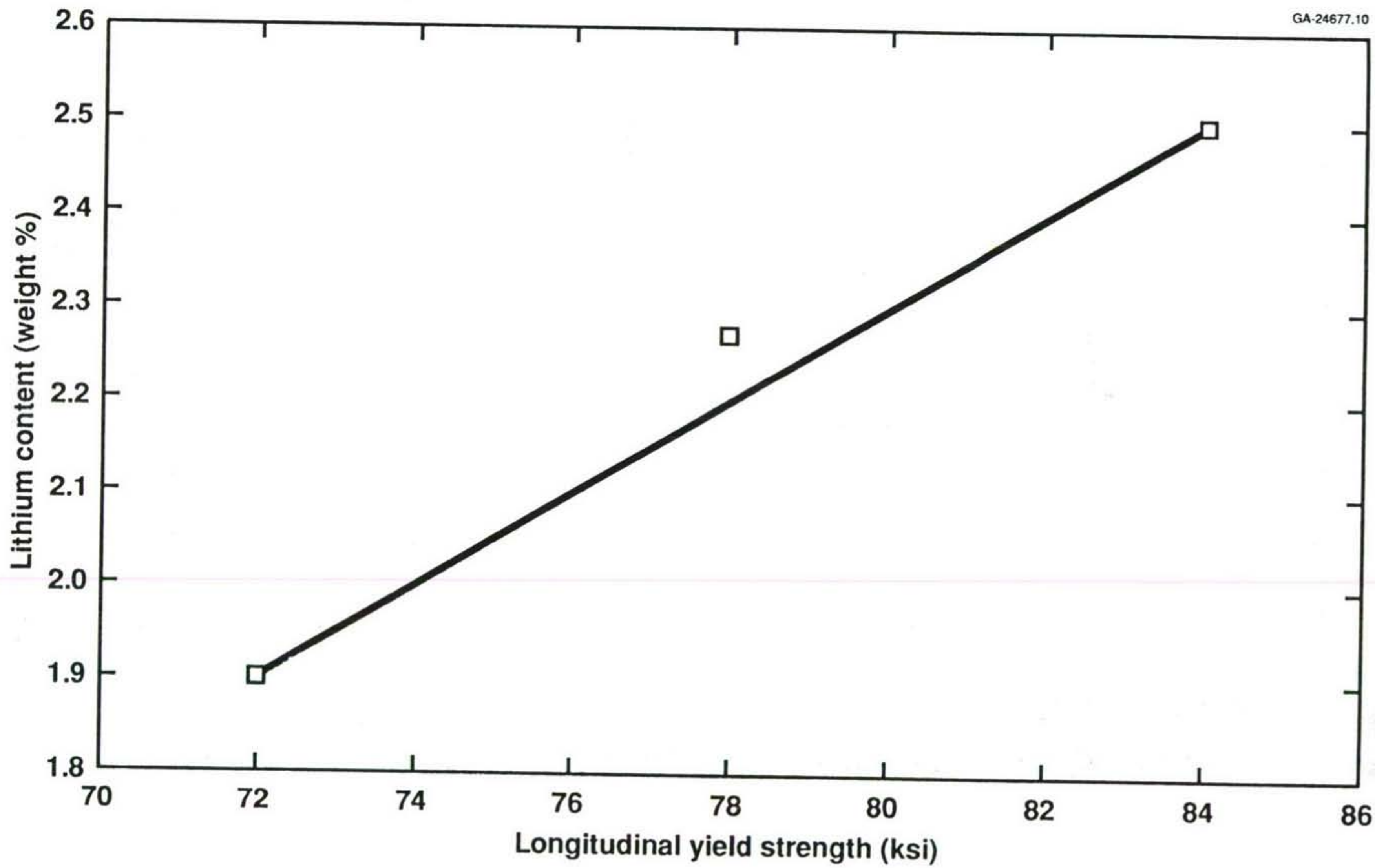
GA26604.6

Lessons Learned in Materials Fabrication and Supply

Lesson #1	A product should not be widely distributed for test/evaluation until commercial viability is reasonably understood.
Lesson #2	Broad product introduction should only be made when <u>all</u> production processes are under control.
Lesson #3	Track and obtain feedback on T&E material to refine processes for optimum property combinations.

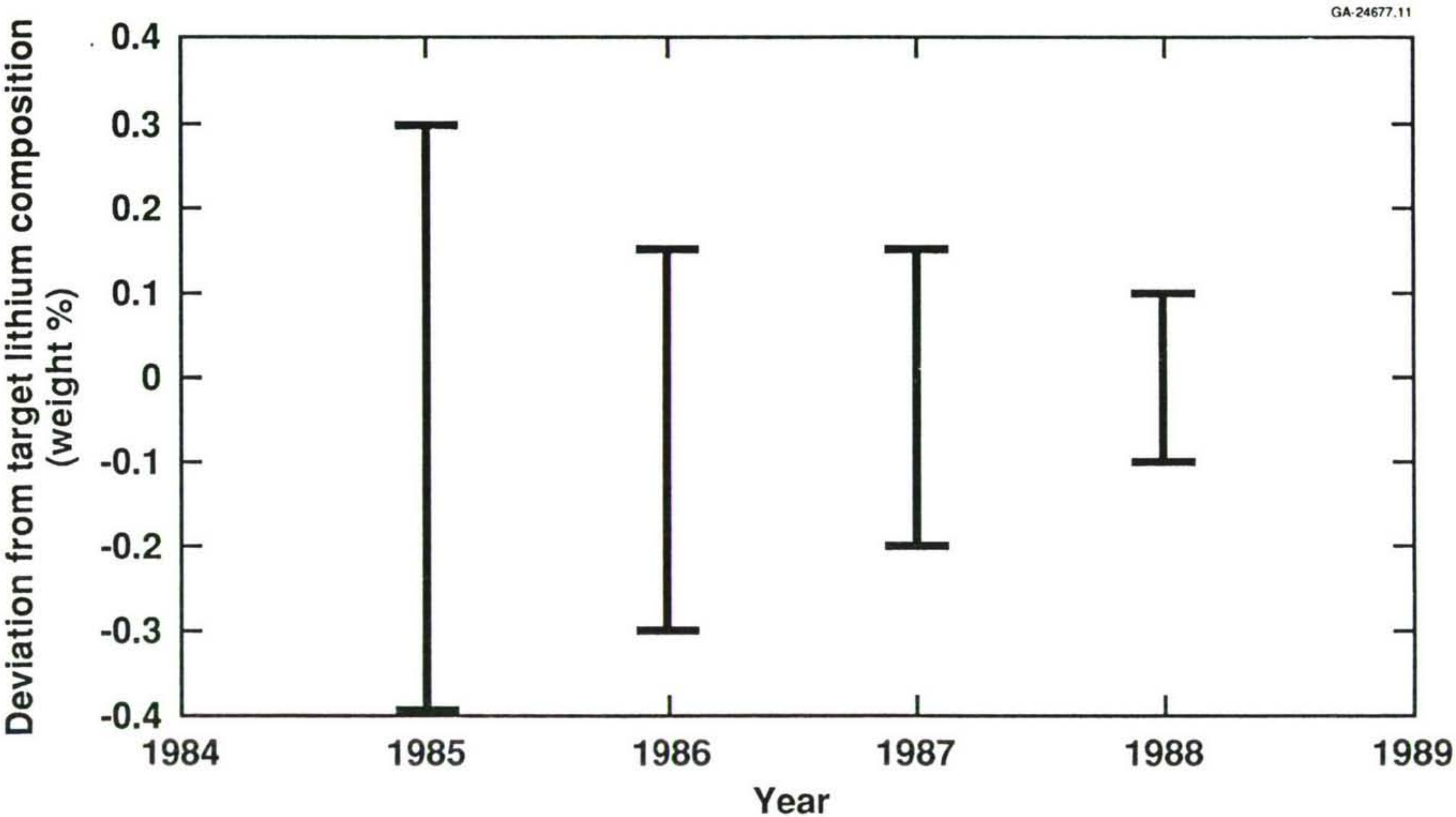
The Effect of Li Content on Alloy 2090 Tensile Yield Strength

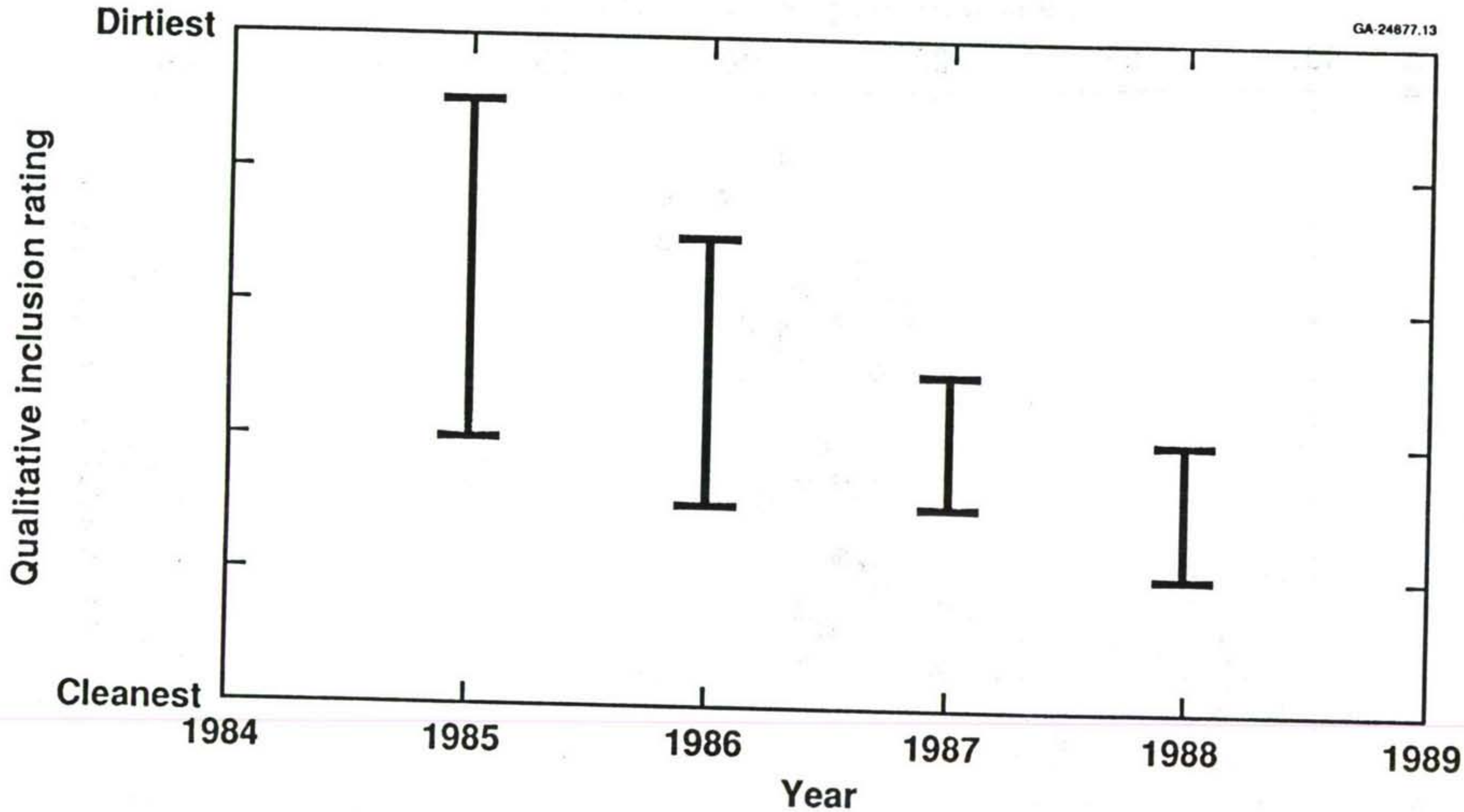
(Composition and Processing Held Constant)



Alcoa Historical Lithium Variability, Plotted by Year.

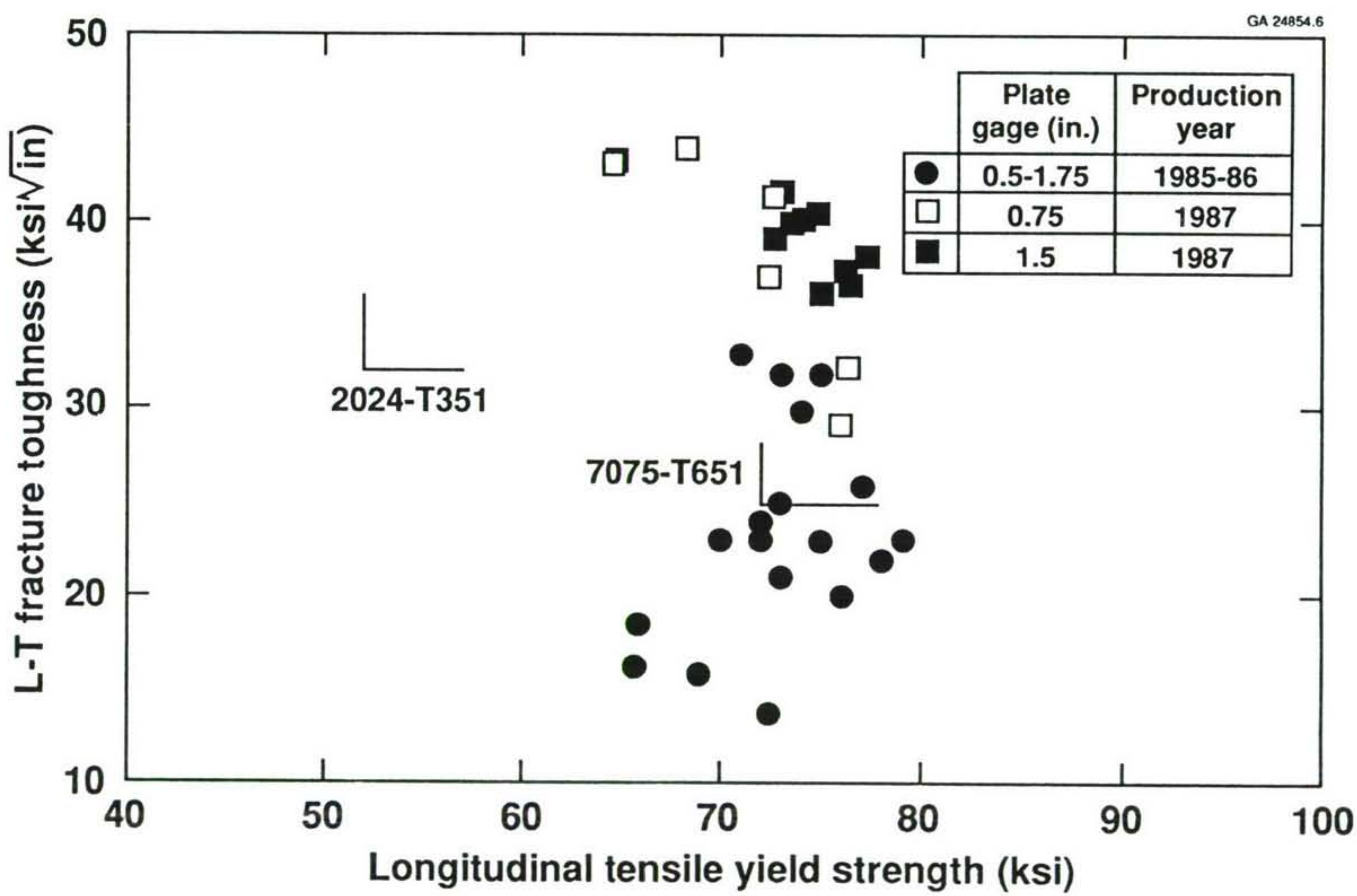
Consistent Composition Has Allowed Improved Property Guarantees.



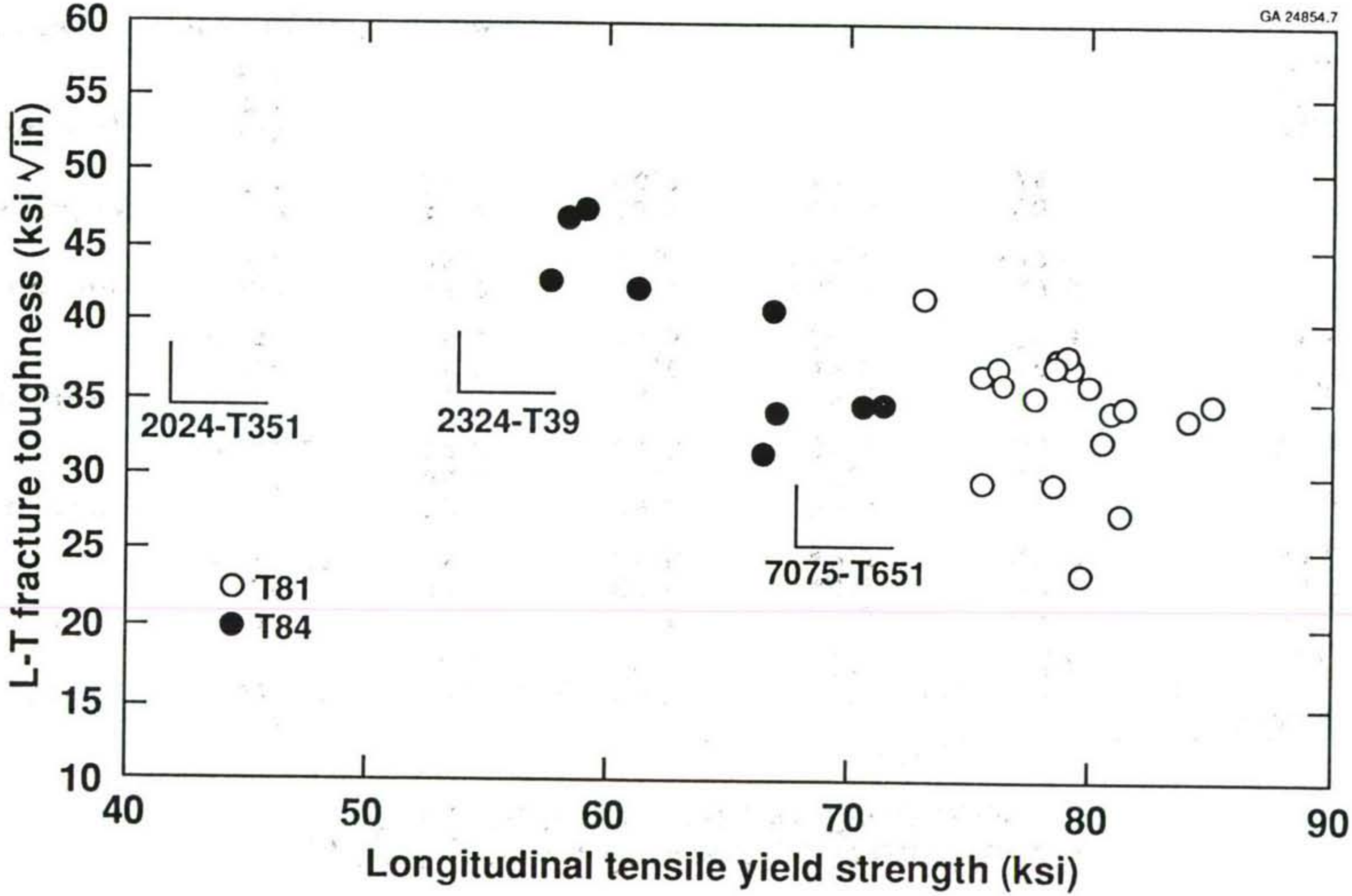


Inclusion Content Measured From Molten Metal Samples

2090 Plate, Chronological Strength/Toughness Property Improvements



Alloy 2090, (0.5 to 1.5 in) Plate Strength/Toughness, L, L-T Present Capability; 1988



GA26604.20

Typical (Mean) Properties from Recent Al-Li Plate and Sheet Products

Property	Test Direction	2090 thin sheet* T83	2090 thin sheet* T84	2091 thin sheet T8	2091 thin sheet T3	2090 plate (0.5 - 1.5 in. thick) T81
TYS (ksi)	L	76.6	71.2	52.8	47.5	77.3
	LT	74.2	66.5	48.1	43.0	75.8
	45° or ST	65.4	55.9	43.0	39.7	63.0
UTS (ksi)	L	82.0	78.8	63.4	58.5	82.4
	LT	80.1	76.4	67.2	61.9	81.8
	45° or ST	72.9	67.5	62.8	58.3	71.0
% elongation	L	5.1	4.3	15.8	18.7	6.6
	LT	7.1	7.6	13.6	16.2	5.4
	45° or ST	9.8	11.5	20.3	20.4	1.5
K_C (ksi$\sqrt{in}$)	L-T	45.2	72.4	138.7	132.6	
	T-L	40.3	68.8	119.1	129.6	
Kapp (ksi$\sqrt{in}$)	L-T	40.7	58.4	92.0	90.7	
	T-L	37.2	56.3	78.5	78.7	
K_{Ic} (ksi$\sqrt{in}$)	L-T					34.4
	T-L					29.9
	S-L					8.2

* thin sheet = 0.04 to 0.160 in. gage

These properties represent an average from recent lots (at least 5) fabricated according to standard practices. Design properties (at different stages of completion) are available upon request.

GA26604.7

Lessons Learned in Materials Fabrication and Supply

Lesson #4	**Make sure accelerated or coupon tests are good indicators of component performance in real life.**
Lesson #5	**Make sure producer tests correlate with user tests to be able to compare data.**
Lesson #6	**Resolve conflicting test data - otherwise it will come back to haunt you.**

GA26604.8

Comparison of Exfoliation Ratings[a] in Accelerated and Outdoor Exposure Tests

Alloy/ Temper	Plate Gage (in.)	Specimen Plane	EXCO[b] Rating 4 Day	MASTMAASIS[c] Wet	MASTMAASIS[c] Dry	ASSET	Pt. Judith, RI Sea Coast Environment Rating (Mos.)
2090 (As rec'd)	0.5	T/2	ED[d]	P			N(24)
2090-T81	0.5	T/10	ED[d]	P	P	P	N(24)
2024-T3	0.75	T/2	EC	EA	ED		ED(24)
7150-T6	0.75	T/2		ED	EC		EC(24)

(a) N – no appreciable attack
P – pitting
EA – superficial
EB – moderate
EC – severe
ED – very severe

(b) Per ASTM G34

(c) Per ASTM G85, Annex 2A

(d) Specimens removed from test after 2 days

These results show that the accelerated EXCO and MASTMAASIS tests do not predict alloy 2090 exfoliation performance in a sea-coast environment. Note also that alloys 2024-T3 and 7150-T6 show exfoliation in both simulated and real environments.

GA26604.9

Conclusions: Al-Li Alloy Corrosion Behavior Exfoliation

Ref: J. J. Thompson (NADC), ASTM Workshop on Exfoliation Corrosion, Baltimore, 1988 May 17

- The exfoliation corrosion susceptibility of conventional Al Alloy 7075 in Naval environments can be distinguished utilizing accelerated laboratory testing.
- Al-Li alloys in commercial T8 tempers are resistant to exfoliation in Naval environments.
- Al-Li exfoliation ratings from accelerated tests did not correlate well with shipboard exposure results.
- SO_2 salt spray and mastmaasis accelerated laboratory environments do not produce exfoliation in Al-Li alloys. Exposure beyond two weeks produces deep pitting not found in shipboard exposure specimens.

GA26604.10

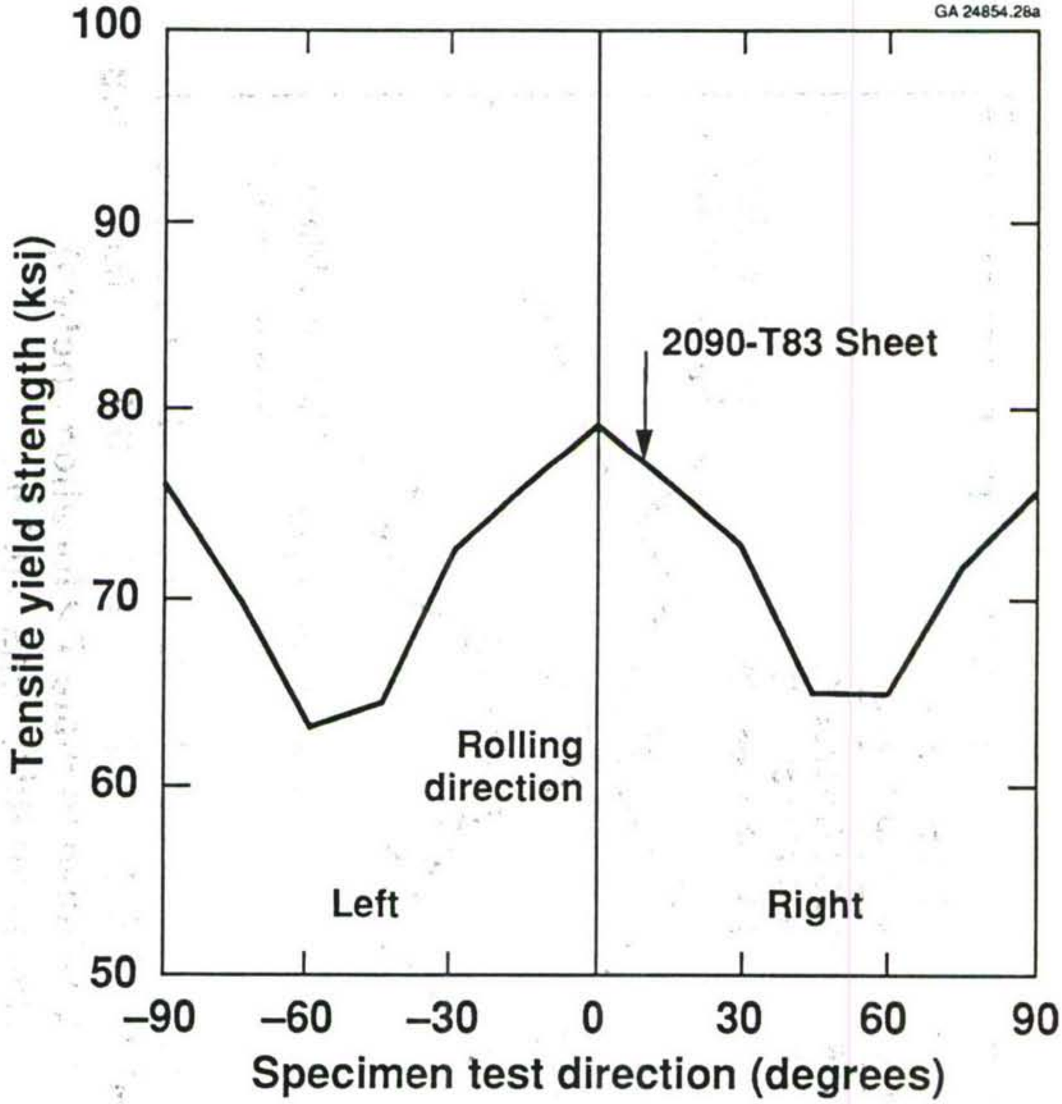
Lessons Learned in Materials Characteristics and Implications to Design

Lesson #7 Properties of a product may be irrelevant if components cannot be manufactured.

Lesson #8 Properties must be measured in several directions (anisotropy effects).

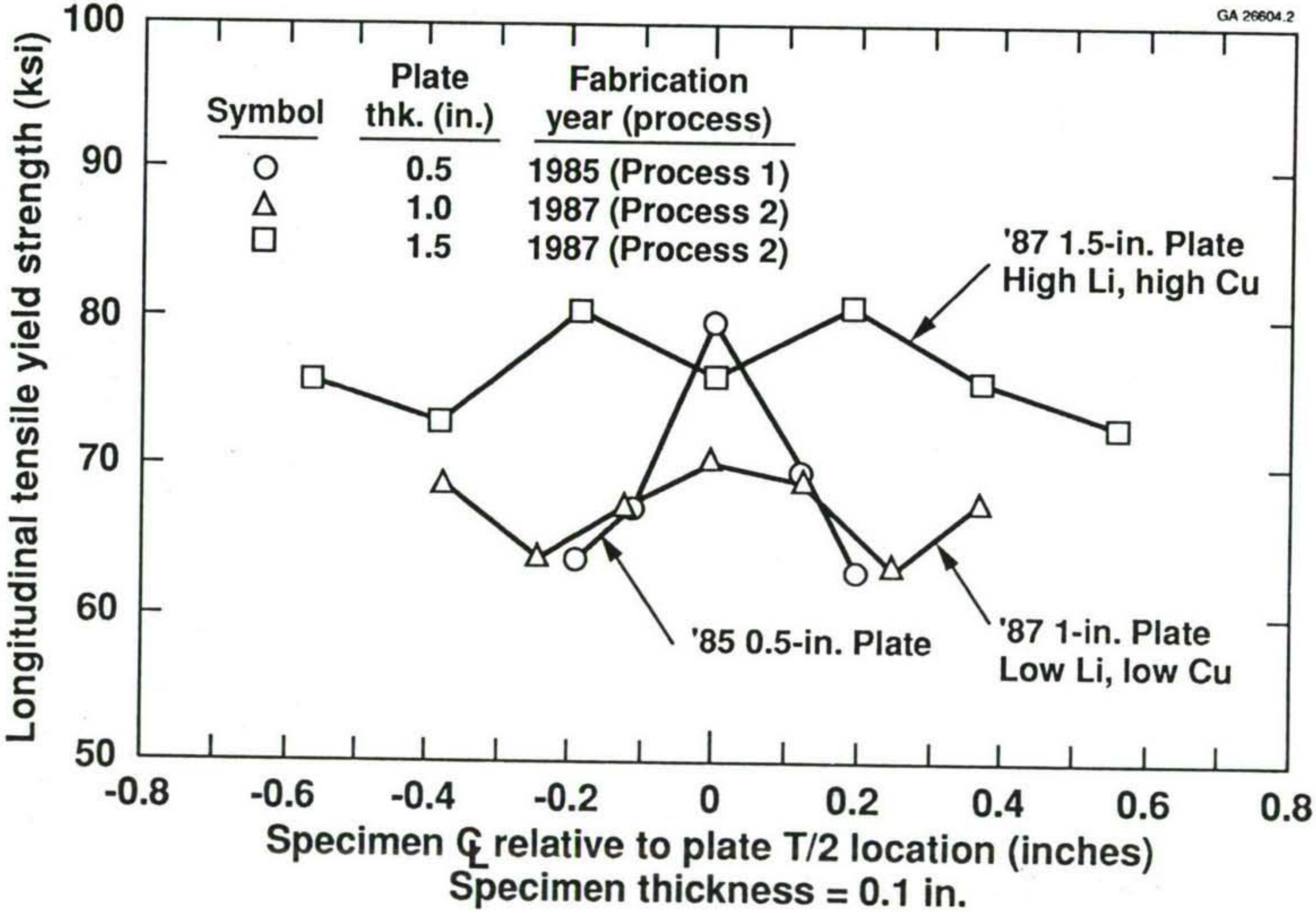
Lesson #9 Al-Li products (2090 in particular) need cold work to attain optimum combinations of strength and toughness, similar to 2024 and 2219.

Lesson #10 Al-Li - T8X products are aged below peak strength. Therefore, simulated temperature/time exposures may be required for performance validation.

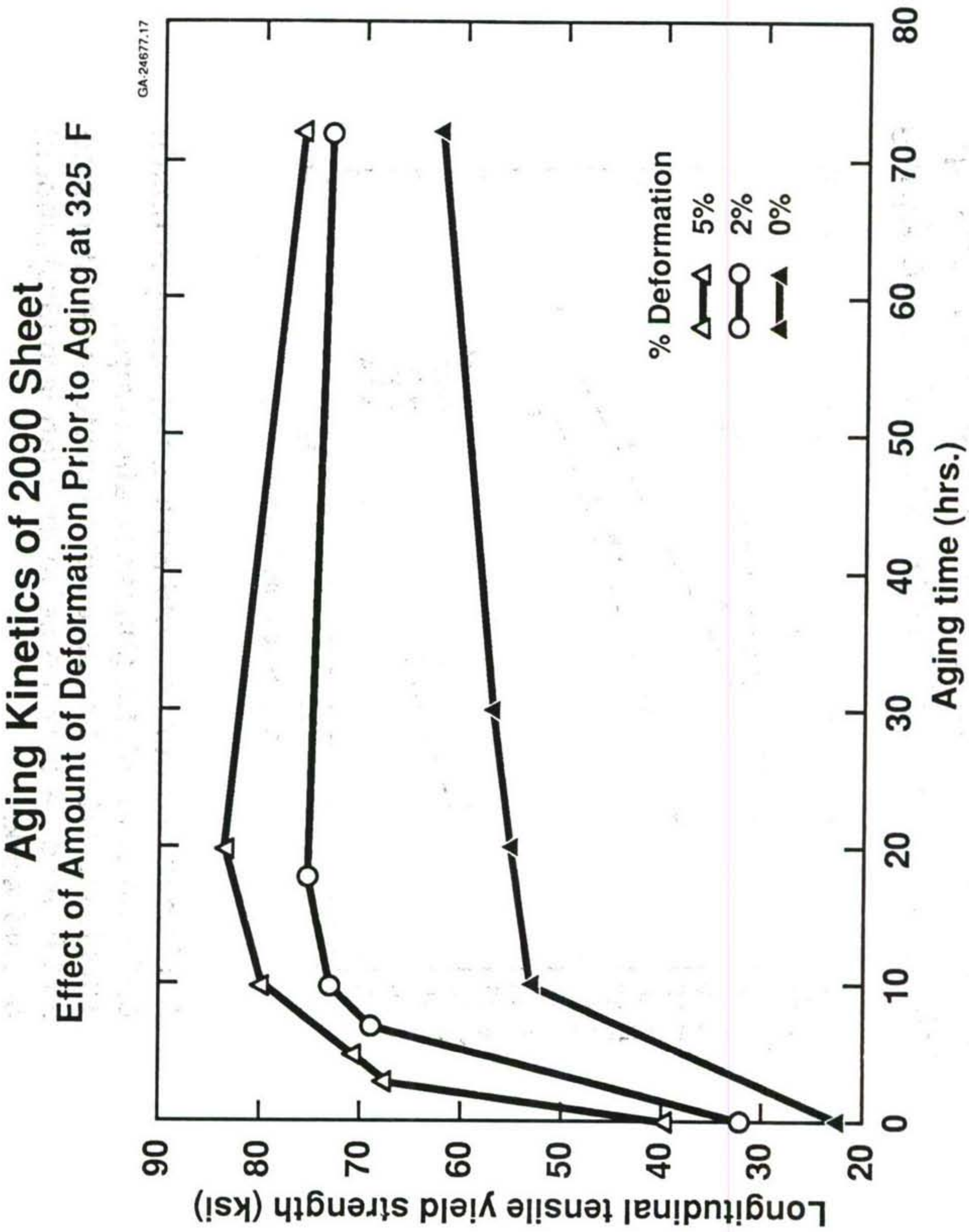


Yield strength on 2090-sheet varies more than standard Al alloys and is at a minimum at 45° from the rolling direction. Designers must determine the design strength to be used for 2090 on application by application basis, or use the lower 45° properties if the part doesn't require 7075 type strength. Al-Li alloy 45° property allowables will be incorporated into MILHDBK-5.

Effect of Processing Improvement on the Through-Thickness Tensile Strength Gradient of 2090-T81 Plate Product

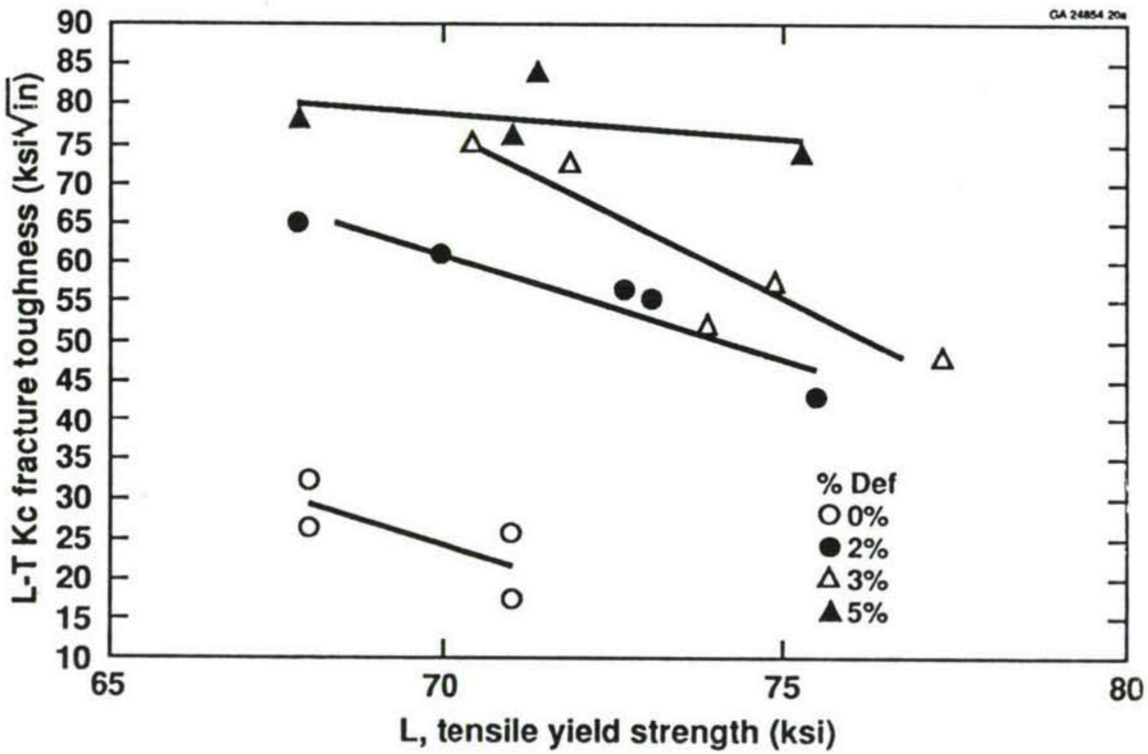


Note: '87 Process improved material yields a more uniform product.



Aging Kinetics of 2090 Sheet
Effect of Amount of Deformation Prior to Aging at 325 F
GA-24677.17
% Deformation
5%
2%
0%
Longitudinal tensile yield strength (ksi)
20
30
40
50
60
70
80
90
Aging time (hrs.)
0
10
20
30
40
50
60
70
80

2090 Sheet Strength/Toughness Relationship
Effect of Amount of Deformation Prior to Aging



A minimum amount of cold work is necessary to obtain optimum strength/toughness relationships.
This is important because not all customer heat treated and formed parts will receive the standard (3%) amount of cold work.

GA26604.11

Lessons Learned Determining Correct Property Targets

Lesson #11 Just because a property is not listed in a specification it does not mean that it is not important to design.

Lesson #12 Discuss engineering requirements with structures, producibility and systems support personnel.

GA26604.19

Conclusions

- The potential for 8-10% structural weight savings using Al-Li alloys is a significant application driver.
- Al-Li alloys are being used/specified in a number of commercial and military aircraft programs.
- Al-Li alloys in general, and 2090 in particular, are not a direct substitute for conventional aerospace Al-alloys.
- Properties and characteristics of Al-Li alloys, and alloy 2090 in particular, have improved over the last five years. This is typical of a new materials technology climbing its experience curve.
- Taking advantage of the lessons learned we believe that successful implementation requires:
  - Selection of applications with full definition of performance requirements.
  - Extensive communication with designers, manufacturers and users.
  - An incremental building block approach to achieve comfort with the use of a new material, starting first with secondary structure applications.
- The integrated (team) approach that is so essential for success in advanced composite product development should be exploited to implement Al-Li alloys in structural applications.

JT8D ENGINE RELIABILITY AND STRUCTURAL INTEGRITY ANALYSIS

1989 USAF STRUCTURAL INTEGRITY PROGRAM CONFERENCE

DECEMBER 5-7, 1989

BY:
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San Antonio, Texas 78228

The Federal Aviation Administration (FAA) has an aggressive program on the problem of aging aircraft. At the heart of this aging aircraft issue is the structural integrity of not only the aircraft structure but other major subassemblies, such as power plants. The most notable example of fuselage structural integrity problems was provided by the April 28, 1988, failure of the upper front half of an Aloha Airlines B-737 aircraft. Numerous actions have been taken for assessing and maintaining the structural integrity of B-737 fuselages, to include improved non-destructive inspection techniques, corrosion control procedure enhancements, and a review of the cumulative pressurization cycles on the aircraft skin.

The FAA held an International Conference on Aging Airplanes on June 1-3, 1988, to review the overall problems. The conference was conducted in several working group sessions, one of which involved power plants. There was general agreement by the participants in this engine subpanel that aircraft engines do not age like the aircraft structure. This aging difference is due to structured programs of inspections, tear-downs, and shop visits. Data from these structured programs is collected by Service Difficulty Reports (SDR's) and forwarded to the FAA. However, some airline representatives in the meeting said the FAA distribution of this information was of little use to them since the shop findings of the initially reported failure were rarely included in the report. Therefore, without some specific cause and effect documentation of an engine component failure, reliability performance trending was impossible.

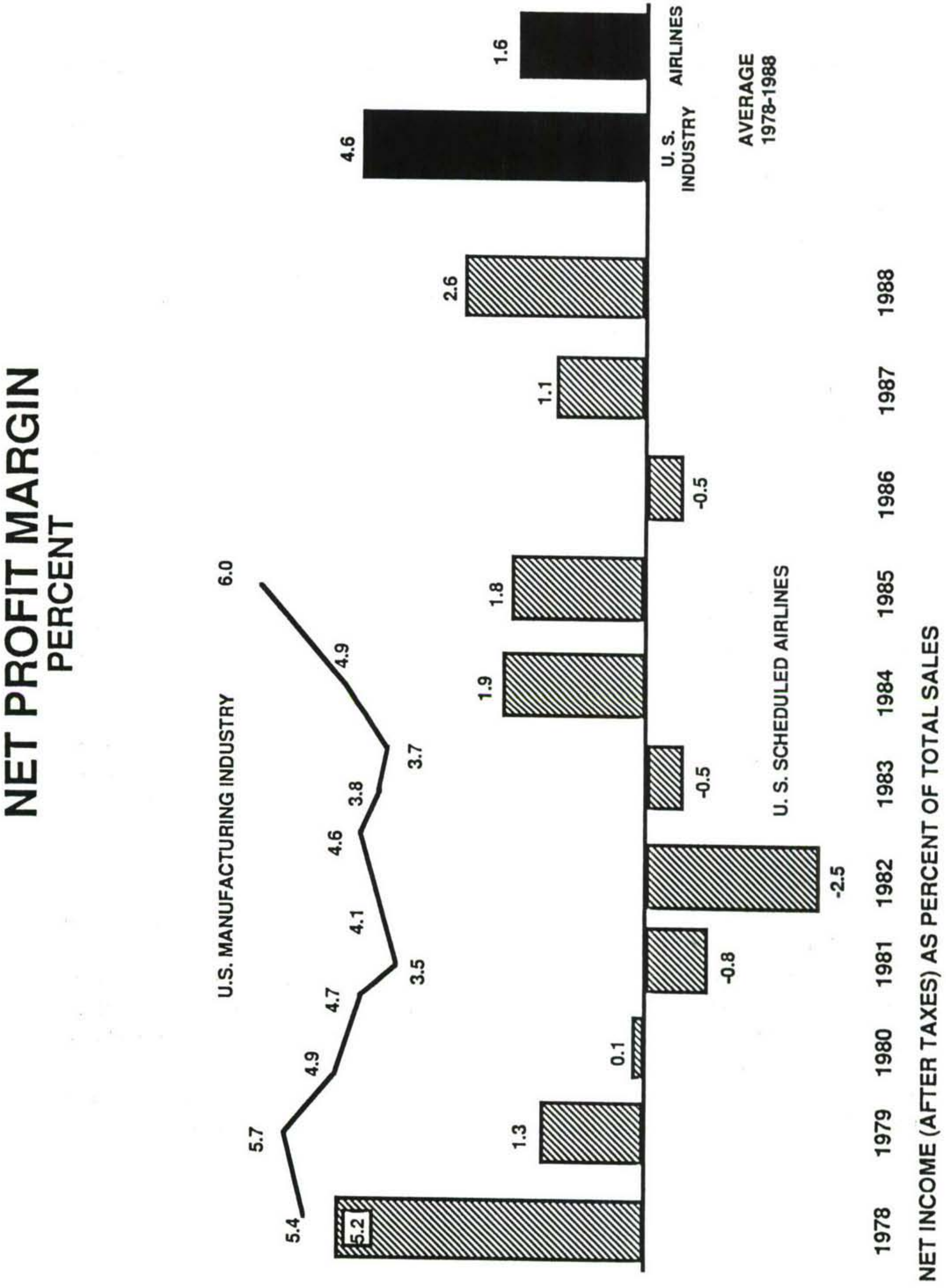
The engine manufacturers had some equally discouraging comments about the incorporation of Service Bulletins issued on engine components. The engine component reliability enhancements developed by the engine manufacturers are generally introduced by Service Bulletins. However, there is no central trending available regarding the incorporation of SB's, particularly for specific engine components. If the incorporation of a SB, or group of SB's is considered serious enough by the FAA for mandatory compliance, then an Airworthiness Directive (AD) is issued. The Airlines must document their compliance of AD's.

Given the above procedural guidelines and the economic impact of airline deregulation, it is reasonable to assume that some airlines with cash flow problems may not be incorporating engine reliability enhancements offered by the engine manufacturers. The October 20, 1989, issue of Aviation Week and Space Technology provided the chart shown in Figure #1 of the net profit margin of the major United States Scheduled Airlines over the last ten years. Considering the negative cash flow impact of the 1981-83 timeframe upon decisions to buy and incorporate reliability engine modifications, it is reasonable to assume some airlines made the decision to not buy some enhancements. Reliability enhancements can take from two to five years to be completely incorporated, based upon the maintenance concept and shop visit rate of specific engines. For some airlines, the lack of engine reliability investment in 1982 and 1983 has now caught up with them in a high rate of inflight engine shutdowns, and engine removals.

Responding to the engine reliability trend, Science Applications International Corporation, SAIC, has initiated a reliability analysis program on the JT8D engine with the FAA. This effort involves two major areas of JT8D engine reliability improvements: modification reliability enhancements, and improved nondestructive inspection procedures of engine cases for structural integrity verification.

With over 5,000 JT8D engines delivered to the commercial airlines and the multiple airframe application on the B-727, DC-9 and B-727 fleets, this engine is a good candidate for a reliability analysis. The twenty-year age of the JT8D inventory has resulted in over 2500 Service Bulletins being issued on JT8D components.

The initial reliability scan will be accomplished using FAA Propulsion Actuarial data on all commercial carriers. In-flight engine shutdowns and engine removals will be trended, by month, with variances from the norm plotted. The trend will be carried by engine/airframe applications so three separate airframe plots will be made. For this initial analysis, an airlines' monthly performance that is one standard deviation from the norm is considered yellow (possible reliability problem) while variances greater than one standard deviation are considered red (definite reliability



NET PROFIT MARGIN
PERCENT
U.S. MANUFACTURING INDUSTRY
5.4
5.7
4.9
4.7
3.5
4.1
4.6
3.8
3.7
4.9
6.0
U. S. SCHEDULED AIRLINES
5.2
1.3
0.1
-0.8
-2.5
-0.5
1.9
1.8
-0.5
1.1
2.6
4.6
1.6
U. S. INDUSTRY
AIRLINES
AVERAGE 1978-1988
1978
1979
1980
1981
1982
1983
1984
1985
1986
1987
1988
NET INCOME (AFTER TAXES) AS PERCENT OF TOTAL SALES

Figure #1

problem). Figure #2 shows the reliability trend of B-737/JT8D performance and DC-9/JT8D performance for a ten month period.

Figure #3 documents the performance of the B-727/JT8D inventory. One can note that multiple entries in a given month by a specific airline means two segments of its fleet were at variance to the norm for that month. A number in parenthesis (7/17/15) reflects the dash model of the JT8D inventory.

Using this trending as a starting point, JT8D engine component failure modes will be cross-referenced for those carriers with excessive shutdowns and removals. Finally, reliability enhancements applicable to the documented failure modes will be recorded. A comparison of reliability enhancement incorporation to inflight performance should reveal the effectiveness of specific airline engine maintenance programs.

The engine subpanel of the June '88 International Conference on Aging Airplanes also identified the single biggest problem for engine structural integrity: lack of a nondestructive inspection procedure for engine cases and frames. The composite maintenance concept for the airline engines stresses maximum use of on-aircraft maintenance and phased maintenance of rotors and static parts based upon life limits. However, since engine cases and frames provide the skeleton to which other components are attached, these cases and frames are rarely removed and inspected. Some cases now have in excess of 30,000 hours of exposure. Without specific life limits assigned to cases, there is no requirement for a shop removal and in-depth inspection cycle.

The airlines have placed as a high priority, the development of life-predictive methodologies for both repaired and non-repaired engine cases and frames. The purpose would be to determine the useful service life and inspection intervals for cases. Emphasis is to be placed upon the development of on-wing inspection techniques so as to not disrupt current maintenance practices and intervals.

For this portion of the JT8D reliability task, SAIC plans to use the Ultra Image III system, a portable microprocessor-controlled ultrasonic imaging and analysis system. The system consists of four units: the

PRELIMINARY TREND OF AIRLINE JT8D RELIABILITY PERFORMANCE
FEBRUARY THRU NOVEMBER 1988

B/737/JT8D

AIRLINE	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEPT	OCT	NOV
AL #1	Y	Y	---	---	---	---	Y	---	---	---
AL #2	---	Y	---	---	---	---	Y	---	Y	Y
AL #9	R	---	---	---	R	---	---	Y	---	---
AL #8	---	Y	R	---	---	---	Y	---	---	---

DC-9/JT8D TREND

AL #1	Y Y(7)	Y	---	R(7)	R	---	---	---	Y	---
AL #5	R(17) Y(7)	---	---	---	---	Y	R	---	Y	Y
AL #6	---	Y(7)	Y	Y	---	---	R	---	---	Y(17) Y(15)

Figure #2

PRELIMINARY TREND OF AIRLINE JT8D RELIABILITY PERFORMANCE
FEBRUARY THRU NOVEMBER 1988

B727/JT8D TREND

AIRLINE	FEB	MAR	APR	MAY	JUN	JUL	AUG	SEPT	OCT	NOV
AL #1	R	R	R	---	---	Y(7) Y	Y	---	---	---
AL #2	---	---	---	---	---	---	Y	---	Y	Y
AL #3	R	Y	---	R	R	R	Y	R	---	Y
AL #4	R	---	---	---	---	---	Y	Y	R	Y
AL #5	---	---	---	R	---	Y	Y	Y(15) Y(7)	Y	Y
AL #6	---	---	Y(7)	Y	Y	---	---	---	---	Y(17) Y(7) Y(15)
AL #7	---	Y	---	R(7) Y(9)	---	---	Y	---	---	---
AL #8	---	Y	---	R	---	Y	Y	Y(7) Y(15)	---	---

Figure #3

microprocessor, the ultrasonic pulsar/receiver/gate package, the dual diskette drive, and a display package of a 5 inch black and white monitor and a 3.5 inch cathode-ray tube (CRT). Each system weighs less than 30 pounds and the system can be set up for operation by trained personnel in 15 minutes.

Calibration of the system is performed on a step block with known thickness. Using the second multiple of reflections, in the calibration block, the system eliminates errors due to variances in transducers, delay lines, couplant, and electronics. System controls are monitored by the microprocessor to ensure that the system is, and remains, in calibration throughout the data acquisition period. All set-ups and the result data are permanently recorded on a dual-sided, double-density floppy diskette which can contain information from up to 15 tests.

The inspection results are easily analyzed through an interactive color graphics system which provides instant viewing of the ultrasonic signal amplitude and depth over the scanned area immediately after the scanning process is completed. The image can be sliced in X or Y direction and enlarged to many times its original size to aid in analysis of deflect shapes.

For the JT8D cases of interest, the inspection focus will be placed upon the radius area immediately adjacent to the front and rear flanges. It is this area that receives the greatest thermal expansion stress and presents the most difficult surface for on-wing inspection. Defective and serviceable cases will be used for procedural prototypes.

This task is expected to take about twelve months for completion and will be jointly sponsored by the Air Force's Commerical Aircraft Division at Tinker AFB and the FAA. The information gained will add to the FAA's growing data bank on flight safety information. The NDI procedure development will be applicable to numerous Air Force and airline engines and will be a contribution towards the engine structural integrity program for airline engines.

Air Force One
ENSIP Assessment

Premier Application on Commercial Engine

1989 United States Air Force
Structural Integrity Program Conference
San Antonio, Texas
5-7 December, 1989

S.R. Staffier
GE Aircraft Engines
Capt. G. Chestnut
WPAFB

MAC9-49141-111489

Presentation Overview

- Background and general overview of ENSIP assessment program
- Outline of accomplishments with specifics on special items of importance
- Significant program results and lessons learned
- ENSIP assessment benefits for Air Force One

MAC9-49142-111489

Air Force One – What Is It?

- Two aircraft (Boeing 747-200)
- Ten engines – CF6-80C2B1 (F103-GE-102)
  - Military application of commercial engine
- Important, prestigious program for USAF, GEAE and Boeing

MAC9-49143-111489

Air Force One
ENSIP Assessment

Premier Application on Commercial Engine

General Electric Aircraft Engines
Boeing Military Aircraft Company
United States Air Force

T2140.02 · 890919

Air Force One ENSIP Overview

- First ENSIP assessment of an “off the shelf” commercial engine

- Analytical and experimental program on the CF6-80C2 engine
  - Prediction of major component fatigue and damage tolerance capability
  - Conducting engine and subcomponent tests to support analyses
  - ENSIP engineering development programs (materials data, life methods)
  - Generation of ENSIP driven damage tolerance inspection interval information

MAC9-49144-111489

Air Force One ENSIP Overview (Continued)

- Planned output
  - Assessment of application of ENSIP (Mil-Std-1783) design criteria to commercial engine environment
  - ENSIP documentation (plans and reports)
- Assessment program milestones
  - Request for proposal: 12/85
  - Authorization to proceed: 1/87 (1 year proposal negotiation)
  - Scheduled completion date: 6/89 (2-1/2 year technical program completed)

MAC9-49145-111489

Air Force One ENSIP Accomplishments

- GEAE has completed comprehensive 2-1/2 year CF6-80C2 ENSIP assessment
  - Stress and life analysis of 140 separate components
  - Crack propagation data generation for 4 materials
  - Probability of detection data generation for 3 materials
  - High pressure compressor thermal data (full scale fan engine test)
  - Turbine rear frame hub full scale heat transfer test
  - Full scale compressor rear frame cyclic fatigue and damage tolerance test
  - Full scale, 3D photoelastic, forward fan shaft test

Air Force One ENSIP Accomplishments
(Continued)

- Four crack growth methodology development programs
- Controls accessories and externals methodology development program
- Casting material data scatter reliability assessment program

• Program plan and results extensively documented . . . over 2300 pages of reports

MAC9-49147-111489

Documentation

- Master plan
  - FAR 33 (FAA) vs. Mil-Std-1783 (USAF) comparison
- Durability and damage tolerance control plan
- Materials characterization plan
- Corrosion prevention and control report
- Durability and damage tolerance report
  - 3 books (text, figures and tables, test reports)
- Structural maintenance plan

MAC9-49148-111489

Major Components Analyzed

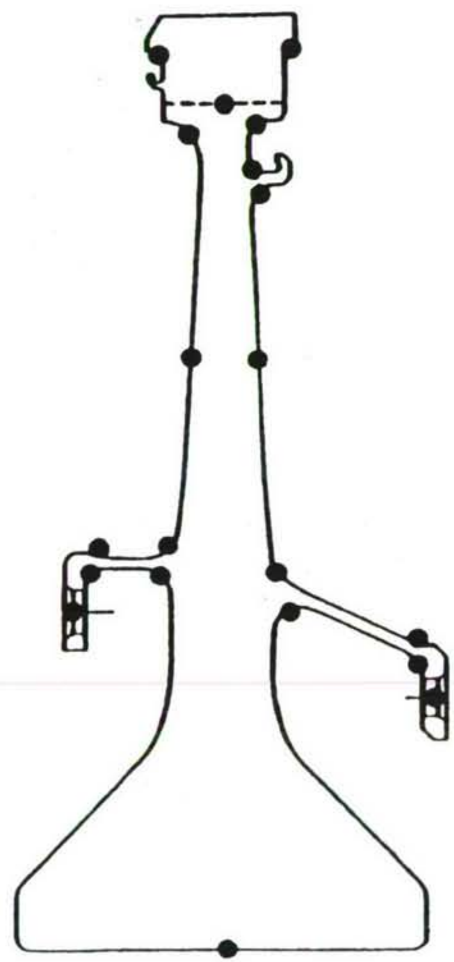
- Fan
- High pressure turbine
- Forward mount
- Turbine rear frame
- High pressure compressor
- Low pressure turbine
- Compressor rear frame

MAC5-47893-100489

Analyses Performed

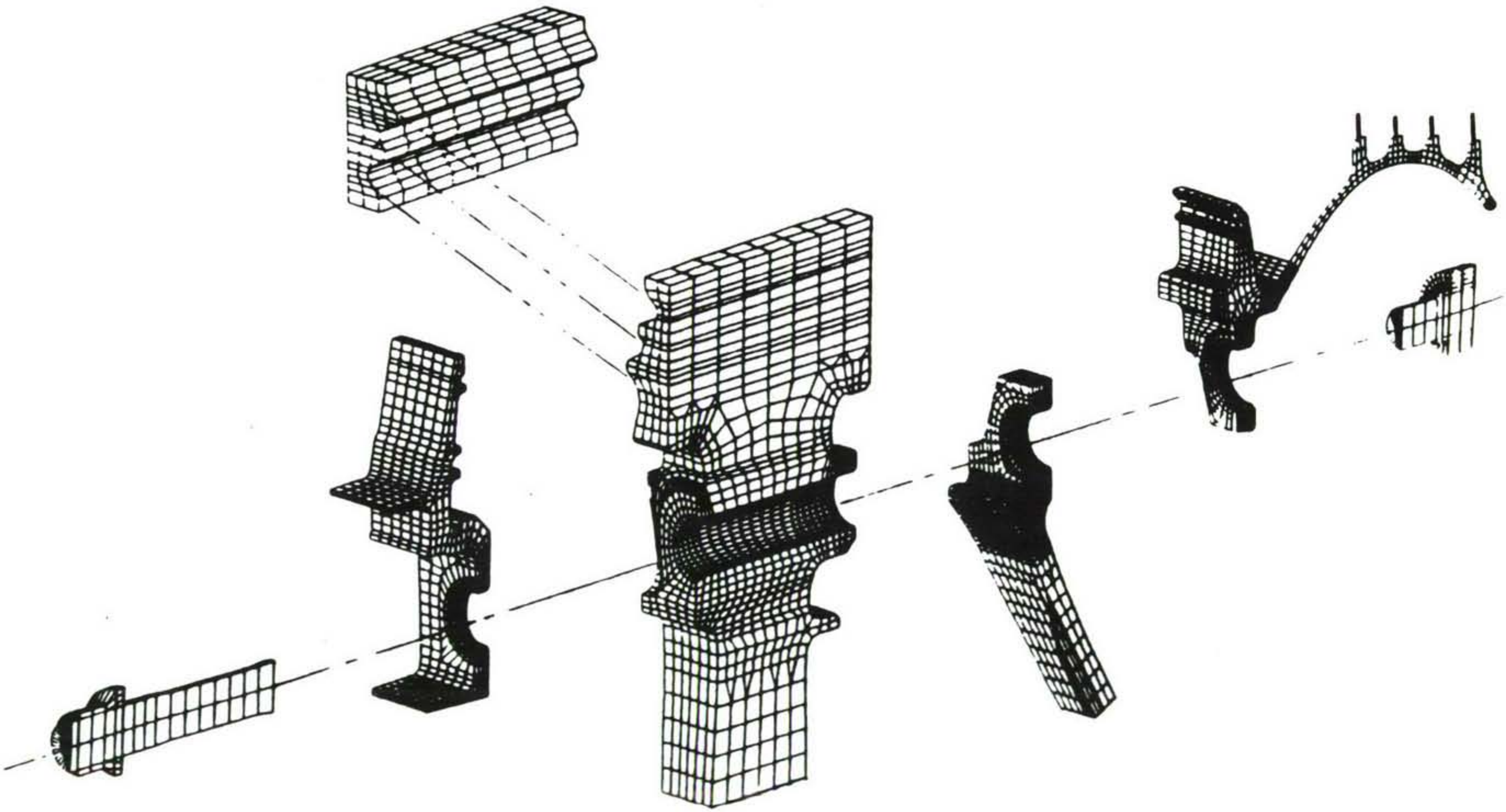
- Thermal
- Stress
- Strength/creep
- Low cycle fatigue
- Damage tolerance
  - Surface
  - Subsurface

Typical locations studied in detail



MAC9-49149-111489

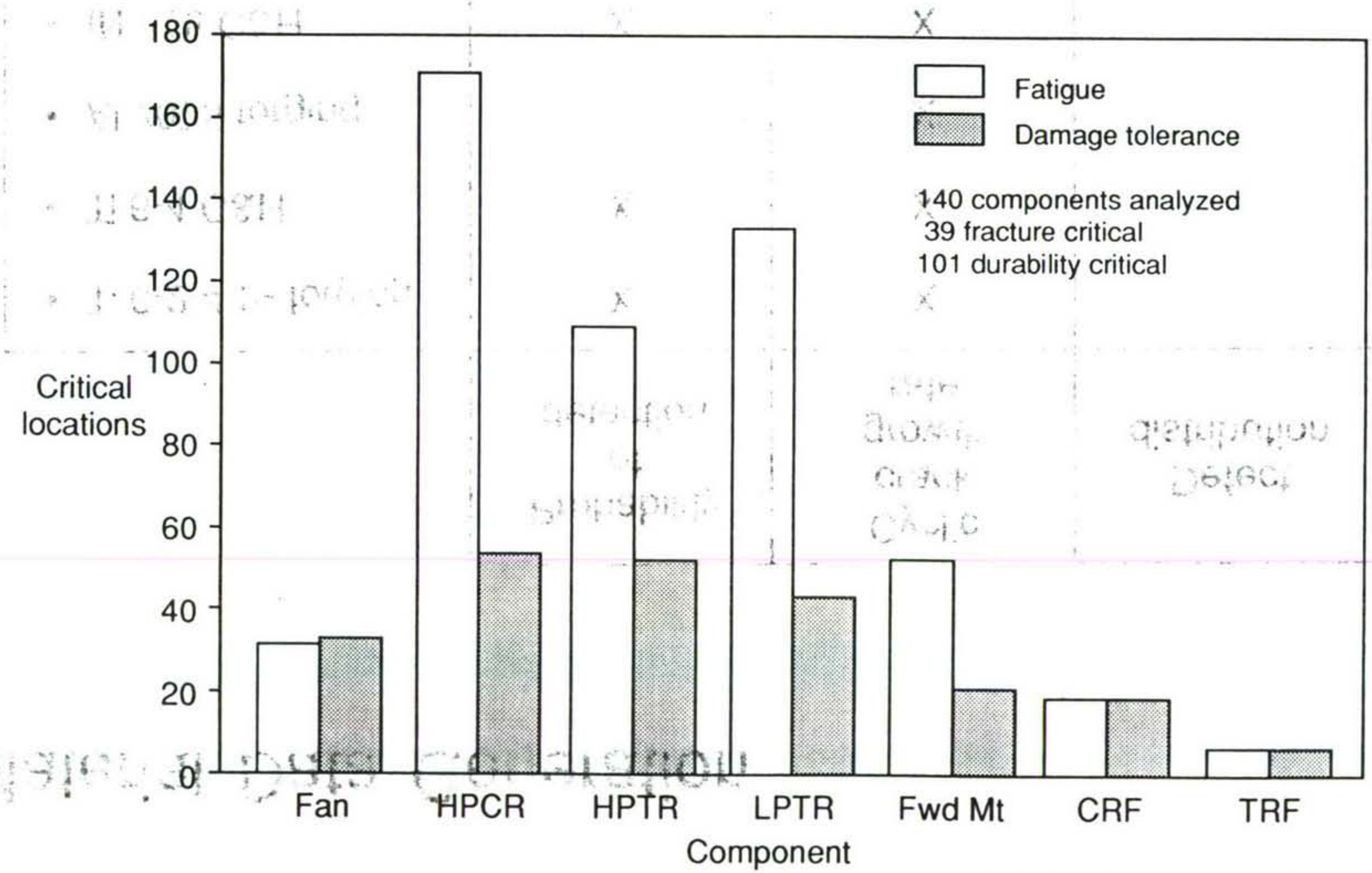
Typical 3D Finite Element Model



MAC9-49150-111489

Air Force One ENSIP Analyses

Some Quantities of Interest



MAC9-49151-111489

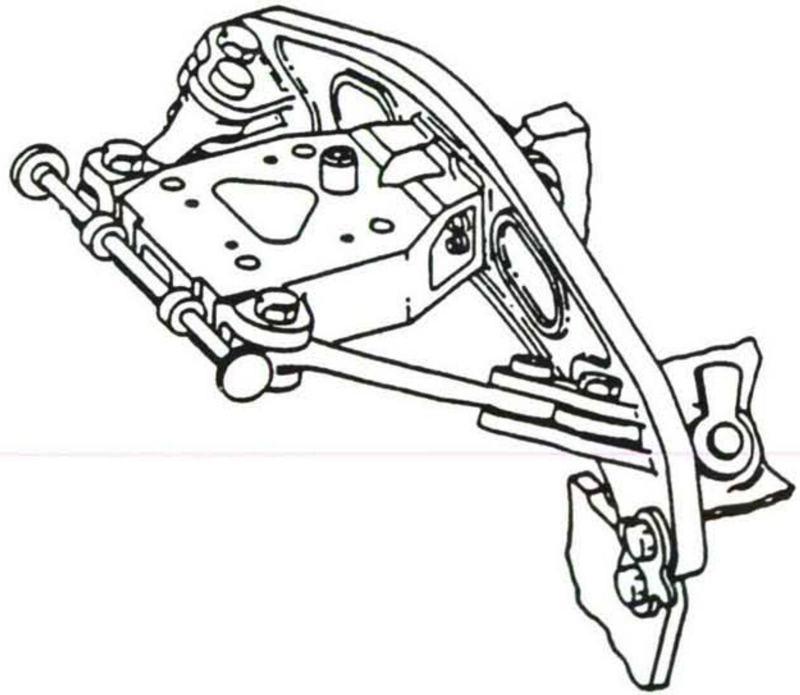
Material Data Generation

	Probability of detection	Cyclic crack growth rate	Defect distribution
• Ti 6-2-4-2- forging	X	X	
• Ti 6-4 C&H	X	X	
• Al 7075 forging		X	
• IN 718 C&H	X	X	
• IN 718 forging			X

MAC9-49152-111489

Analytical Methods Development

- Improved residual life models for
  - Forward mount
    - Links
    - Bolts
    - Platform and yoke
  - Compressor rear frame
    - Combustion case pads

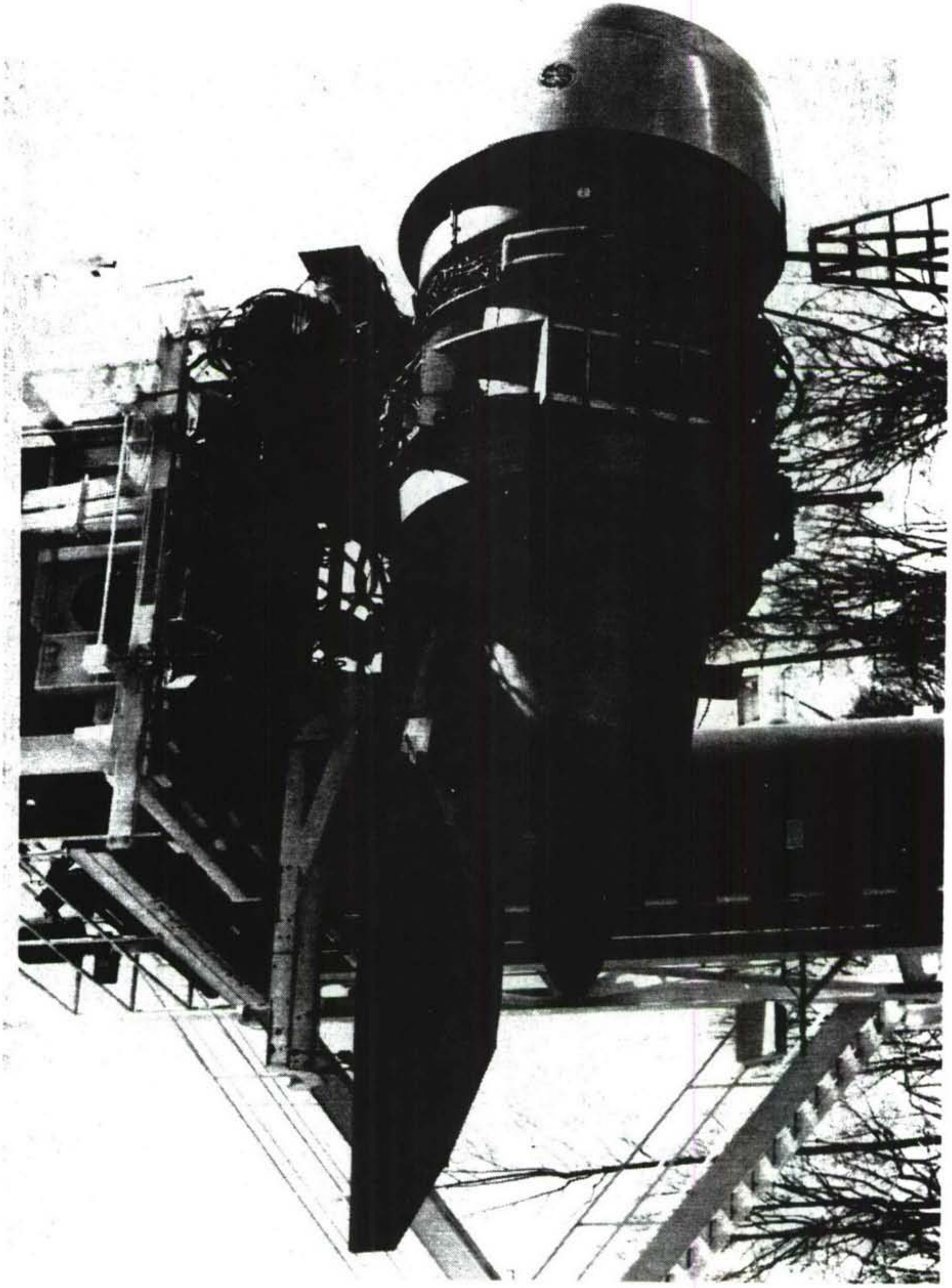


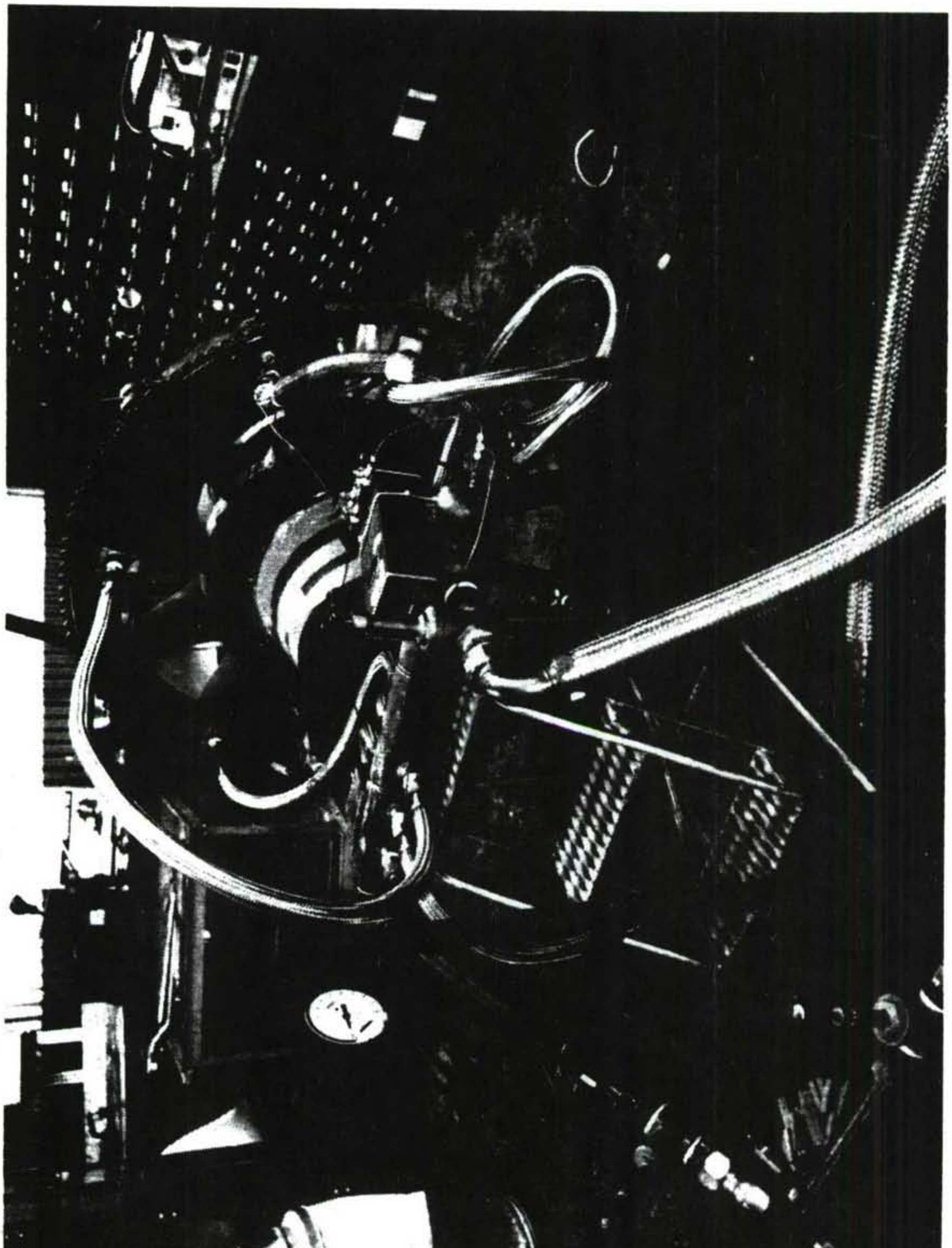
MAC9-49153-111489

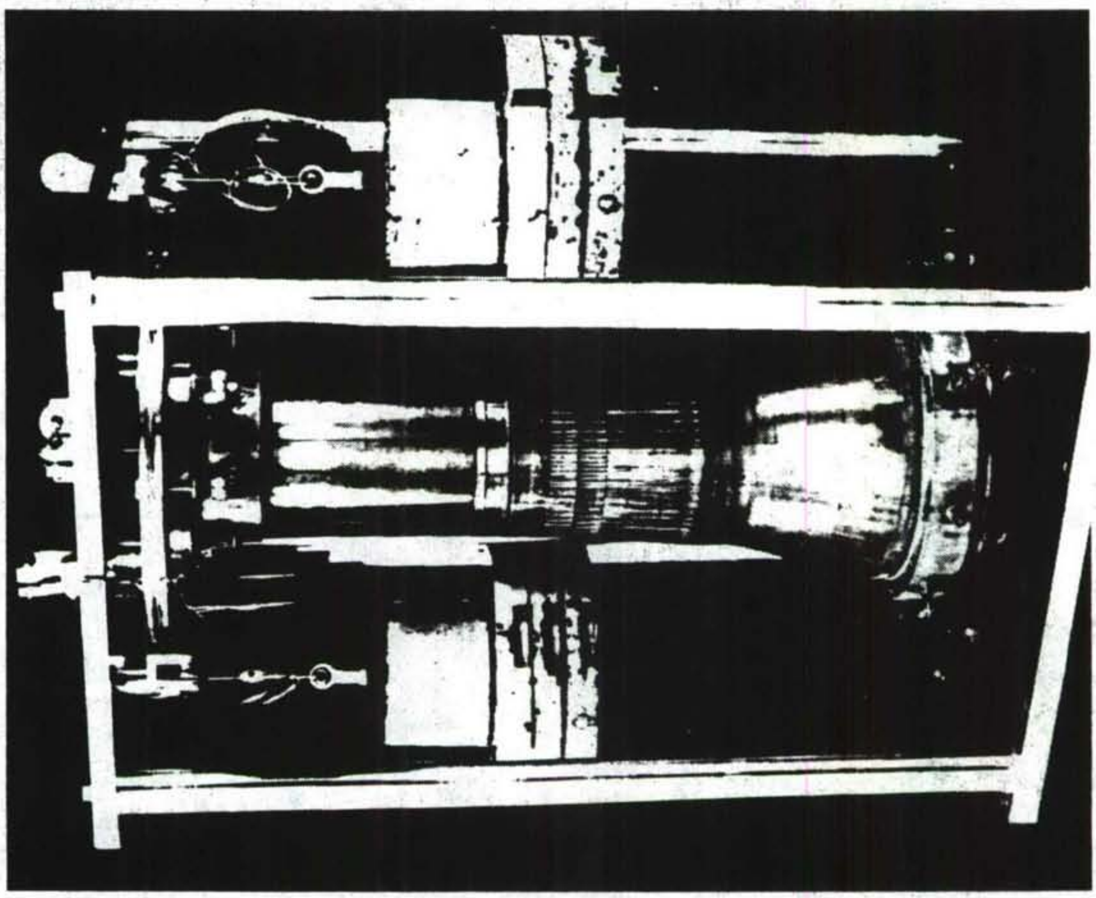
Supporting Tests Performed

- Fan engine test
  - Additional verification of steady state and transient temperature predictions for the high pressure compressor
- Turbine rear frame heat transfer update
  - Steady state (heat transfer coefficients)
  - Transient (thermal response)
- Fan forward shaft spline photoelastic test
  - Spline stresses for LCF and residual life analysis
- Compressor rear frame hydraulic pressure test
  - LCF life verification
  - Residual life verification

MAC9-49154-111489







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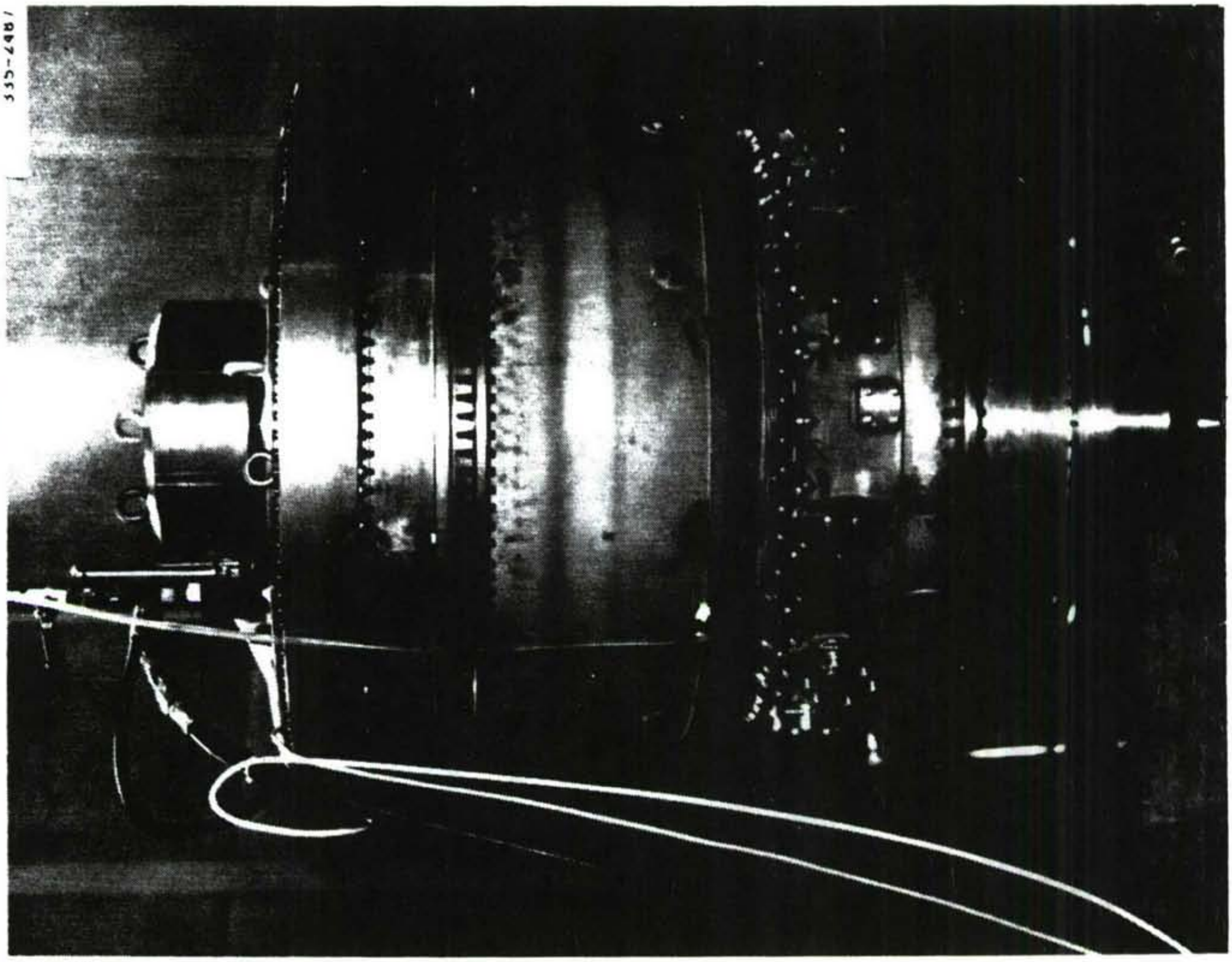


Results Overview

- Comprehensive tabulation of predicted component fatigue and residual lives for a wide range of critical engine hardware
- Based on analysis results and extensive positive fleet experience, recommended use of successful commercial practice – “on condition” maintenance
- Provided all required residual life and NDE data for possible enhanced field inspection and implementation requirements
- Initiated life enhancement activities on three components

MAC9-49155-111489

Commercial Engine in Military Application – Lessons Learned

- Assessment design data
  - Mission use and life requirements must be targeted at beginning of program
- Formats for reports
  - Formats must be tailored between contractor and customer to meet objectives
- Chain of communication
  - In comprehensive integrity program "shorter is better"
  - Direct interface between contractor and USAF

MAC9-49156-111489

Commercial Engine in Military Application – Lessons Learned (Continued)

- Proprietary data (nontechnical “drag”)
  - Commercial engine business sensitivity associated with misinterpreted design criteria and program results
  - Limited/unlimited data rights negotiated up front
  - 2-1/2 year behind the scenes effort (special rights, report annotation, deliverable/nondeliverable CDRLs, etc.)

Recognize Environments Are Different and Factor Into Program Plan

MAC9-49157-111489

ENSIP Assessment Benefits For Air Force One

- ENSIP data provided for Air Force One maintenance planning
- Enhanced life component designs
  - High pressure compressor stage 1 dovetail
  - Fan spinner
  - Fan forward shaft spline
- Ground work laid for other programs
  - C-17
  - C-27

    •

    •

    •

    ?

MAC9-49158-111489

Air Force One Summary

- Comprehensive assessment successfully completed!!
- Resulted in benefits to the USAF via maintenance planning data and improved life designs
  - Spinner, HPC dovetail, fan shaft spline
- Results extensively documented in formal reports
- Provided “lessons learned” for both customer and contractor
- Laid groundwork for other programs
  - C-17
  - C-27
  - Etc.

MAC9-49159-111489

Some Aspects of Probabilistic Fracture Mechanics In Relation To USAF ENSIP Damage Tolerance Philosophy

1989 USAF Structural
Integrity Program
Conference

P.G. Roth, A. Coles
GE Aircraft Engines
December 7, 1989

MAC5-49343-112089

Some Aspects of Probabilistic Fracture Mechanics In Relation To USAF ENSIP Damage Tolerance Philosophy

- Elements of a probabilistic design methodology
- ENSIP damage tolerance design considerations – Overview
- ENSIP damage tolerance – Issues and Evolution
- Present status and recent developments
- Application
- Summary

MAC5-49344-112089

Elements of a Probabilistic Design Methodology

• Failure modes	• Inspection
• Operational data	• Structural response
• Material characterization	• System integration
• Defect occurrence	• Experimental validation

- Comprehensive strategies being developed
  - Southwest Research Institute
  - Jet Propulsion Laboratory
- Natural evolution of ENSIP damage tolerance design philosophy
  - Many elements already in place

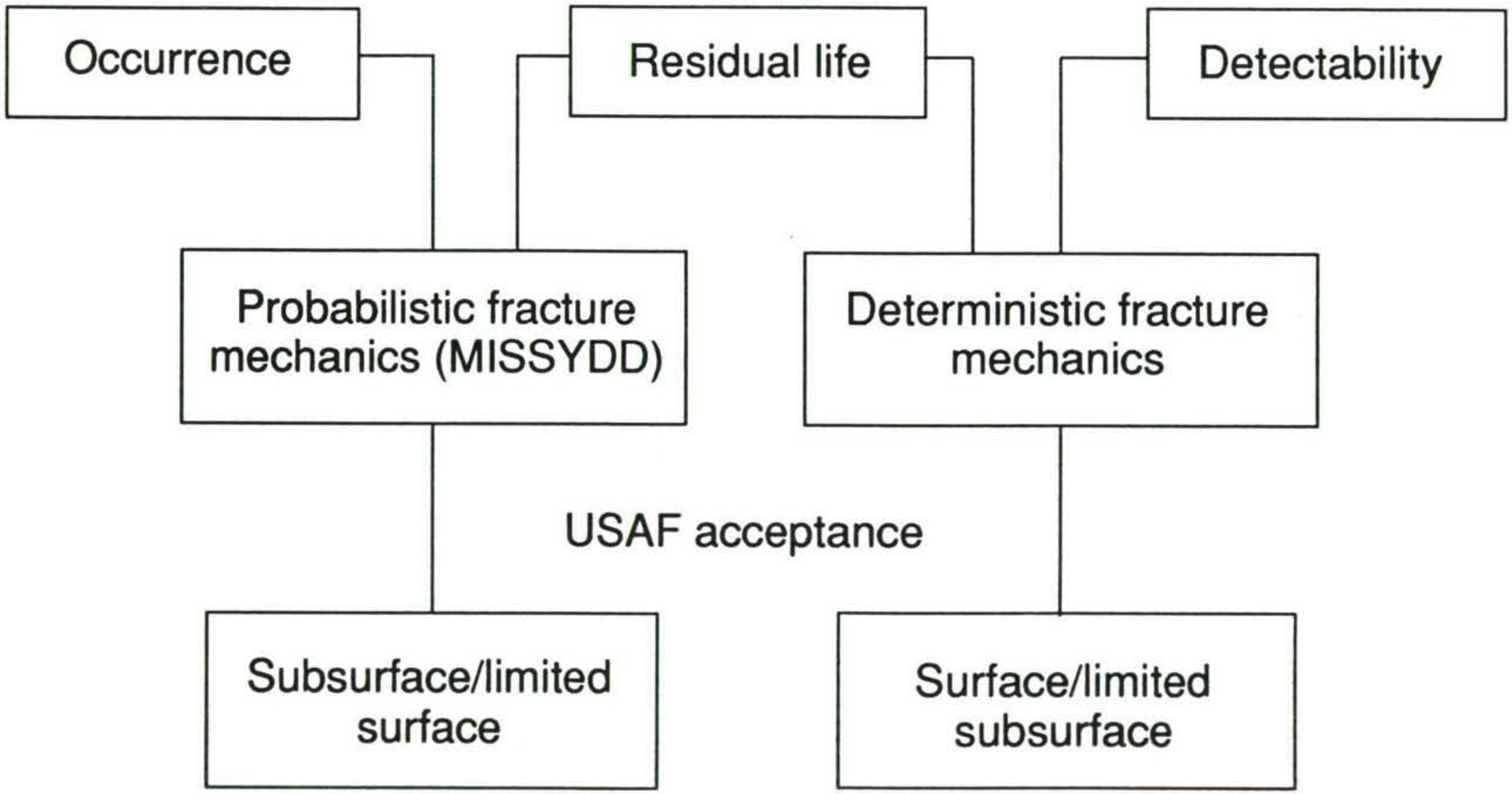
MAC5-49345-112089

ENSIP Damage Tolerance Design Overview

- ENSIP safety inspections reduce failure risk
- Deterministic framework defined
  - Structural integrity equated with absence of sharp cracks
  - Sharp cracks dichotomized: detectable or non-detectable
  - Non-detectable sharp cracks assumed present but predictable
- Refinements made
  - Inspectability defined statistically
  - Impact of embedded defects assessed probabilistically
  - Safety factors suggested by statistical analysis
- Statistical elements implicit in ENSIP

MAC5-49346-112089

ENSIP Damage Tolerance



MAC5-49347-112089

ENSIP Damage Tolerance – Issues and Evolution

- Definition of inspection intervals – issues

  <u>Detectability</u>
  - Sharp crack calibration possibly conservative . . . nicks and dings may be more inspectable
  - Spontaneous occurrence of large sharp cracks invalidates analysis

  <u>Residual life</u>
  - ENSIP damage tolerance addresses sharp cracks
  - Initiation activity not recognized

  <u>Occurrence</u>
  - Surface damage relatively rare
  - Critical zones localized
- Inspectable flaw 10% non-detectable
  - Risk believed small . . . as yet unquantified

MAC5-49348-112089

Rationale for Probabilistic Evolution

- There is always some risk
- Risks reduced by safety inspections
- Qualitative analysis requires:
  - Identification of the threats
  - Evaluating detectability
  - Assessing impact as crack initiators
- Quantitative analysis must be statistical
  - Utility in fleet management
  - Potential in design (USAF, NASA initiatives)
  - Attempt only when appropriate

MAC5-49349-112089

Present Status

- Probabilistic fracture mechanics gaining credibility in industry
- The GEAE program MISSYDD (<u>MISS</u>ion <u>SY</u>nthesis given <u>D</u>efect <u>D</u>istribution) has been applied to F101, F110-100, F110-129 and AF-1 CF6-80C hardware in ENSIP analysis of <u>embedded defect</u> damage tolerance with USAF acceptance

- Analysis developments have erased an artifical boundary with the surface, making exact the risk approach

MAC5-49350-112089

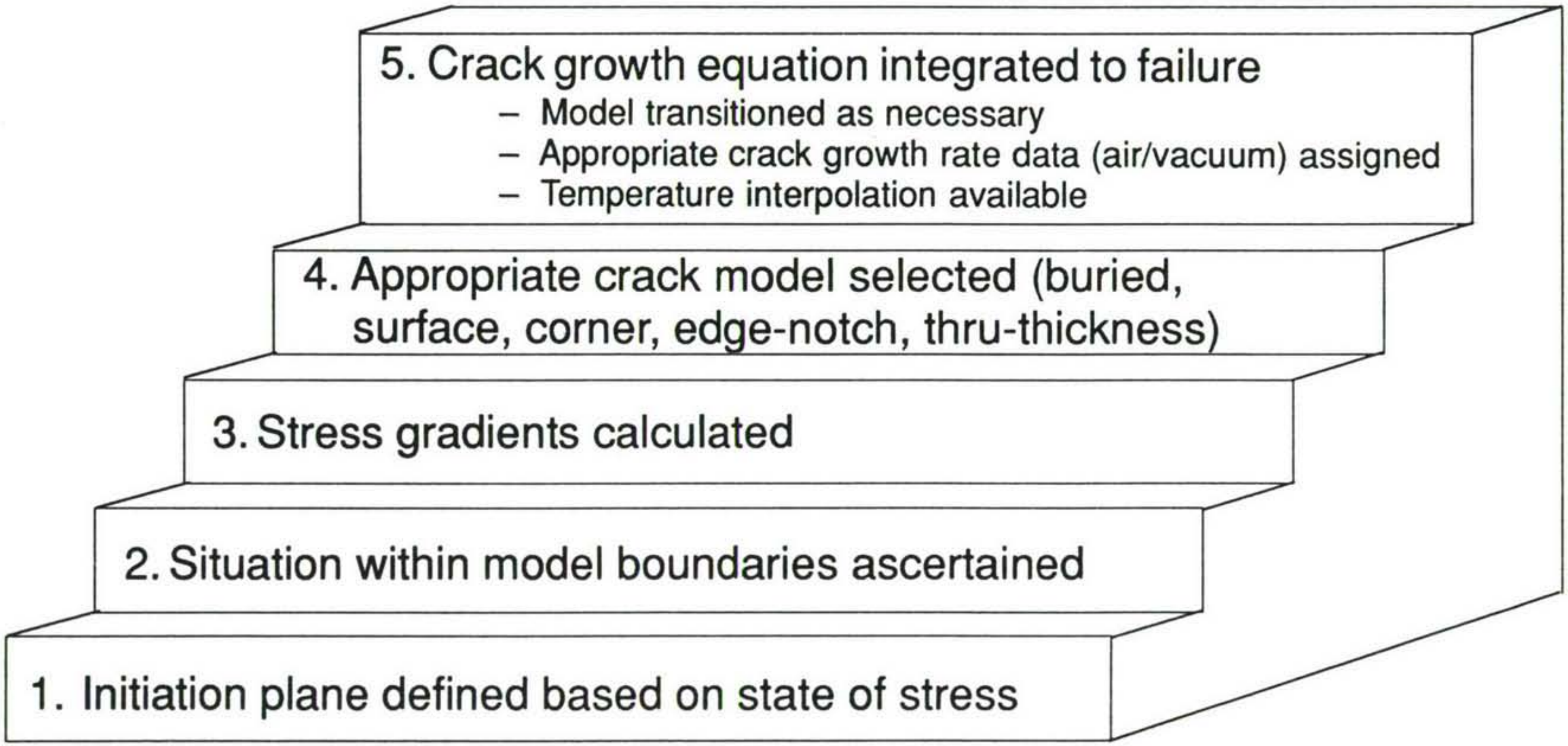
Outline of Recent Developments

- Statistical combination – three steps
  1. Single defects, specific sizes
     - Geometric effect of defect placement evaluated
  2. Single defects, random sizes
     - Geometric failure distribution integrated with defect relative size distribution
  3. Multiple defects, random sizes
     - Based on Poisson model for defect occurrence
- Flexibility enables calculation with very general distributional models

MAC5-49351-112089

- Geometric failure distributions evaluated by Monte-Carlo simulation
  - Defects randomly placed and fracture mechanics lives calculated
  - Convergence guaranteed
  - Adaptive Monte-Carlo algorithm allows adjustment of sampling rates
- Integration with defect distribution direct
  - Computationally intensive Monte-Carlo calculations minimized

- Given a defect location, major and minor axes:

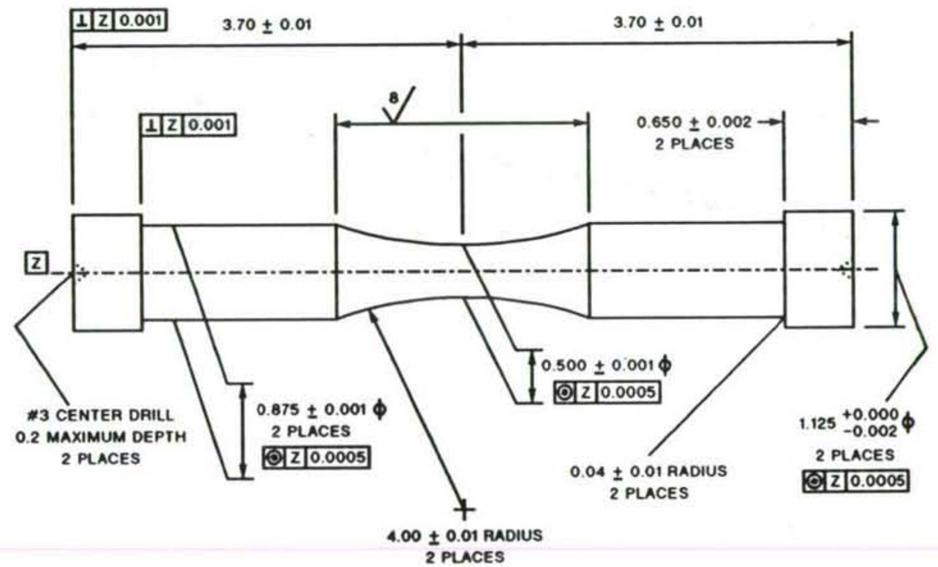


- Software recently developed for efficient computation of fracture mechanics lifetimes at any point of a finite element model (both 2-D and 3-D)

MAC5-49353-112089

Application: MISSYDD Verification Study

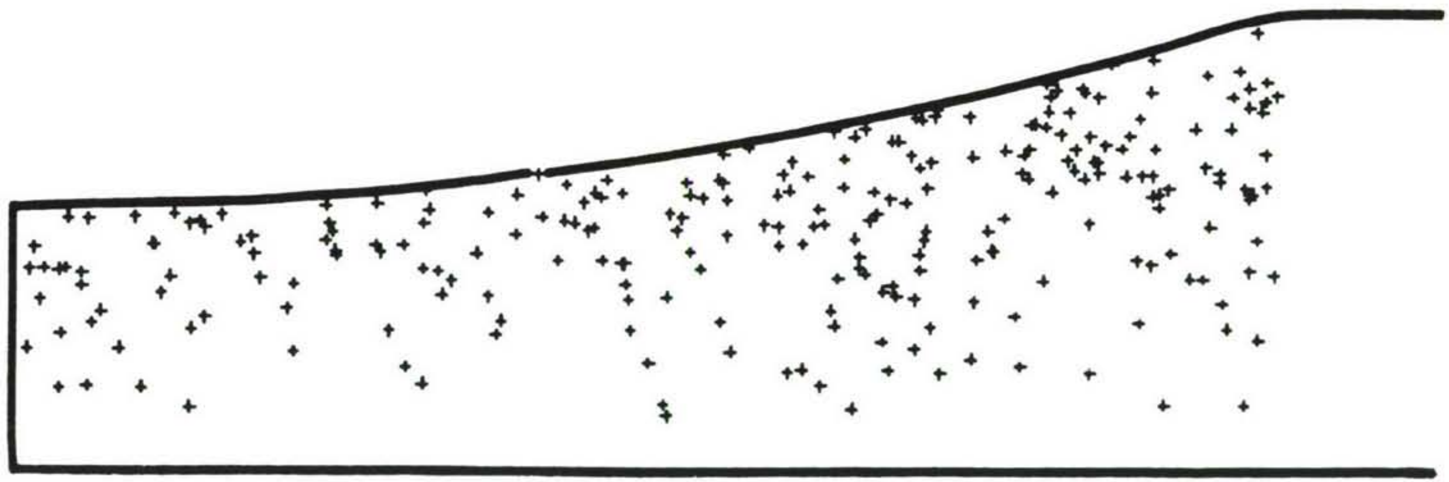
- Hourglass specimens manufactured from seeded as-HIP PM Rene'95 and fatigue tested



- MISSYDD predictions based on estimated seed distribution
- Observed lives of specimens failing from seeded defects were consistently longer than calculated . . . the comparison suggested a simple multiplicative adjustment

MAC5-49354-112089

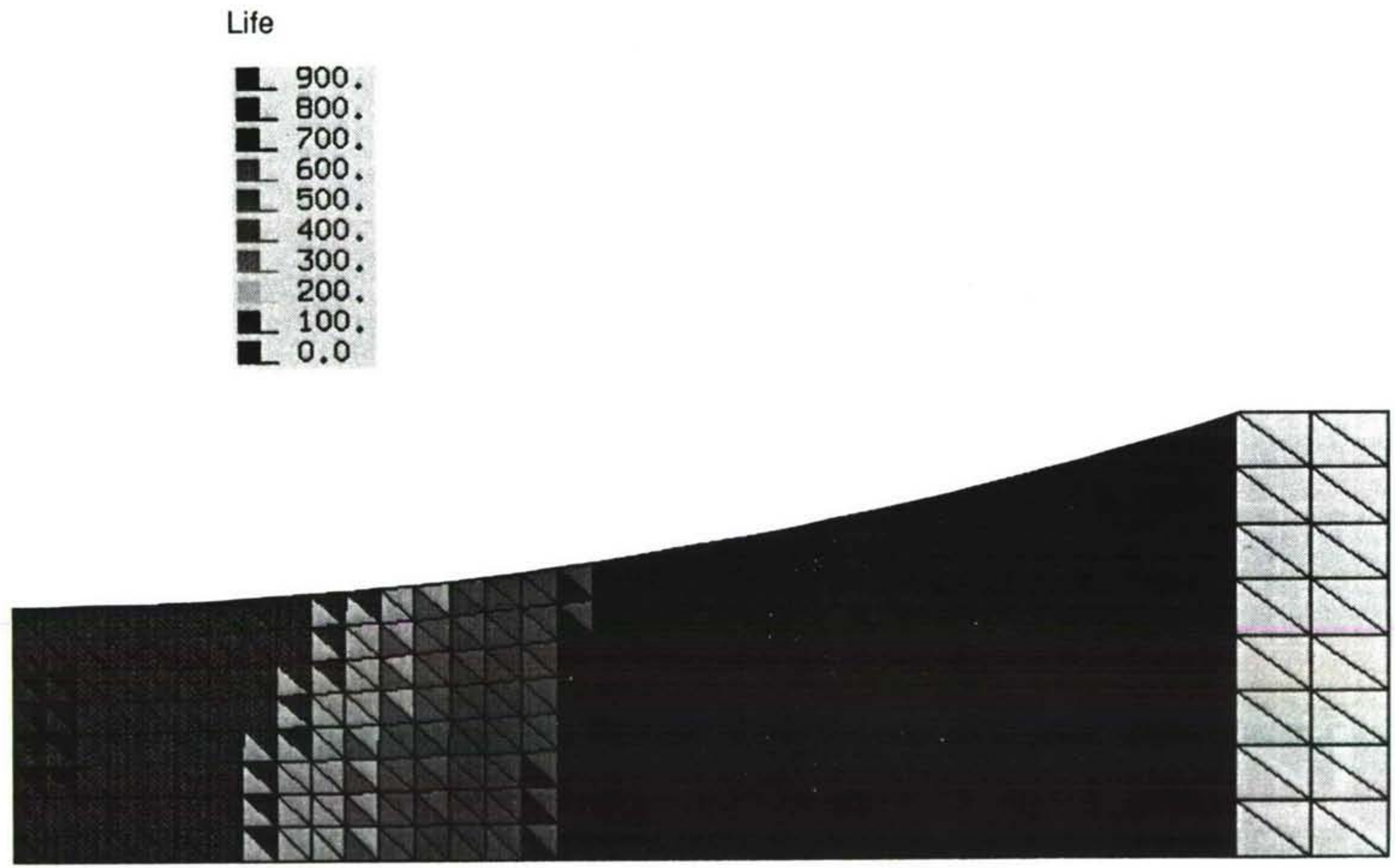
Monte-Carlo Placements, 250 Trials



MAC5-49355-112089

Geometric Failure Distribution

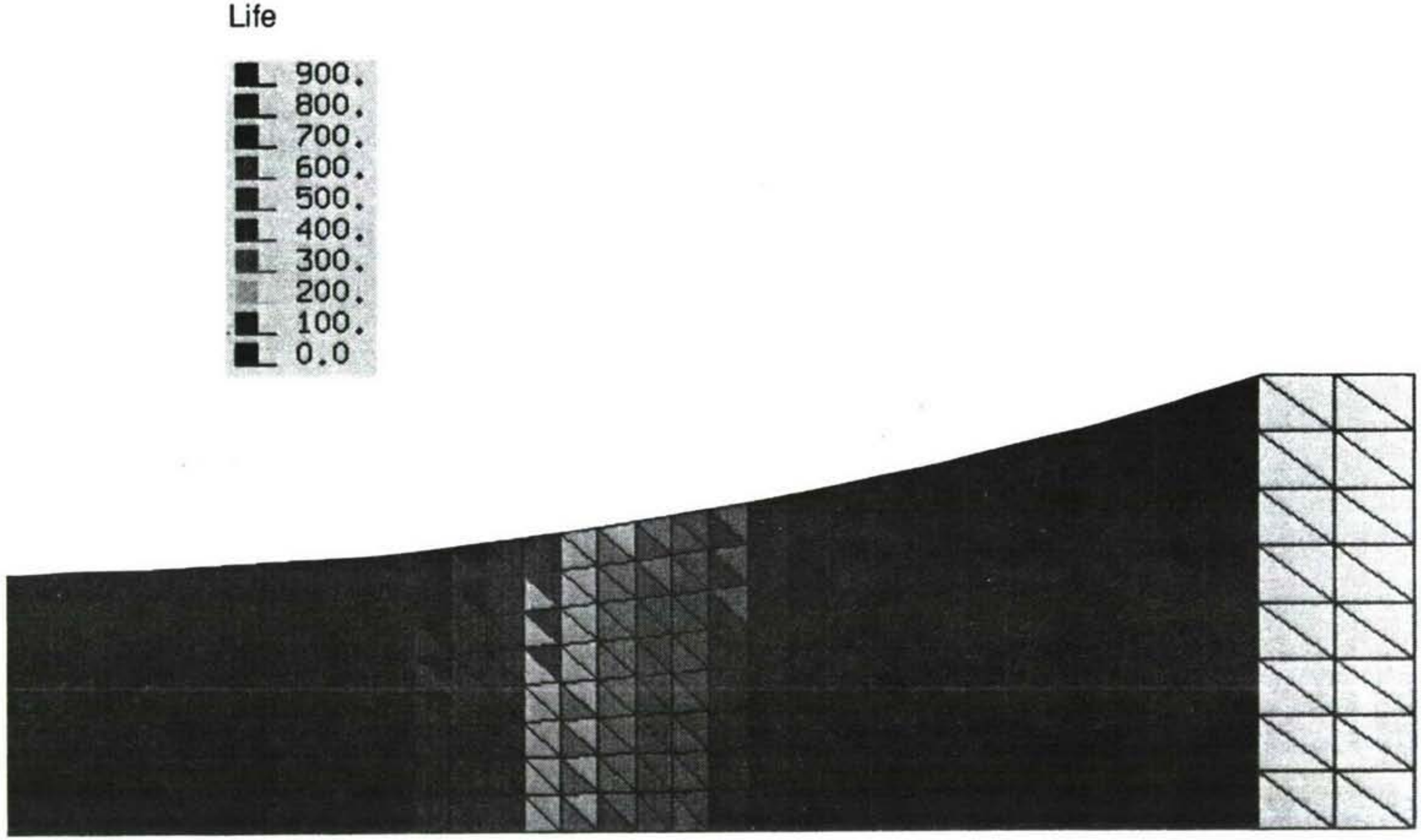
Defect Size: 1260 mil^2



MAC5-49356-112089

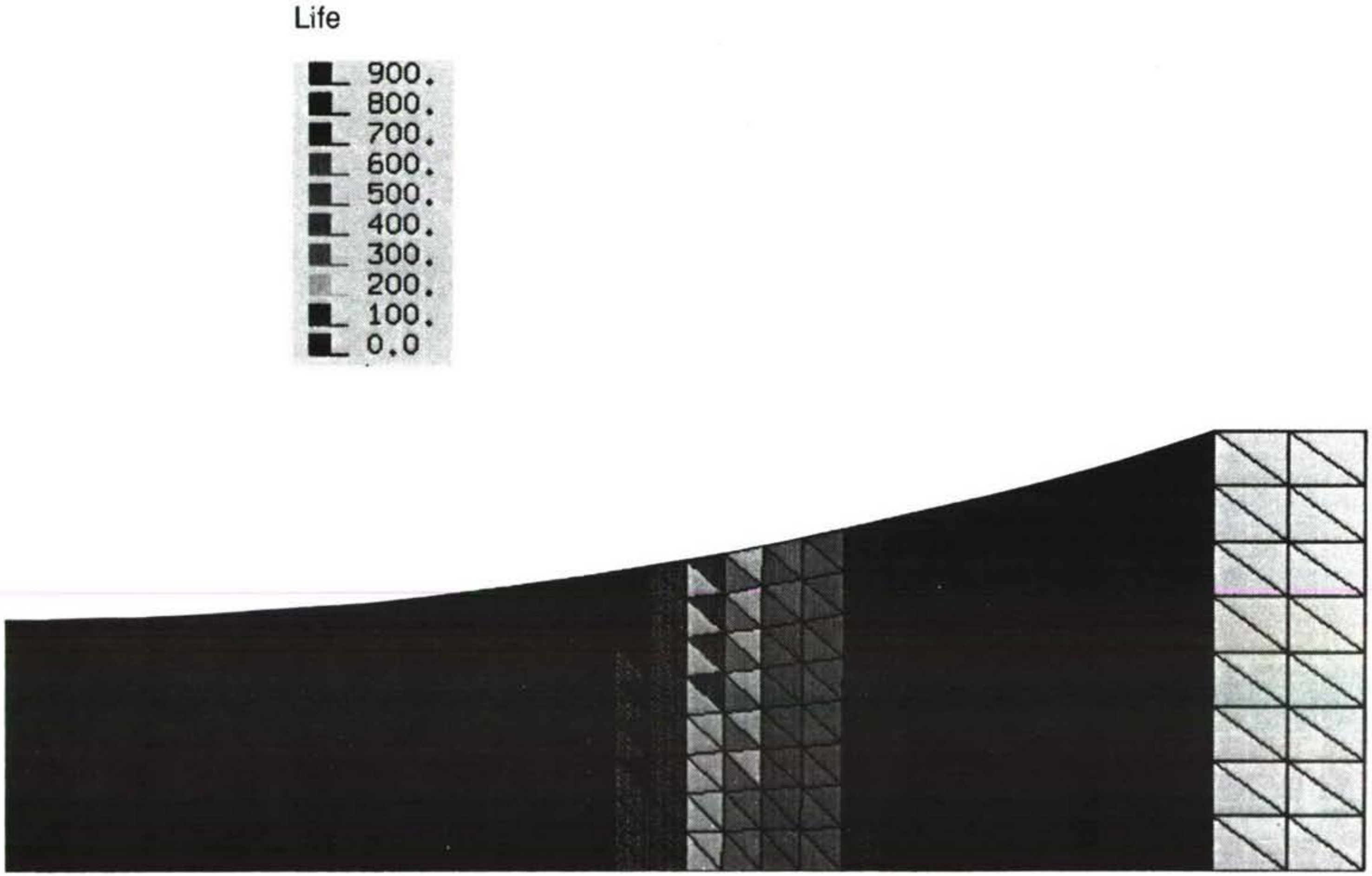
Geometric Failure Distribution

Defect Size: 3550 mil^2



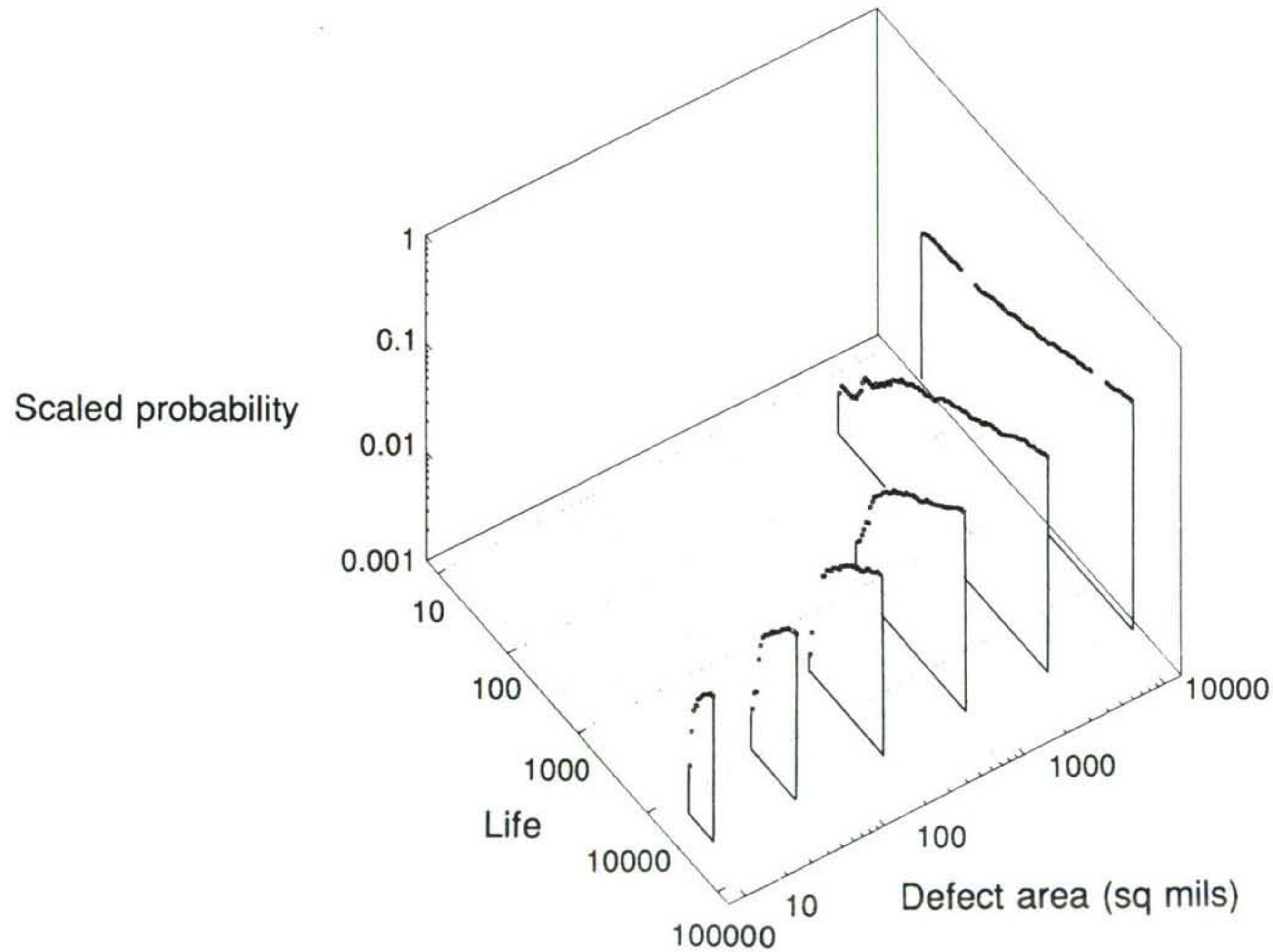
MAC5-49357-112089

Geometric Failure Distribution
Defect Size: 10000 mil^2



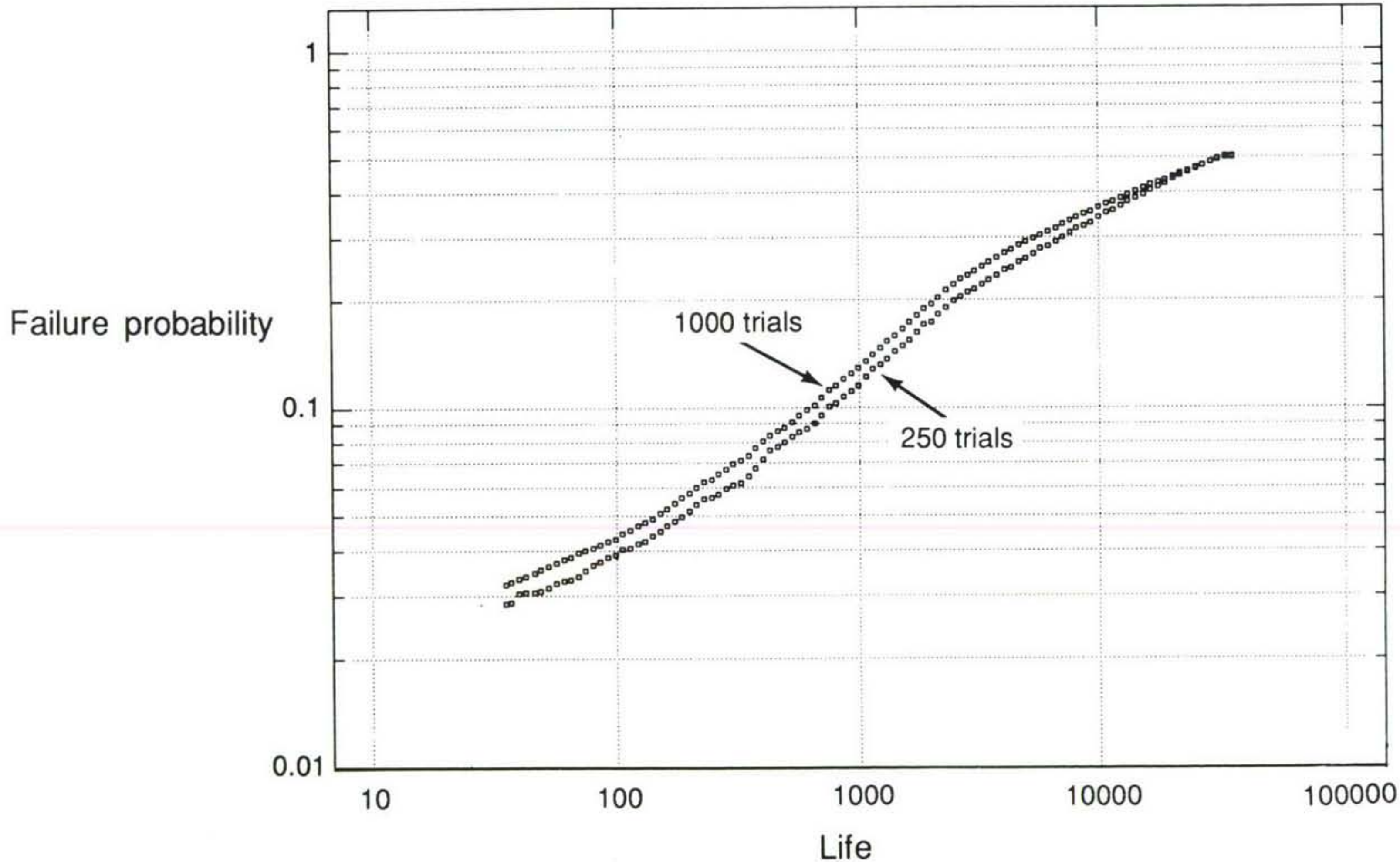
MAC5-49357A-112089

Scaled Geometric Failure Probabilities Hourglass Validation Specimen



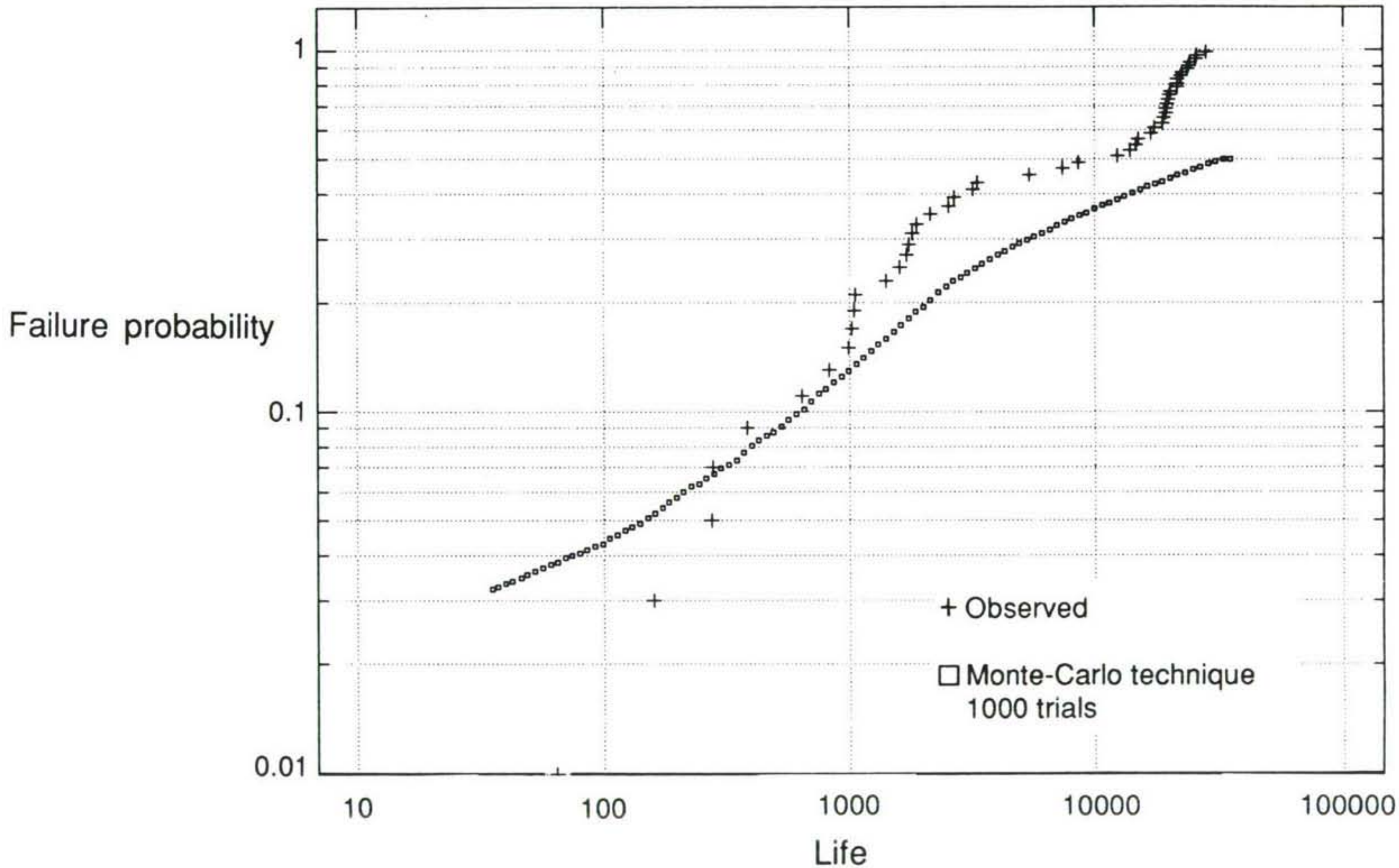
MAC5-49358-112089

Predicted Failure Distribution
Monte-Carlo Technique



MAC5-49359-112089

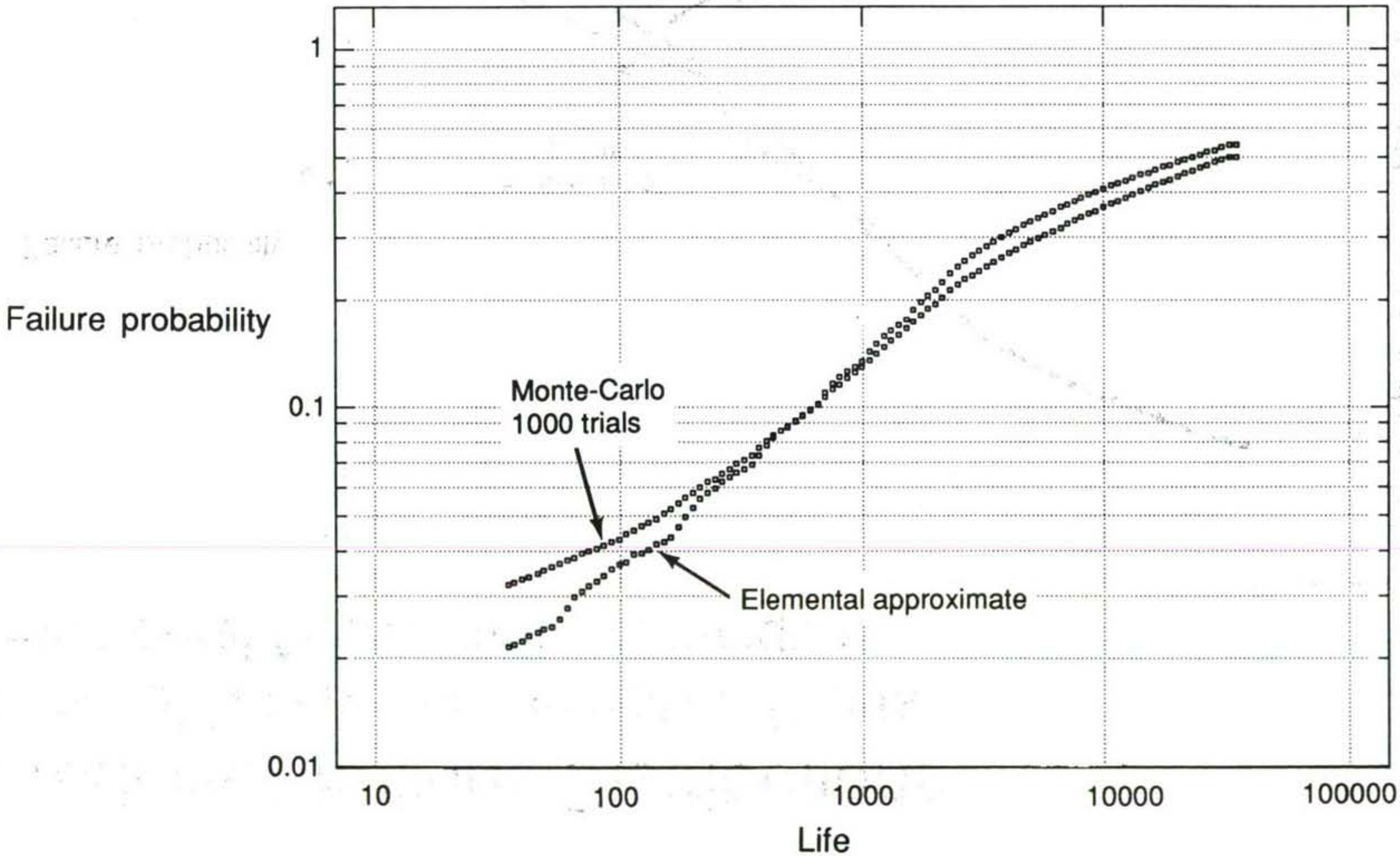
Comparison with Observed Failure Distribution



MAC5-49363-112089

Predicted Failure Distributions

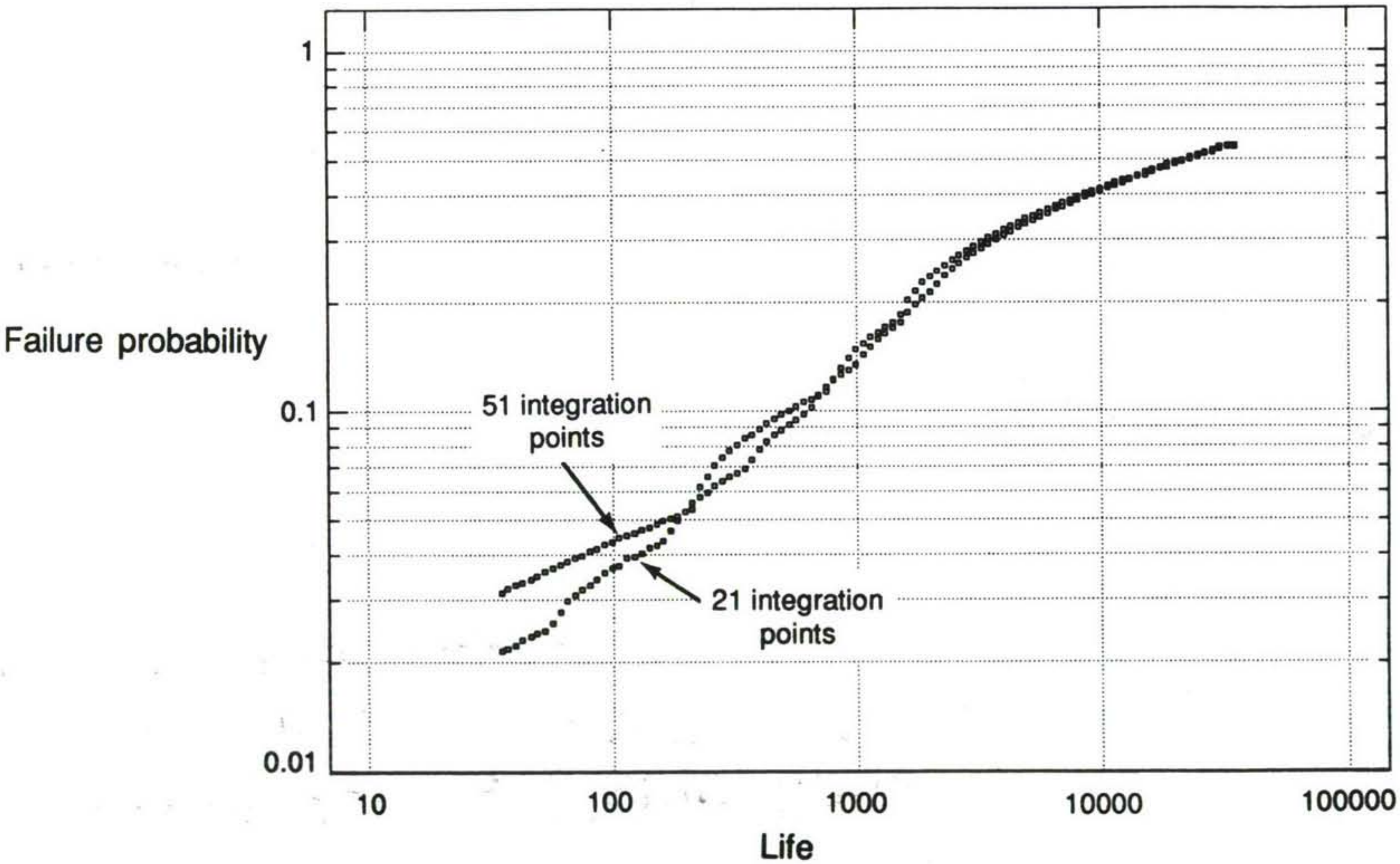
Comparison of Monte-Carlo Exact With Elemental Approximate Techniques



MAC5-49361-112089

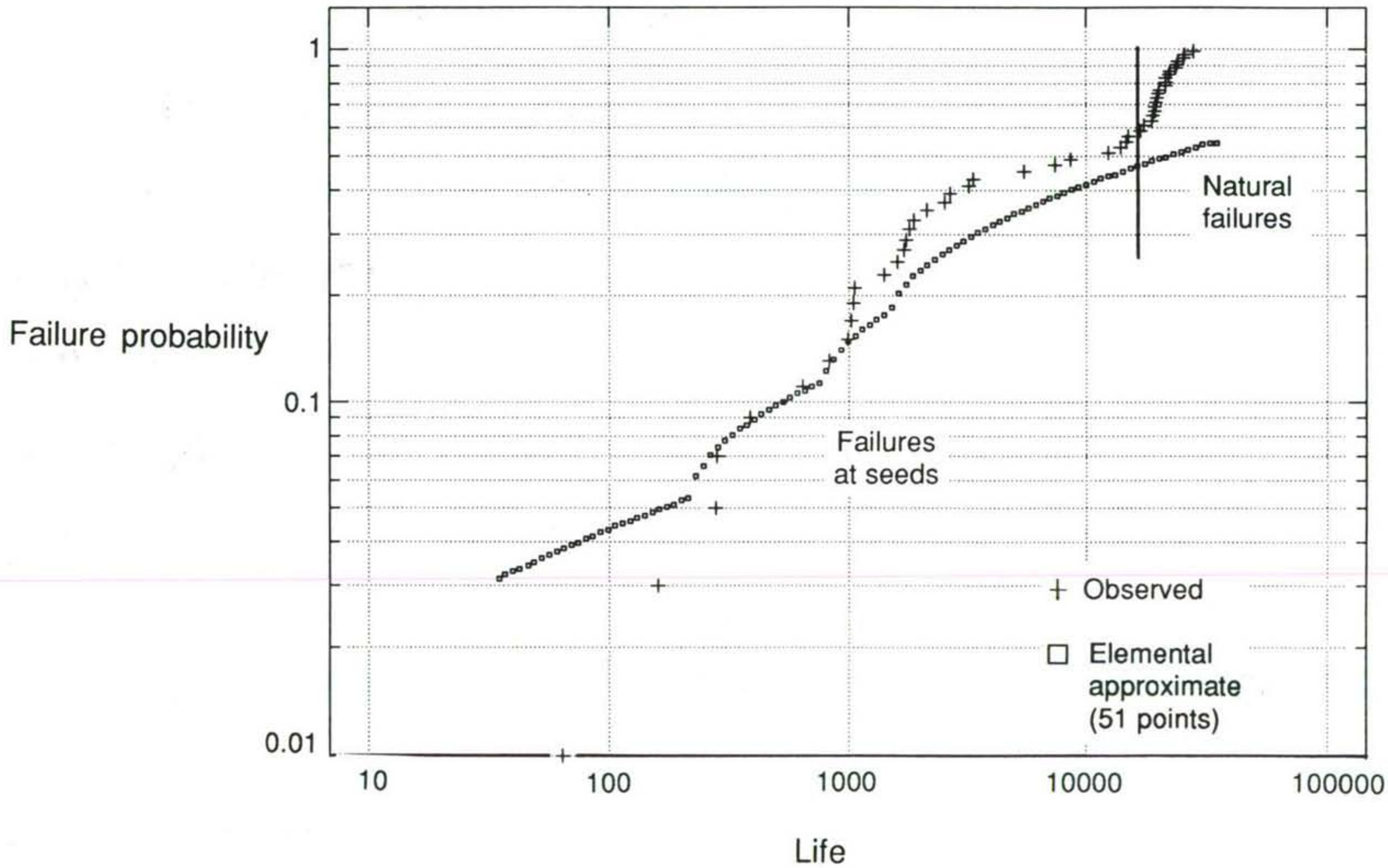
Predicted Failure Distributions

Effect of Number of Integration Points
(Elemental Approximate Technique)



MAC5-49362-112089

Comparison with Observed Failure Distribution



MAC5-49360-112089

Refinements

- Attention has been focused on definition of defect content and quantification of inspectability
- Recognition of stratification in the defect population . . .
- Are defects unavoidably part of the process?
- Are they
  - Totally random?
    - Random within time frames?
      - Random within lots?
        - Random within parts?
          - Random only locally?
- Answers significantly affect calculated risks

MAC5-49364-112089

- Assume that 1 disk yields 10 LCF specimens and that testing 10 disk cutups yields the following results:

Forging	A	B	C	D	E	F	G	H	I	J
# of inclusions	0	0	0	0	0	10	0	0	0	0

- Summary: 10 out of 100 specimens are found to have inclusions
- Dirtiness probability for a disk assuming a homogeneous distribution is calculated as

$$\boxed{1 - (1 - 0.1)^{10} = 0.65}$$

- More realistic value is 0.1
- Though this example is simplistic, it can be shown that assuming homogeneity is always conservative

MAC5-49365-112089

Refinements

- Many other factors contribute to probabilistic nature of risk:
  - Geometric deviations
  - Mission variability
  - Variability in material properties
- GEAE implementation of probabilistic fracture mechanics has to date:
  - Ignored geometric deviations within tolerances
  - Addressed limiting mission elements
  - Assumed utilization of average material properties
- These points reassessed in development plans

MAC5-49366-112089

Summary

- Evolution of the ENSIP approach must be encouraged to meet today's contigencies and future design needs
- ENSIP damage tolerance design reduces risk of failure
  - Risks need quantification
- Needs:
  - Better physical understanding of the inspection process and of damage mechanisms
  - Probabilistic treatment of probabilistic elements
- Significant payoffs possible:
  - Better assessment of existing engine capabilities
  - Greater confidence in designing for very high thrust-to-weight in the next generation

MAC5-49367-112089

CALIBRATING FIGHTER ENGINE ACCELERATED MISSION TESTING (AMT) THROUGH FIELD USAGE SURVEY

CAPT. T. FOWLER, USAF/ASD/YZEE
P. A. MALETTA, GE AIRCRAFT ENGINES
1989 UNITED STATES AIR FORCE
STRUCTURAL INTEGRITY PROGRAM CONFERENCE
SAN ANTONIO, 5-7 DECEMBER, 1989

OVERVIEW

- GENERAL INFORMATION AND DATA GATHERED
- SAMPLE OF SIGNIFICANT FINDINGS
- RESULTING ACCELERATED MISSION TEST (AMT) PROFILE ADJUSTMENT
- SUMMARY

BACKGROUND

- FIELD USAGE OF F16/F110 FLEET WAS CONDUCTED IN SEPTEMBER - NOVEMBER, 1988

- ASD REQUESTED THAT GE MODIFY AMT TO INCORPORATE FIELD RESULTS

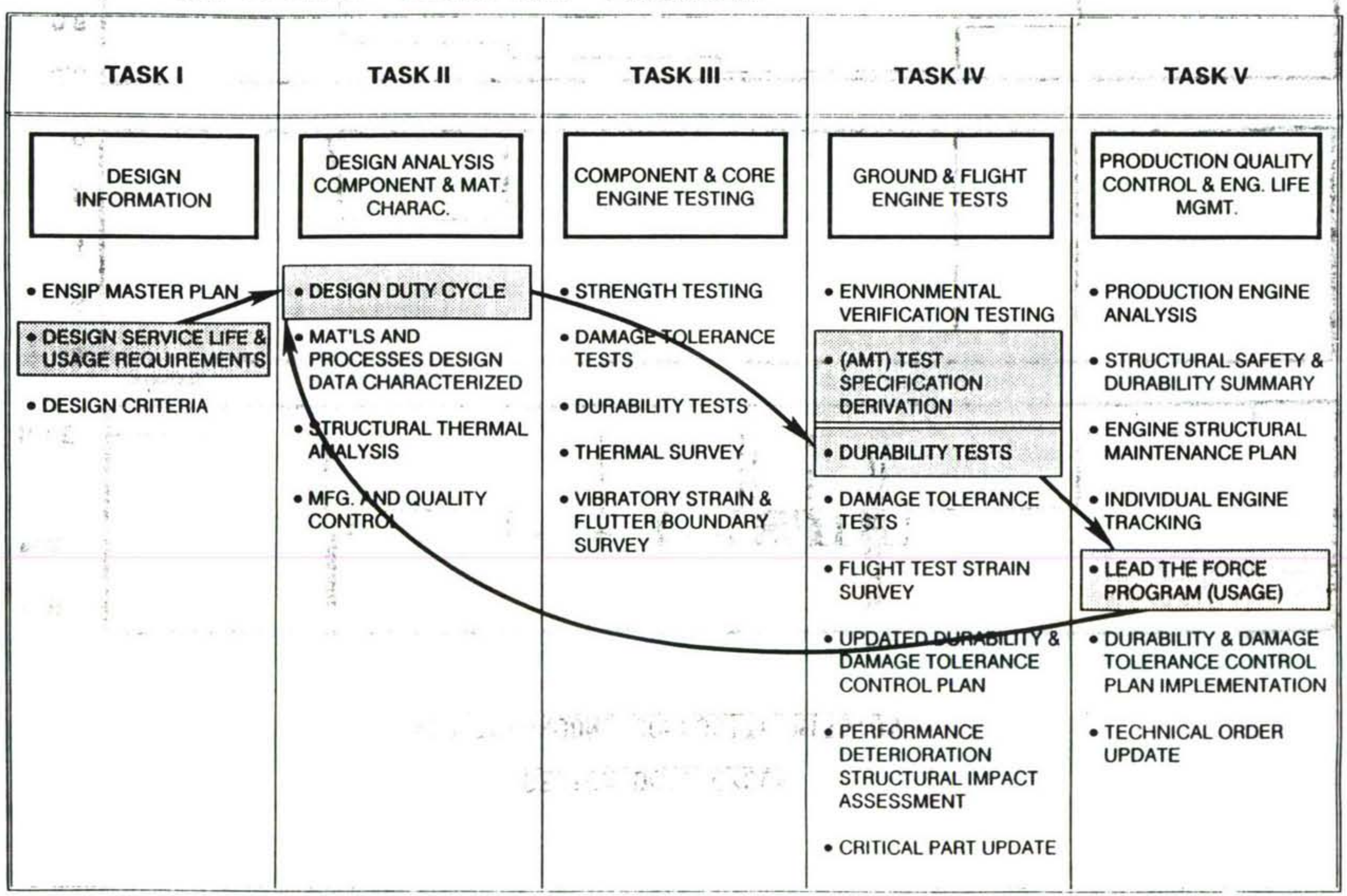
  - INTERIM CHANGE TO SUPPORT FIRST QUARTER, 1989 ENGINE TEST

- AMT IS BEING REDEFINED BASED ON DEVELOPMENT OF NEW FIELD-BASED DUTY CYCLE

BACKGROUND

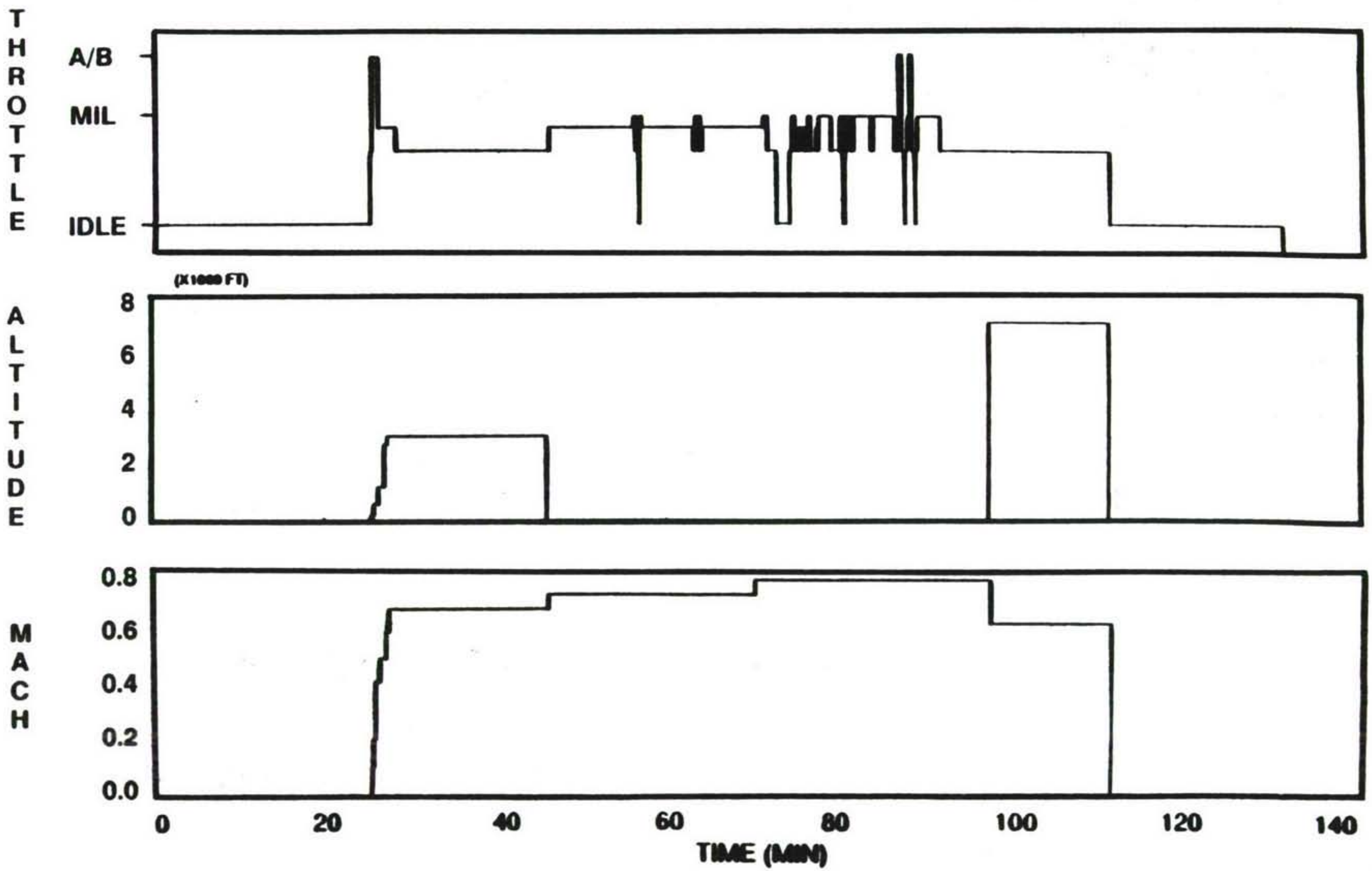
- FLEET USAGE SHOULD DRIVE DESIGN DUTY CYCLE (DDC) USED IN DESIGN ANALYSIS AND AMT CYCLE DESIGN

  - KEY TO ENSIP "CLOSED LOOP" PHILOSOPHY



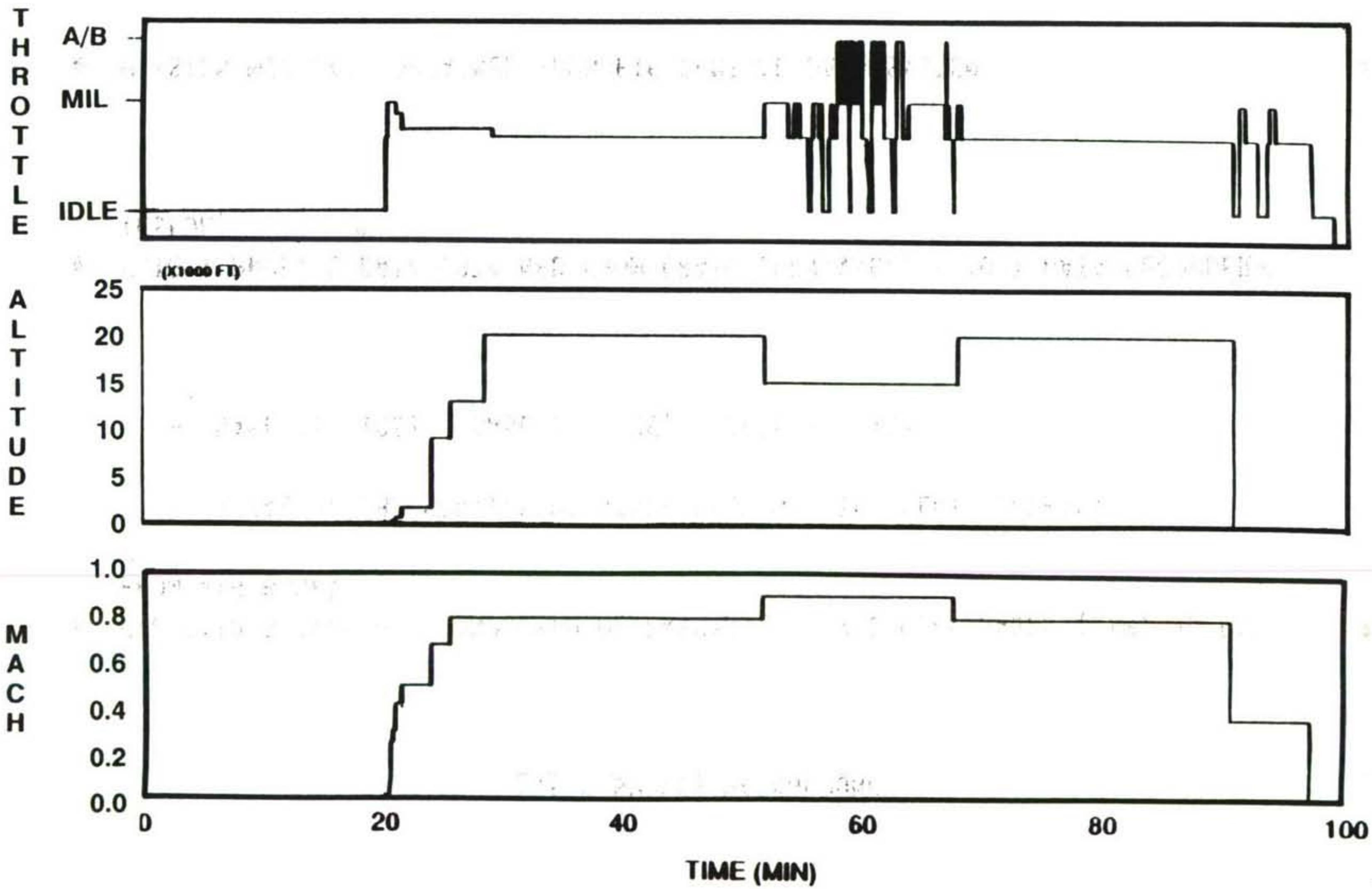
DESIGN DUTY CYCLE

AIR-TO-GROUND COMPOSITE MISSION



DESIGN DUTY CYCLE

AIR-TO-AIR COMPOSITE MISSION



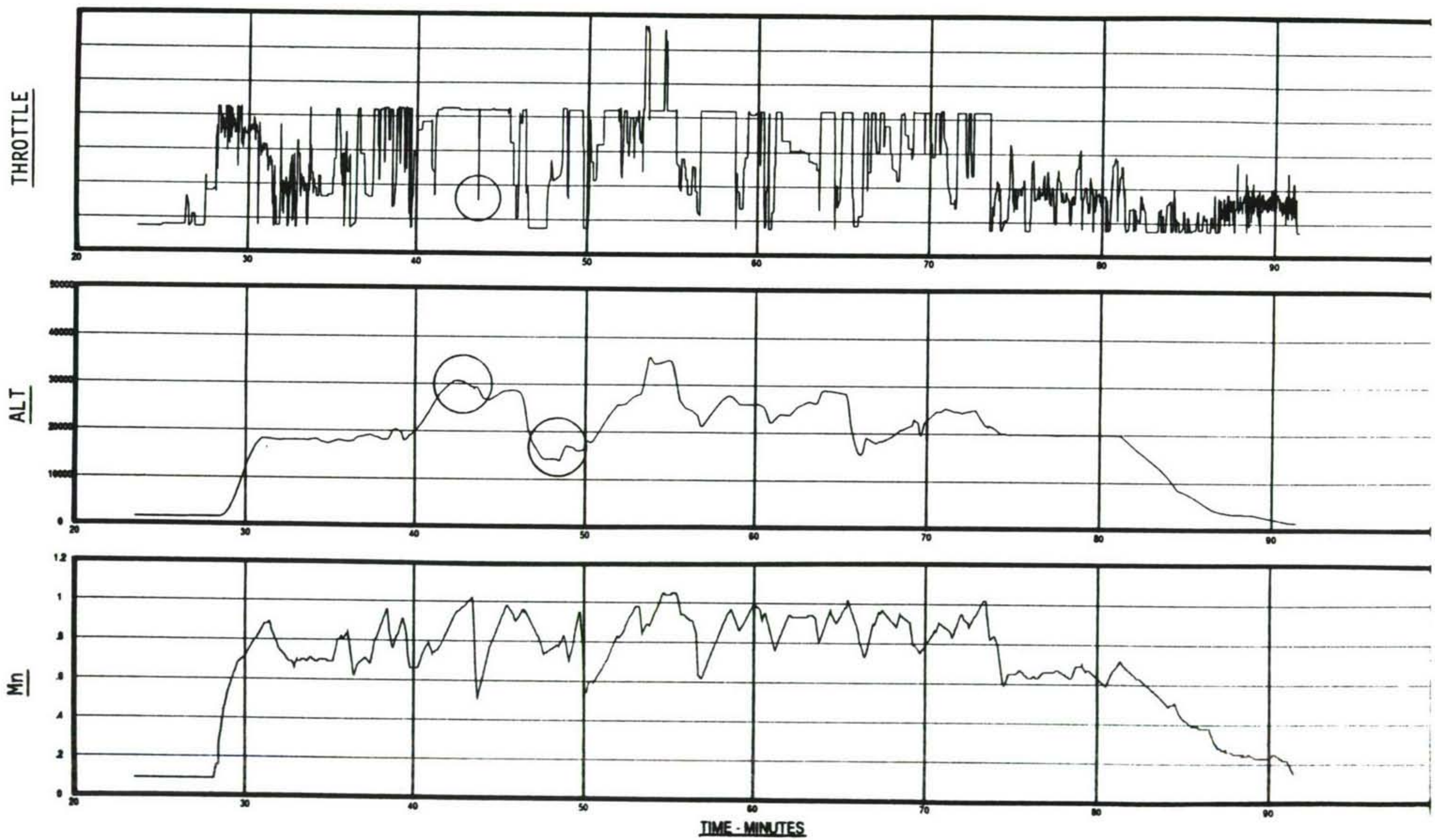
FIELD SURVEY BACKGROUND

- F16/F110 SURVEY WAS CONDUCTED OF PRIMARILY THREE USAF BASES (LIMITED DATA FROM 4TH BASE)
  - TOTAL OF 382 SORTIES OBTAINED FROM 95 DIFFERENT AIRCRAFT
  - BASE 1: 105; BASE 2: 126; BASE 3: 151
- FLIGHT PROFILE DATA OBTAINED FROM CRASH SURVIVABLE FLIGHT DATA RECORDER (CSFDR)
- MISSION MIX DATA OBTAINED FROM F16 CENTRAL DATA SYSTEM

BASIC MISSION PROFILES

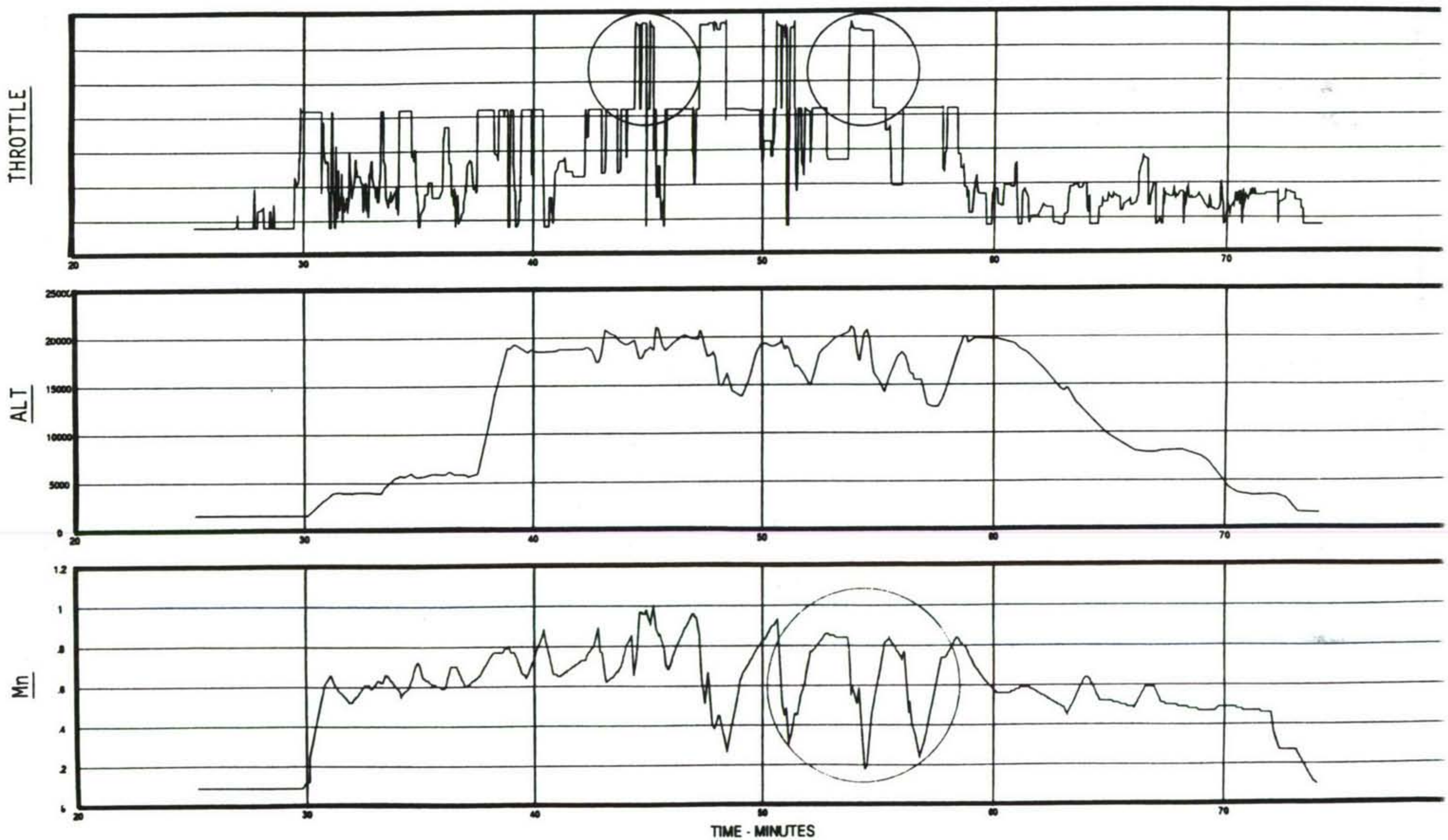
- AIR TO AIR
  - AIR COMBAT TACTICS/INTERCEPTS (ACT, DACT, ACT/INT, INT)
  - AIR COMBAT MANEUVERS (ACM, DACM, DACMBFM)
  - BASIC FIGHTER MANEUVERS (BFM, DBFM)
- AIR-TO-GROUND
  - AIR TO GROUND GUNNERY (AGG, RNGE, WD, NWD)
  - SURFACE ATTACK TACTICS (SAT, SAT/LL, NSAT)
  - WILD WEASEL (WW, WW/LL)
- FUNCTIONAL CHECK FLIGHTS (FCF, OCF)
- "OTHER"
  - CROSS-COUNTRY (XC)
  - INSTRUMENTS (INSTR)

TYPICAL FIELD
AIR COMBAT TACTICS

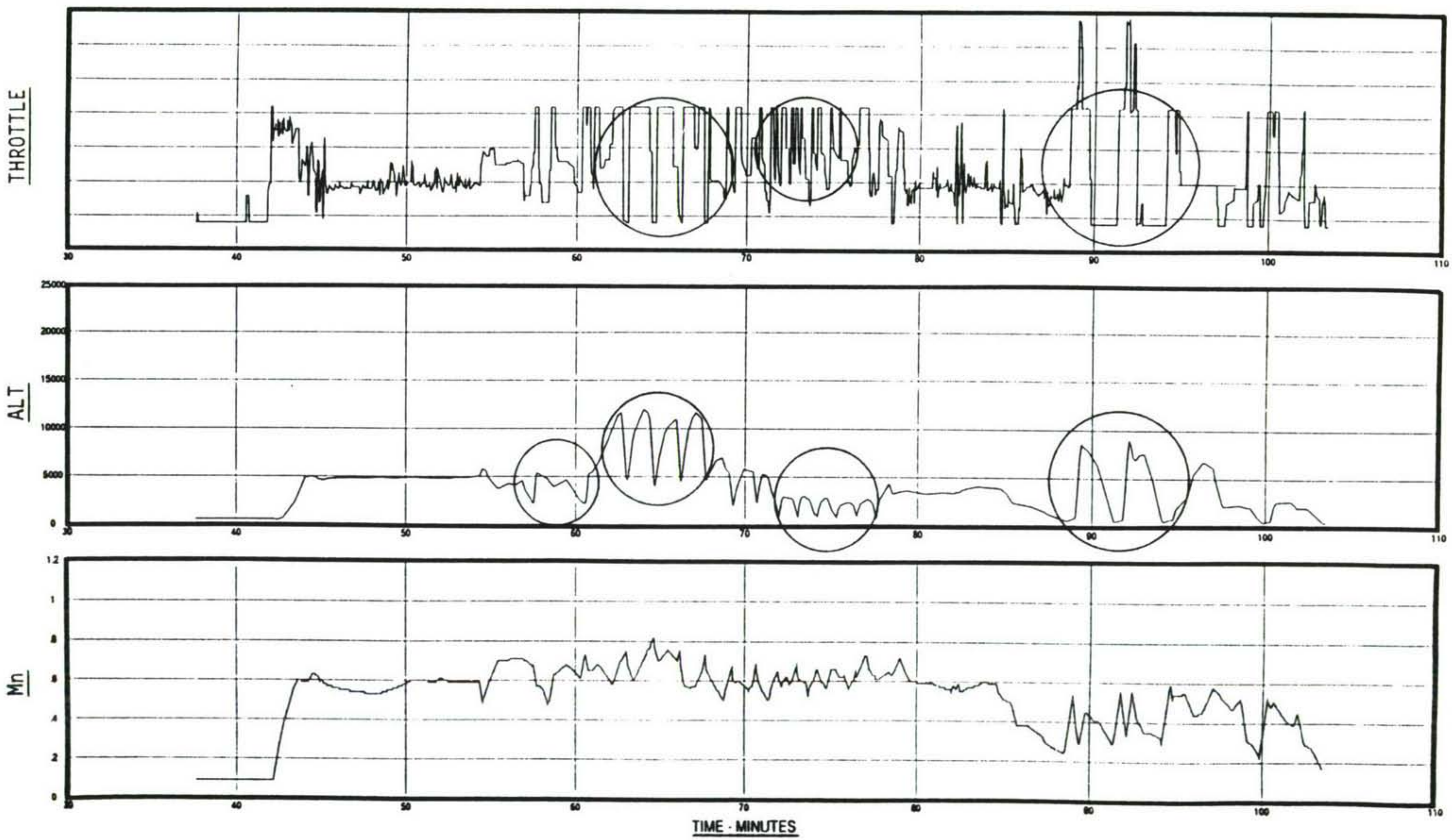


TYPICAL FIELD

BASIC FIGHTER MANEUVERS

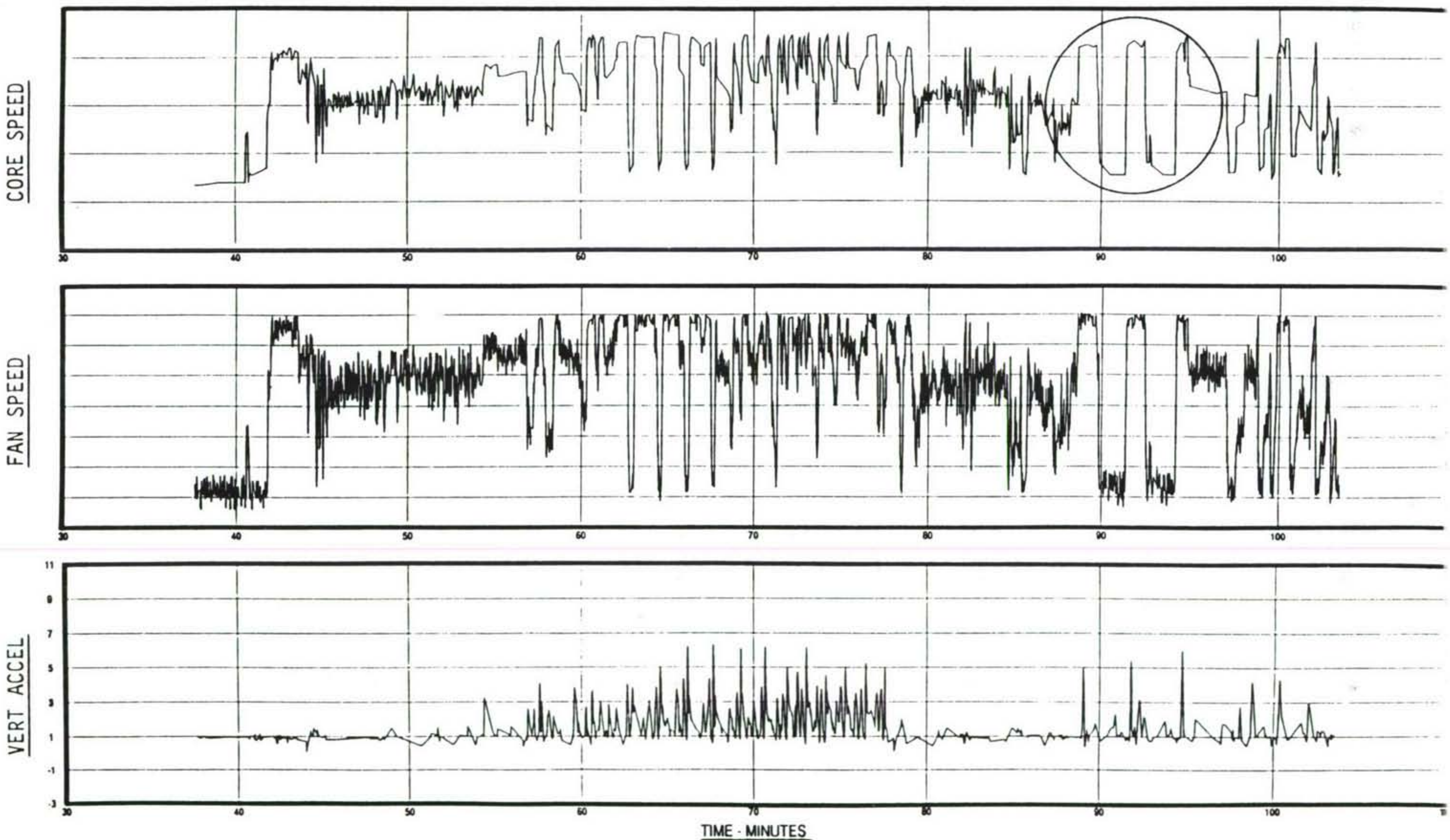


TYPICAL FIELD
AIR TO GROUND GUNNERY



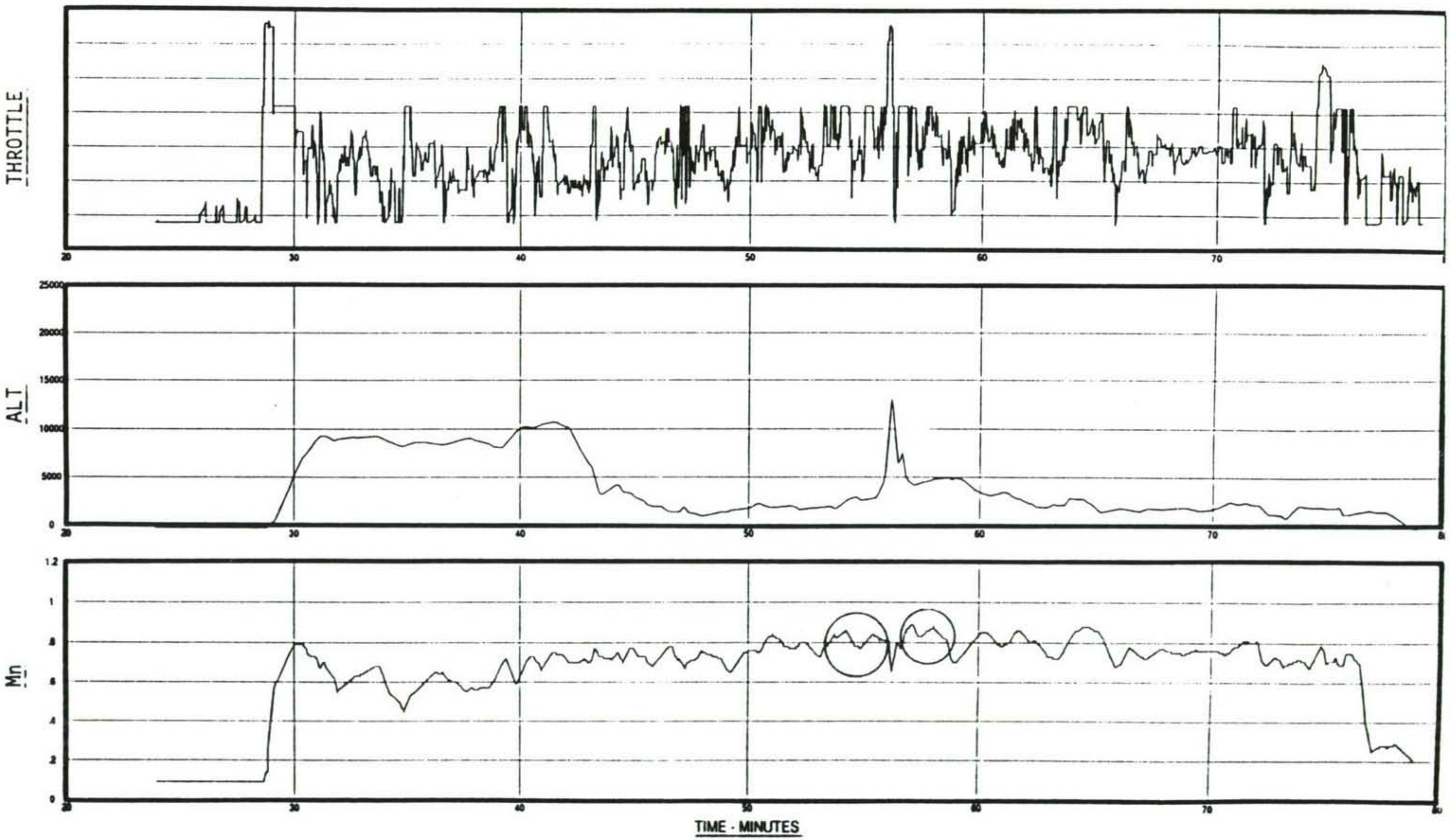
TYPICAL FIELD

AIR TO GROUND GUNNERY



CORE SPEED DURING SIMULATED FLAME-OUT (SFO)

TYPICAL FIELD
SURFACE ATTACK TACTICS



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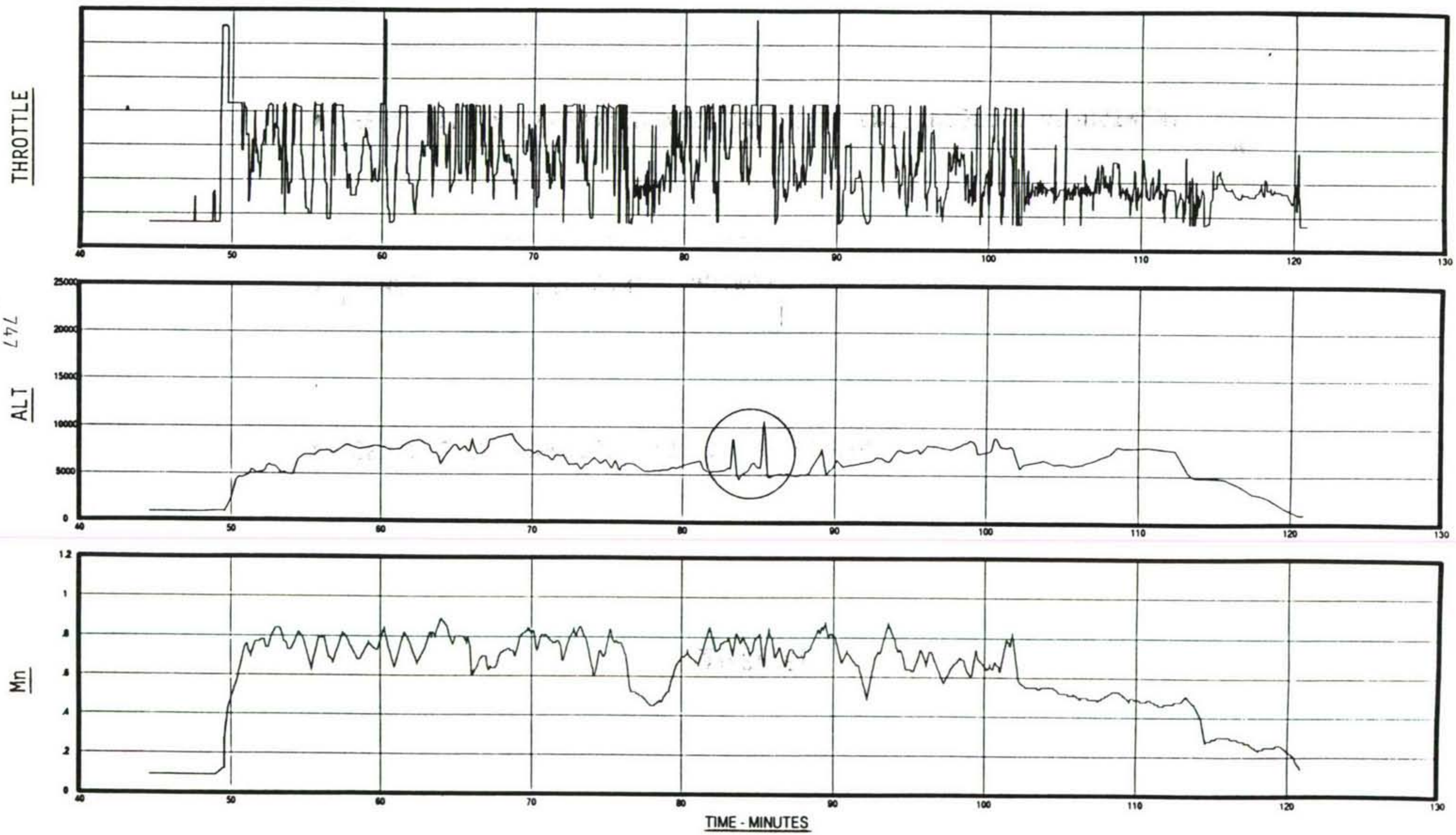
TYPICAL FIELD

WILD WEASEL



OVERVIEW

- GENERAL INFORMATION AND DATA GATHERED
- $\boxed{\text{SAMPLE OF SIGNIFICANT FINDINGS}}$
- RESULTING ACCELERATED MISSION TEST (AMT) PROFILE ADJUSTMENT
- SUMMARY

SIGNIFICANT FINDINGS

AIR-TO-GROUND RANGE PROFILES

- NUMBER OF ATG RANGE SORTIES AND NUMBER OF RANGE PASSES PER SORTIE VARIES BASE TO BASE BECAUSE OF RANGE AVAILABILITY AND PRESCRIBED MISSION
- MACH NUMBER OSCILLATES BETWEEN 0.6 AND 0.8
- ALTITUDE FLUCTUATES SEVERAL THOUSAND FEET WITH EACH RANGE PASS
- APPROXIMATELY ... ONE IDLE - MIL - IDLE CYCLE PER PASS
- AND/OR ... ONE PART POWER - MIL - PART POWER CYCLE PER PASS

SIGNIFICANT FINDINGS
USAGE VS. DESIGN DUTY CYCLE

- "COMPOSITE" DESIGN DUTY CYCLE LACKS FIDELITY
  - ACT/BFM/ACM = 1 ATA   SAT/AGG/(WW) = 1 ATG
- ALL DDC PARTIAL THROTTLE CYCLES LACK ADEQUATE EXCURSION
  - 3% CORE SPEED RANGE IN DDC VS. 15% CORE SPEED RANGE IN FIELD DATA
- ATG DDC LACKS TIME AT HIGHER MACH NUMBERS
  - 2-3 MIN .82/500', 1-2 MIN .85 - .90/500' (540-600KTS)
- SIMULATED FLAMEOUT MANEUVER NOT INCLUDED IN DDC
  - TURBINE WHEEL RIM STRESS DRIVER
- ACTUAL EFH ON ATA MISSIONS LOWER THAN DESIGN DUTY CYCLE
  - APPROX. 1.10 HRS. VS. 1.29 HRS.

AIR-TO-SURFACE MISSION DATA

BASE OR MISSION	EFH	EOT	CYCLE COUNTS			A/B LITES	TIME AT INT. & ABOVE (MINS)	RAM INLET TIME (%EFH)		
			FTC	PTC	TACS			17-22 PSI	20-25 PSI	25-40 PSI
BASE #1	1.53	2.26	10	10	2.5	2	12.0	13.8%	0.5%	0.0%
BASE #2	1.29	1.90	9	10	2.7	4	10.0	11.6%	1.0%	0.0%
BASE #3	1.27	1.82	8	7	2.6	2	7.3	16.1%	1.7%	0.0%
FIELD AVERAGE	1.36	1.99	9	9	2.6	3	9.8	13.8%	1.1%	0.0%
DESIGN DUTY	1.44	2.19	5	0	1.6	3	13.3	80.5%	0.0%	0.0%
CURRENT AMT	0.38	.54	5	0	1.6	3	13.3	0.0%	0.0%	0.0%

- HOT TIME SLIGHTLY LOWER THAN DDC/AMT
- FIELD BASED COUNTS GREATER THAN DDC/AMT
- DDC/AMT PARTIAL CYCLES NEED TO GO LOWER IN POWER
- NO RAM PRESSURE INLET IN AMT

AIR-TO-AIR MISSION DATA

BASE OR MISSION	EFH	EOT	CYCLE COUNTS			A/B LITES	TIME AT INT. & ABOVE (MINS)	RAM INLET TIME (%EFH)		
			FTC	PTC	TACS			17-22 PSI	20-25 PSI	25-40 PSI
BASE #1	1.08	1.71	9	5	3.1	3	10.9	1.2%	0.1%	0.0%
BASE #2	1.27	1.79	6	6	2.1	8	17.3	1.6%	0.0%	0.0%
BASE #3	1.06	1.57	7	5	2.7	5	11.2	2.0%	0.2%	0.1%
FIELD AVERAGE	1.14	1.69	7	5	2.5	5	13.1	1.6%	0.1%	0.0%
DESIGN DUTY	1.29	2.02	8	0	2.3	8	12.2	0.0%	0.0%	0.0%
CURRENT AMT	0.35	.52	8	0	2.3	8	12.2	0.0%	0.0%	0.0%

- FIELD BASED COUNTS CONSISTENT WITH DDC/AMT
- FIELD HOT TIME COMPARABLE TO DDC/AMT
- FIELD A/B LITES LESS THAN DDC/AMT
- **DDC/AMT PARTIAL CYCLES NEED TO GO LOWER IN POWER**

<u>OVERVIEW</u>

- GENERAL INFORMATION AND DATA GATHERED
- SAMPLE OF SIGNIFICANT FINDINGS
- $\boxed{\text{RESULTING ACCELERATED MISSION TEST (AMT) PROFILE ADJUSTMENT}}$
- SUMMARY

RESULTING AMT PROFILE ADJUSTMENT

- AMT ADJUSTMENT IS REQUIRED TO LEAD FLEET AND DELIVER IMPROVED RELIABILITY

RECOMMENDED CHANGES:

- AIR-TO-SURFACE AMT PROFILE (ATS AMT)
  - INCORPORATE 9 FTC's & 9 PTC's vs. 5 FTC's & 10 PTC's
  - USE LOWER ROTOR SPEED POINT FOR PTC's
  - INCLUDE RAM INLET CONDITIONS
- AIR-TO-AIR AMT PROFILE (ACM AMT)
  - USE LOWER ROTOR SPEED POINT FOR PTC'S

AIR-TO-SURFACE AMT ADJUSTMENTS

- WANTED TO RUN ATS CYCLES WITH RAM INLET TO OBTAIN HIGH P3 LEVELS:

  1/3 AT T1 = 65°F
  1/3 AT T1 = 97°F
  1/3 AT T1 = 130°F

  NOT ATTAINABLE DUE TO FACILITY LIMITIATIONS

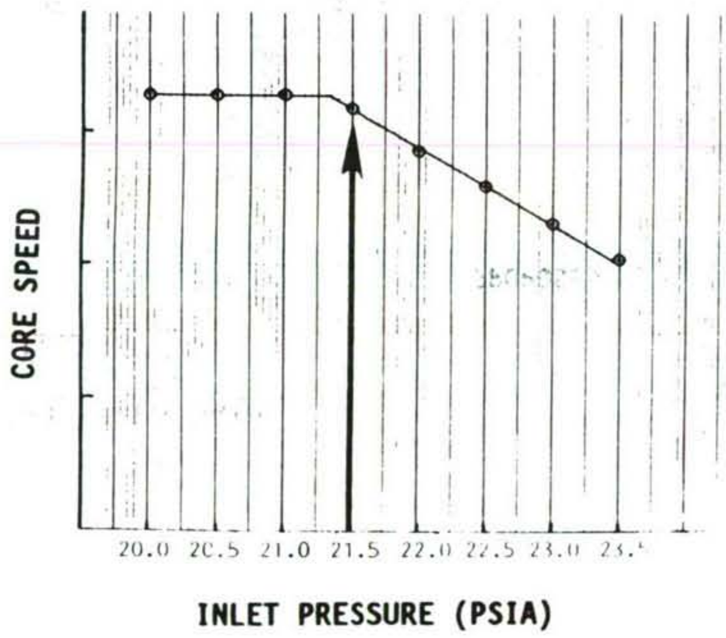
- TEST FACILITIES CAPABILITY CAN PROVIDE P1 = 23.5 PSIA (MAX) AT 130°F (MIN) FOR CONSISTENT YEAR ROUND TESTING

- AT HIGH INLET PRESSURES, CORE SPEED DROPS DUE TO P3 LIMIT BEING REACHED

- SELECTED T1 = 130°F AND P1 = 21.5 PSIA FOR ATS RAM CONDITIONS

  - RAISES P3 ≈85 PSIA OVER SEA LEVEL (70°F)

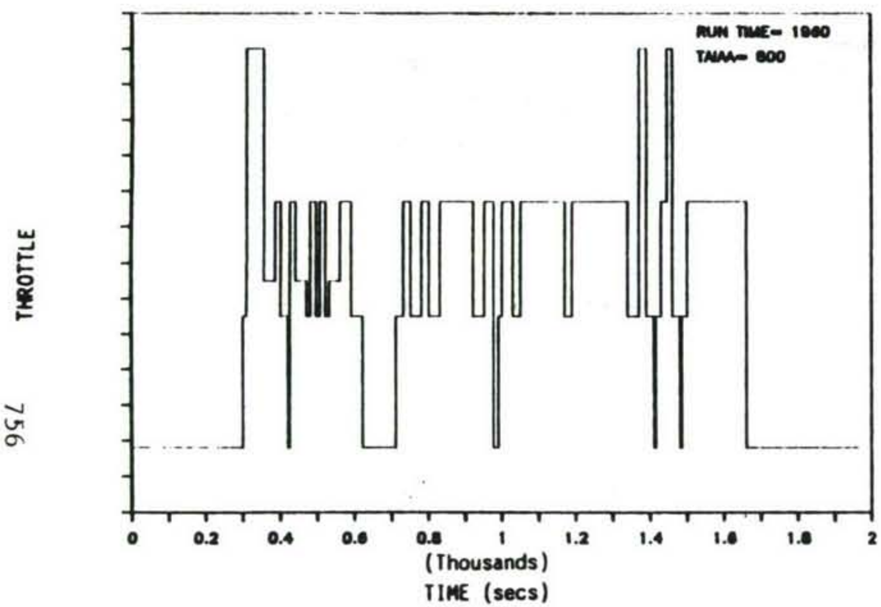
  - AT MIL POWER AND ABOVE, ON ENGINE P3 LIMIT



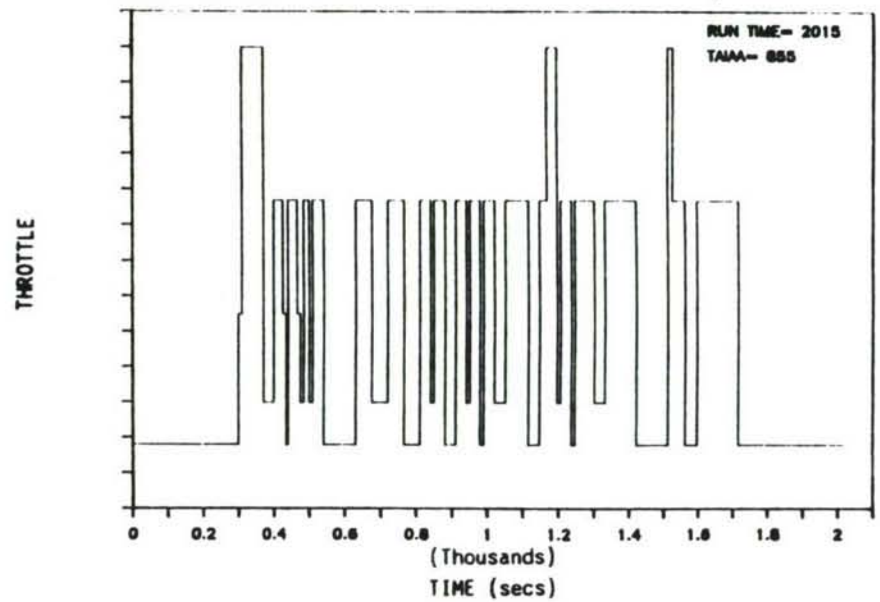
ADJUSTMENTS FROM FIELD DATA

ATS CYCLE

CURRENT



PROPOSED



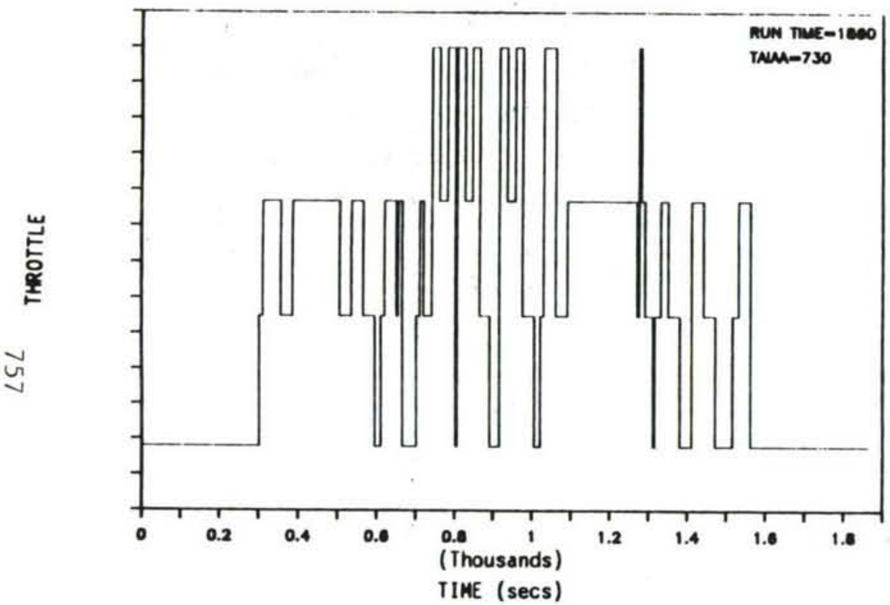
- PTC LOWER POWER POINTS REDUCED TO REPRESENT FIELD DATA
- DWELL TIMES AT IDLE ADJUSTED TO SIMULATE ACTUAL ATS MISSIONS
- ENTIRE CYCLE RUN AT RAM INLET CONDITIONS
- RAM PRESSURE & TEMPERATURE LEVELS SET TO MAXIMIZE SEVERITY WITHIN CONSTRAINTS OF FACILITY'S CAPABILITIES:

P1 = 21.5 PSIA AND T1 = 130°F

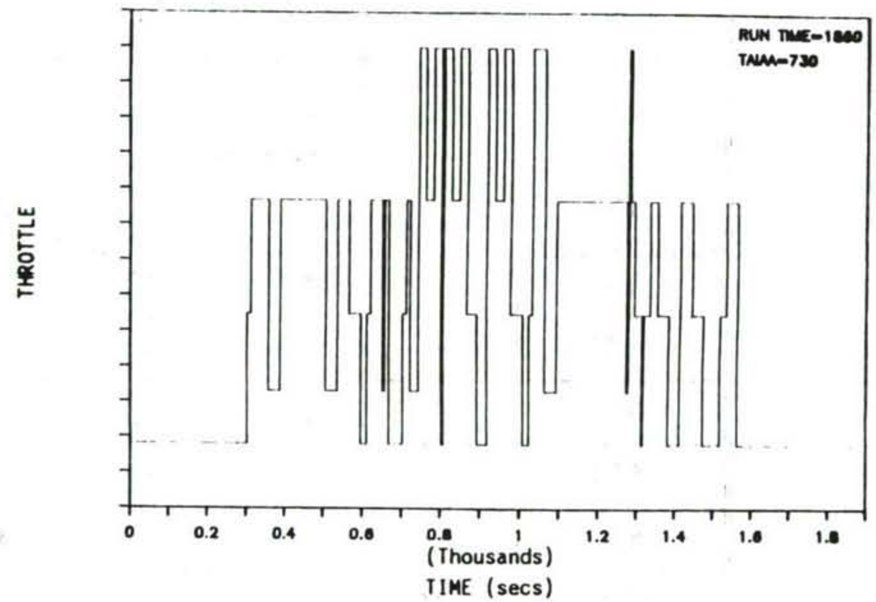
ADJUSTMENTS FROM FIELD DATA

ACM CYCLE

CURRENT



PROPOSED



- PTC LOWER POWER POINTS REDUCED
- NO OTHER CHANGES TO AIR-TO-AIR CYCLE

PROPOSED AMT MISSION MIX

- REVIEWED MISSION MIX OF ATS AND ACM FOR CURRENT FIELD DATA
  - ATS % IS HIGHER THAN CURRENT TESTING
  - GROUP ALL "OTHER" MISSIONS INTO ACM GROUP

	BASE	NO. MISSIONS	ATS	ACM	FCF	GTC	OTHER*
	BASE 1	(4,384)	48%	45%	1%	-	6%
	BASE 2	(4,462)	61%	26%	1%	-	12%
	BASE 3	(5,168)	59%	35%	2%	-	4%
	BASE 4	(3,220)	53%	36%	1%	-	10%
	BASE 5	(4,489)	63%	28%	1%	-	8%
FIELD MIX	TOTAL MISSIONS =	(21,723)	57%	34%	1%	-	8%
AMT MIX	CURRENT		51%	42%	4%	3%	-
	PROPOSED		57%	42%	1%	-	-

*OTHER = CROSS COUNTRY, INSTRUMENTS, FERRY, ETC.

- PROPOSED AMT MISSION MIX
  - KEEPS ACM PERCENTAGE THE SAME
  - INCREASES ATS TO 57% TO REFLECT FIELD USAGE
  - REDUCES FCF BY 3% TO REFLECT FIELD USAGE
  - ELIMINATES GTC CYCLE

AMT COMPARISON (ONE BLOCK)

	CURRENT	PROPOSED
MISSION MIX PERCENT (CYCLES) ATS	393 (51%)	437 (57%)
ACM	325 (42%)	325 (42%)
FCF	27 (4%)	8 (1%)
GTC	25 (3%)	---
INLET PRESSURE (PSIA)	14.7	21.5 (ATS ONLY)
TOTAL ABTIME (AT RAM) HRS	27.3 (0)	27.3* (12.7 HRS - 46%)
TOTAL TAIAA (AT RAM) HRS	171.2 (0)	171.2* (103.8 HRS - 61%)
TOTAL TIME (AT RAM) HRS	411.1 (0)	417.8* (244.6** HRS - 59%)
TACS ACCUMULATED	2153	2565* (+412 TACS)

* TO KEEP ABTIME AND TAIAA EQUAL TO CURRENT AMT, 55 SEC HAD TO BE ADDED TO ATS CYCLE (55 SEC. OF TAIAA AND 24 SEC OF ABTIME). THIS WAS DUE TO RE-PROPORTIONING THE ATS, ACM, FCF AND GTC % MIX.

** INCLUDES 10 MINS/CYCLE IDLE FOR STABILIZATION BEFORE SHUTDOWN AND AFTER START-UP. WITHOUT IDLE, HAVE 172 HRS AT RAM WITH THROTTLE MOVEMENTS.

OVERVIEW

- GENERAL INFORMATION AND DATA GATHERED
- SAMPLE OF SIGNIFICANT FINDINGS
- RESULTING ACCELERATED MISSION TEST (AMT) PROFILE ADJUSTMENT
- $\boxed{\text{SUMMARY}}$

SUMMARY

- FIELD MISSIONS APPEAR MORE SEVERE THAN DESIGN DUTY CYCLE ... BUT FOR GOOD REASON:
  - ADDITIONAL WEAPON SYSTEM CAPABILITY
- PROPOSED AMT IS MORE REPRESENTATIVE OF THE ACTUAL FIELD USAGE
  - SIMULATES ADDITIONAL THROTTLE MOVEMENTS
  - ATS SIMULATES 500 FT, MACH .78
  - INCLUDES PRESSURE CYCLING TO MAX P_3 LIMIT
- ENGINE HAS ACCUMULATED 550 CYCLES IN FACTORY TEST (207 AT RAM INLET CONDITIONS)
- NEW FIELD BASED DUTY CYCLE HAS BEEN DEFINED AND COMPONENT LIVES ARE BEING CALCULATED
- NEW AMT IS BEING DEVELOPED FROM THESE MISSION PROFILES

ENSIP

AEROMECHANICAL TESTING

- WHERE DO WE GO FROM HERE -

"?"

OTHA DAVENPORT
WILLIAM D. COWIE
ASD/YZE

WORDS TO BE USED IN THE PRESENTATION

"ENSIP, AEROMECHANICAL TESTING - WHERE DO WE GO FROM HERE - ?"

Otha B. Davenport ASD/YZE and William D. Cowie, ASD/YZEE

TITLE PAGE	Self explanatory.
CHART 1	(Overview) This chart gives an overview of the presentation. I will first present the status of ENSIP and some benefits we have seen. Next, I will discuss our experience in handling aeromechanical problems and will then present a summary of our current design and verification requirements. Otha will present the latest thinking on potential change to aeromechanical requirements and discuss some emerging technologies that may be of benefit to the aeromechanical area.
CHART 2	This chart shows ENSIP designed engines and those that were assessed for damage tolerance. Aero-mechanical requirements have always existed in the general Mil Specs for engines (MIL-5007) and applied through the 1960s and 1970s. In 1978, ENSIP Aeromechanical Requirements were increased somewhat over MIL-5007D to the current levels which I will cover later.
CHART 3	This chart shows some of the ENSIP payoffs based on fighter engine experience. The growth rate of hot section life is shown over the years from 1946 through 1989. Our AMT tests and field experience have demonstrated today's fighter engine hot section life to be in excess of 2000 flight hours. Cold section life is at least twice the hot section life (4000 flt hours).
CHART 4	Self explanatory.
CHART 5	Self explanatory.
CHART 6	Rotor speed transients - This chart shows an example of a fighter mission rotor speed transient which is in sharp contrast to commercial engine usage. Rotor speed transients on fighter engines are numerous and pass through component critical vibratory modes in both accels and decels throughout the operating range. These speed changes induce wear, thermal

operation which can induce, through modal crossing, short burst of high vibratory stress that when added together often exceeds material 10^7 cyclic vibratory crack initiation limits.

CHART 7 This chart shows we face problems calculating steady state stresses in blade attachment regions because of uncertainties in friction effects. In order to reduce titanium fretting, a coating with a lubricant is usually applied. This can affect joint friction. In the analysis, assumed values of friction can change stress level as well as critical locations. This uncertainty in mean stress can affect HCF predictions when using a Goodman Diagram.

CHART 8 This rotor speed vs frequency chart, usually called a Campbell Diagram, shows the interaction of possible HCF driver stimulating blade or vane natural frequencies in the operating range. It also shows the problem of part-to-part frequency variation. This variation of part frequency must be taken into account if one is to design a vibratory critical mode out of the operating range. ENSIP requires a 10% margin considering minimum frequency variations. Because of a low part count for new engine design, it is difficult to establish production part frequency variation. In addition, analysis of critical part frequencies up front are not always accurate and part frequency characterization must wait for correlation with actual part bench testing. Often, redesign of the part or driver is needed.

CHART 9 This chart shows the trend toward modal compression. Higher order modes can actually overlap due to closeness of the modal frequencies and part-to-part variations.

CHART 10 This chart from the F110/F16 usage data shows drivers that can affect HCF. It shows low altitude and hi speed flight conditions which produce pressures to 40 atmospheres or more and can increase overall vibratory stresses. In general the engine works harder. In addition to the higher pressures, notice that rapid throttle movements are occurring and crossing possible critical vibratory modes in the operating range. In general, commercial engines do not see this kind of usage.

CHART 11 This chart shows that Ti material endurance limits are still going down at 10^7 cycles. Tests have shown that a reduction of roughly 10% can be expected at 10^9 cycles.

CHART 12 This chart shows a typical 10^7 Goodman Diagram. ENSIP requires vibratory stresses no higher than 40% of the endurance limit at the appropriate mean stress. However, even with the low vibratory requirement, a standard Goodman may not be adequate for fighter engine design. For example, 1000 hours of fighter usage would cycle the steady stress from 0-max-0 about 1,000 times, plus 10,000 idle-max-idle steady state cycles. This interaction of a cycling steady stress or low cycle fatigue on a vibratory stress is not accounted for in the standard Goodman Diagram. One would think that lower allowables would occur particularly at the higher mean stress.

CHART 13 We have talked about the first six variables and there are more. Some of these are listed! Temperature affects modal frequency and must be accurately determined throughout the engine for each flight condition. Thermal stress can affect mean stress and must be taken into account in the analysis. Instabilities such as flutter must not be in the operating range. Flutter boundaries must be firmly established for nominal and off schedule vane operation. Ambient temperature effects must be investigated since rotor speed match up, density, pressures, internal temperatures and control system inputs are affected. Damping variabilities directly affect vibrational amplitudes and stresses and must be understood. Gage reliability after the first 10 hours of running is about 40%. Telemetry is sometimes plagued by electrical noise and is limited to 250°F temperatures. Since telemetry is usually cooled by oil which normally runs 450°F, refrigeration of the oil to as low as -40°F is required to protect telemetry system from overtemp. If refrigeration system fails, heat soak back can jeopardize telemetry system operation. TIP rubs - can stimulate blade vibration and cause blade HCF cracking.

CHART 14 Self explanatory.

CHART 15 This chart shows the verification methods by which ENSIP aeromechanical requirements are met.

CHART 16 Self explanatory.
Otha will now talk on potential changes that we are thinking about and emerging technologies which can have beneficial effects on HCF problems.

NO CHART Potential Changes - Thank you Bill for describing the problems that we are experiencing in the area of aeromechanics. It is clear that this is now and will be an area that we must improve both the design and testing methods if we are to realize

the full potential from ENSIP.

Now I would like to discuss some of the changes that are being considered to improve our ability to detect and correct aeromechanical problems in development instead of in operational use.

CHART 17 & 18

HCF Cycle - First, we are considering conducting about half of the HCF cycles at elevated inlet pressures to more accurately simulate the dynamic pressure that may be the forcing function for component responses.

Potential Changes - Second, due to relatively wide variation in installed component responses and aerodynamic and structural damping, it has been shown that instrumenting several components and treating the data as a population sample instead of deterministic point data, can project a a higher risk segment of the population and lead to early identification of potential problems.

Third, it is incumbent on all of us to do a better job in predicting the expected results from the aeromechanical tests instead of running a test to see what happens. Modal analysis is a currently available tool to assist in this and being used to some degree in problem solving. We need to assure that more application is made in a predictive mode and then used to learn why some responses don't match the predictions.

The fourth potential change and one that I feel is quite significant, is the selection and characterization of the test hardware. Current practice is to try to select components that represent a variation of response characteristics based on the range of individual component responses. This is a kind of shotgun approach but may be valid depending on predictive analysis. Another approach would be to predict the critical responses through analysis and rig testing and configure the test article to verify these predictions. Clearly we must maximize the value of the information coming from these very expensive and complex tests. Further consideration of the frequency content of the inlet airflow and the configuration of the test inlet distortion needs to be evaluated in selecting the test setup.

CHART 19

Potential Changes - Other areas where I believe we can improve the quality and quantity of useful information from the aeromechanical

4

tests are:

Spin calibrating the instrumentation. This serves two purposes. First, the confidence in correlating the experimental data with analysis is improved and secondly, it is a darwinian approach to assuring the survival of critical instrumentation in that the weak sisters are eliminated during calibration and can be replaced prior to final assembly.

~~CHART 20~~ Another phenomena we are now observing is that the titanium-aluminides do not appear to possess a true endurance limit and may degrade as much as 10-20% between 10E7 and 10E9 cycles. Further the effects of cyclic steady stresses may have an impact similarly to ground-air-ground cycles in airframes.

Program planners should where possible plan for a second aeromechanical test to explore and characterize the surprises found on the first test after the data has been analyzed. Due to program schedules, it may not be possible to get critical information if the test is not pre-planned. Of course if the first test is well behaved, the second test may be cancelled.

Finally, the technology of the turbomachine design is changing ways that make the aeromechanical work more challenging. Just a few of the changes are noted here.

CHART 20 Emerging Technologies - All is not gloom and doom in the aeromechanics world however. One of the exciting things about technology is the tools somehow seem ôo keep pace with our understanding of the problems. Some of the high potential technologies that will serve to improve our understanding and ability to develop solutions and test methodologies for aeromechanics are:

Computational fluid dynamics - Probably the fastest progressing technology in this area will provide the ability to predict complex aerodynamic and structural interactions and provide estimates of aerodynamic damping which with todays methods is very elusive. The mapping and solutions from the time domain to the frequency domain has an excellent potential to provide solutions within practical computational constraints.

Also the advances in optical measurement systems may provide tools to measure stresses both synchronous and asynchronous without direct

contact on a component. Laser doppler velocimeters may offer ways to directly measure parameters such as local flow direction and velocities that will validate the CFD models and give further insight into the complex flow fields in aerodynamic cascades.

CHART 21 Emerging Technologies - The current generation of telemetry for whole engine tests suffer from many limitations both in capability and capacity. Advances in cooling methods and microelectronics now offer the promise of telemetry systems that will approach core engine tests in data capacity and with better overall data quality.

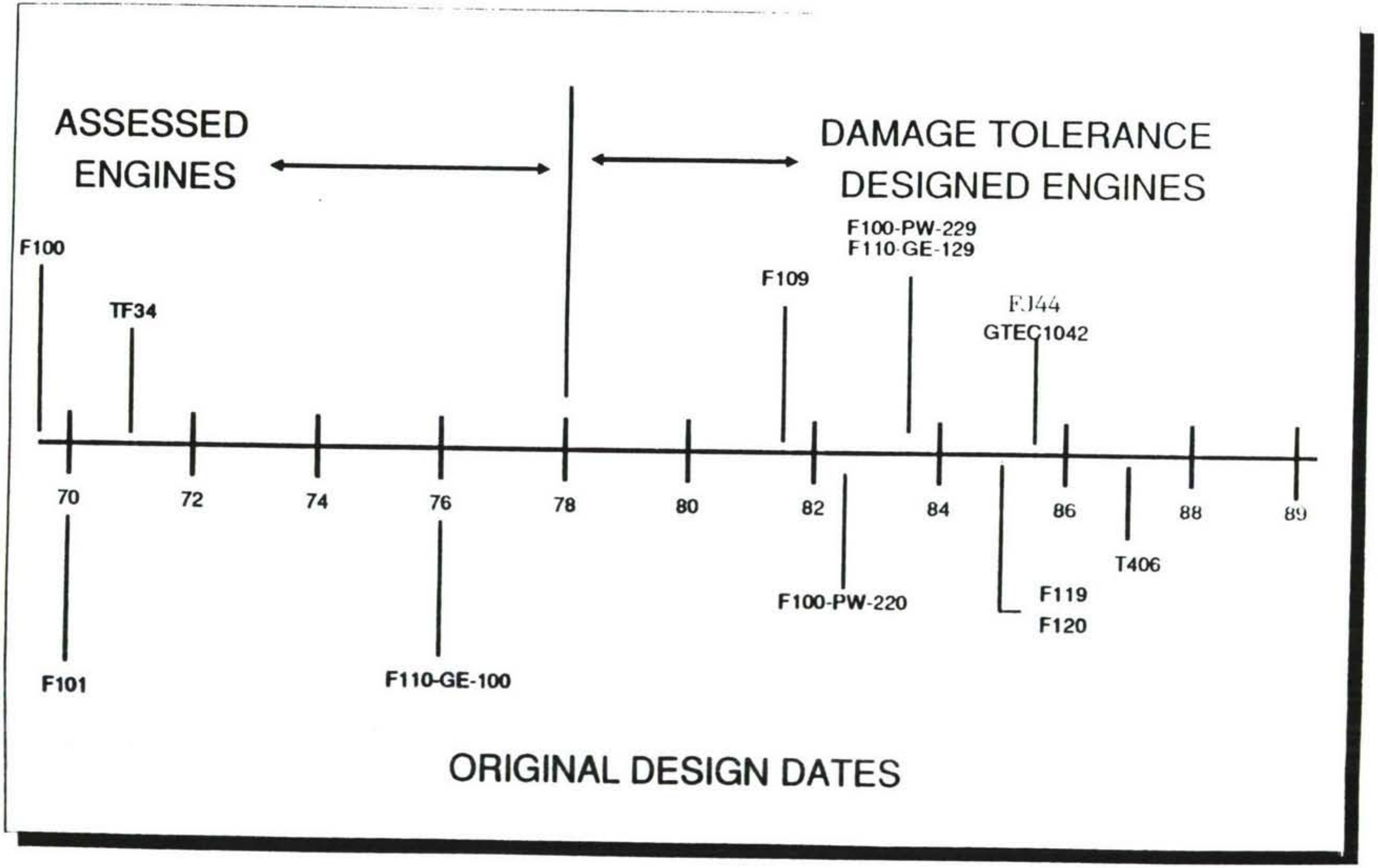
One final area that I want to mention relates to CFD and is the application of spectral methods that have been very successfully used in other areas. Some of the benefits are noted and although the analyses are very complex, the benefits could be substantial.

CHART 22 Summary - In summary, the Air Force propulsion community believes that aerodynamics represent an area that we can significantly improve the current practices and we are taking steps both short term and long term to develop better test methods and better engines as a result.

OUTLINE

- EXPERIENCE
- CURRENT REQUIREMENTS
- POTENTIAL CHANGES
- EMERGING TECHNOLOGIES

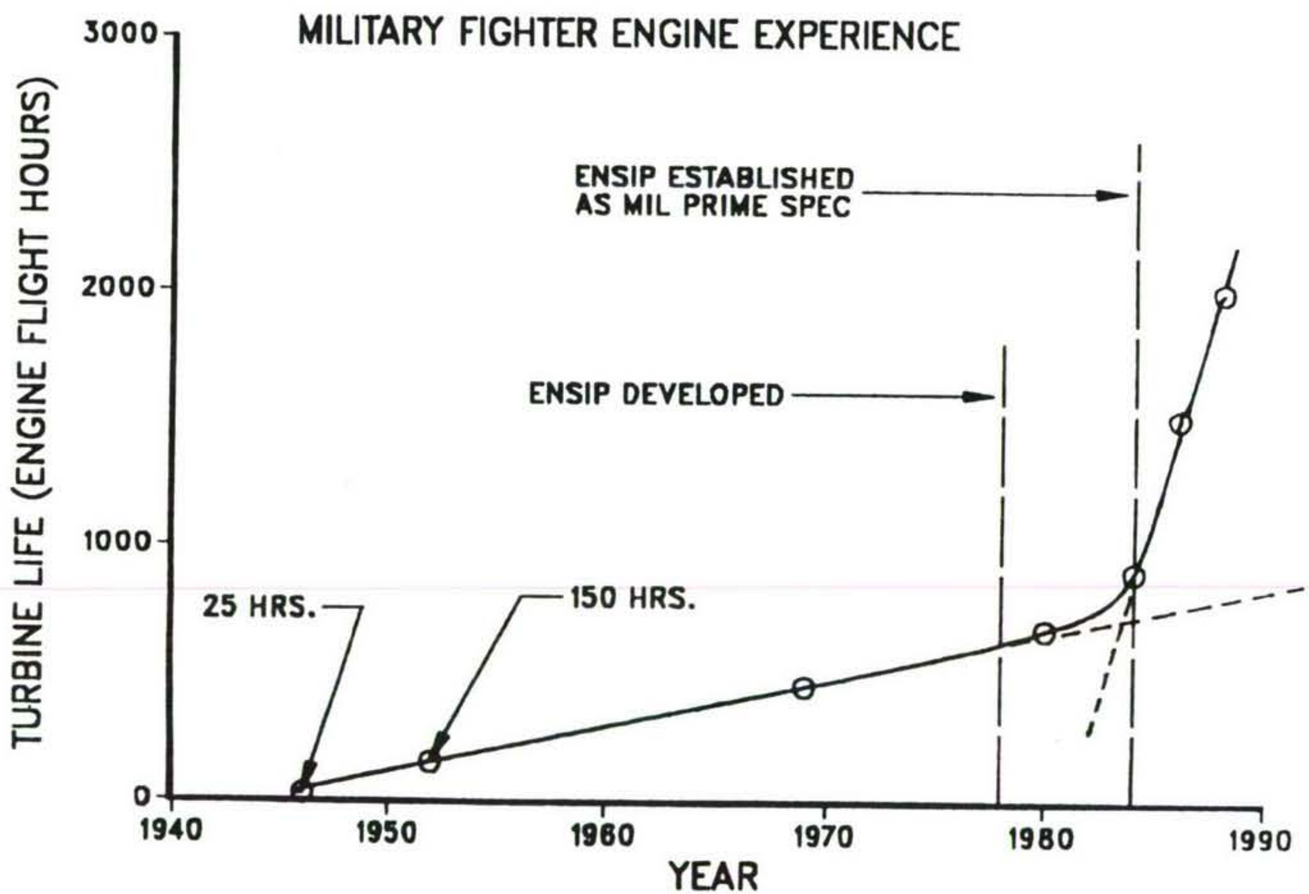
ENSIP ENGINES



2

ENSIP BENEFITS

USEFUL LIFE OF CRITICAL COMPONENTS SIGNIFICANTLY IMPROVED RESULTING IN REDUCED COST OF OWNERSHIP



EXPERIENCE

- ENSIP HAS REDUCED OR CONTROLLED MOST OF STRUCTURAL FAILURE MODES EXCEPT AEROMECHANICAL HIGH CYCLE FATIGUE

- USAF HIGH CYCLE FATIGUE REVIEW
  - SIGNIFICANT PROBLEM ACROSS SEVERAL MANUFACTURERS

AEROMECHANICAL/HCF

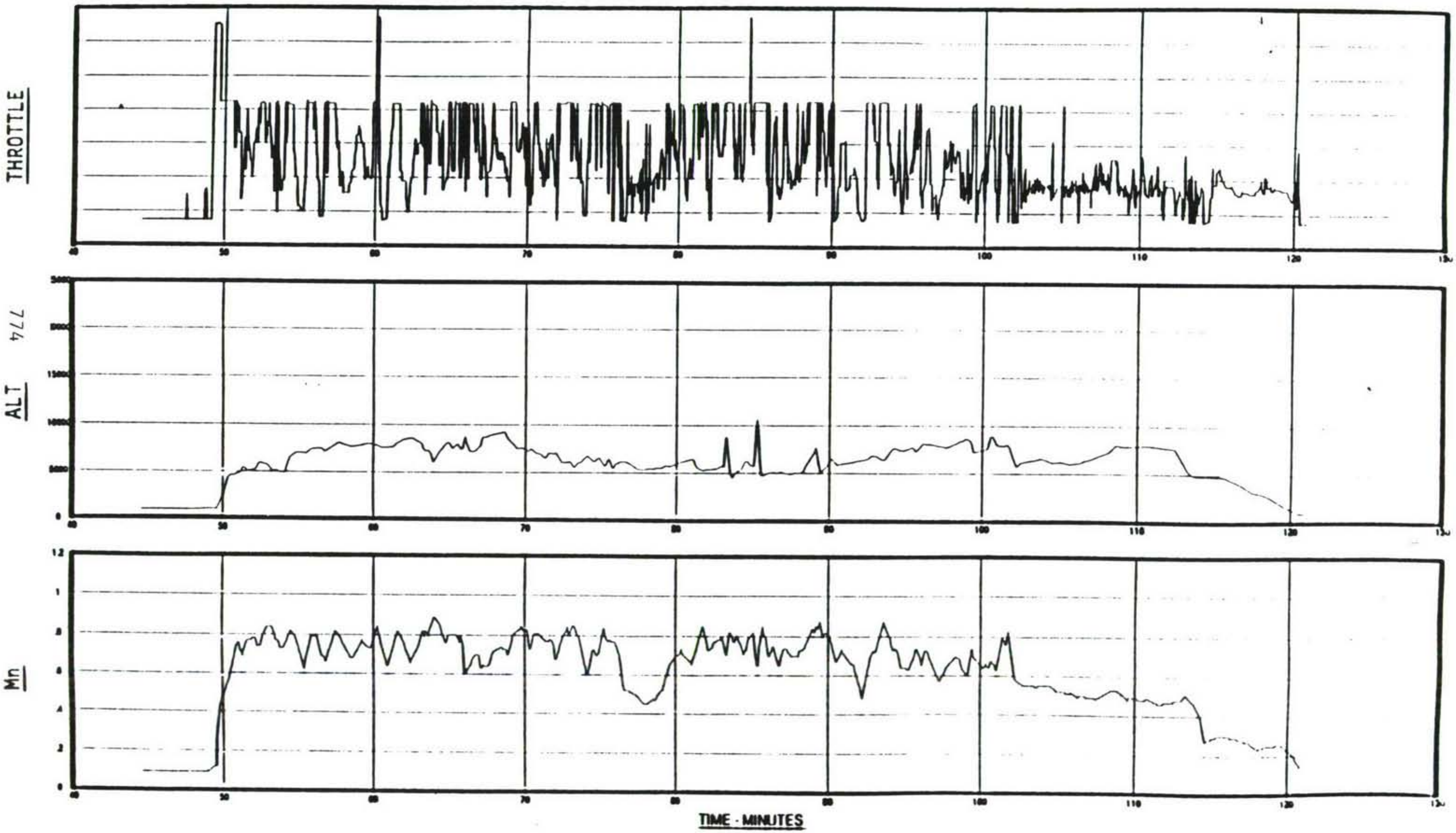
SOME OF THE PROBLEMS

WE ARE FACED WITH!!

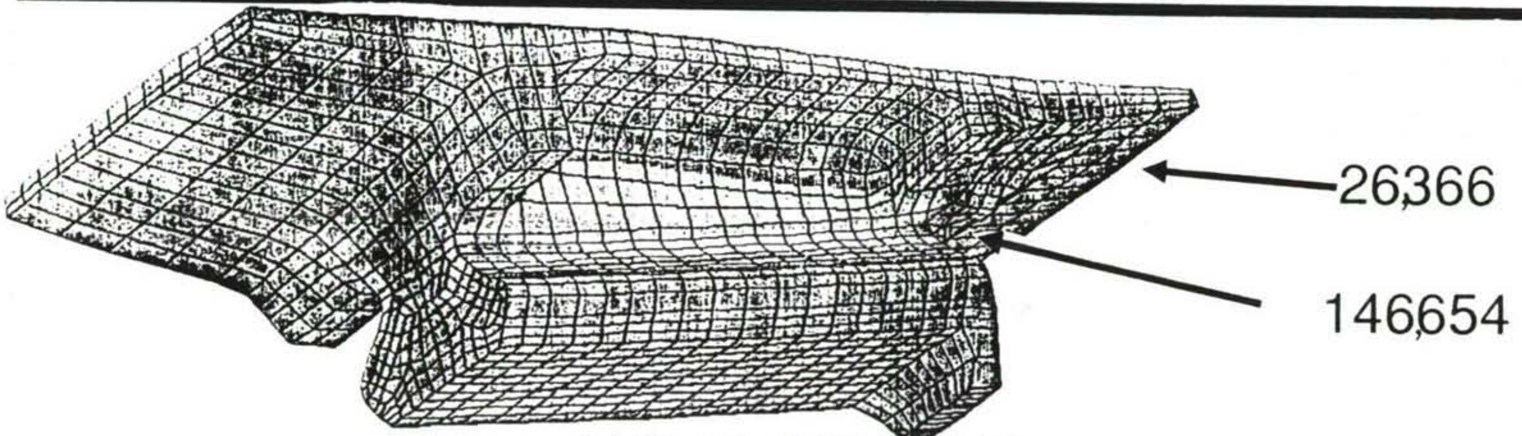
TYPICAL FIELD

WILD WEASEL

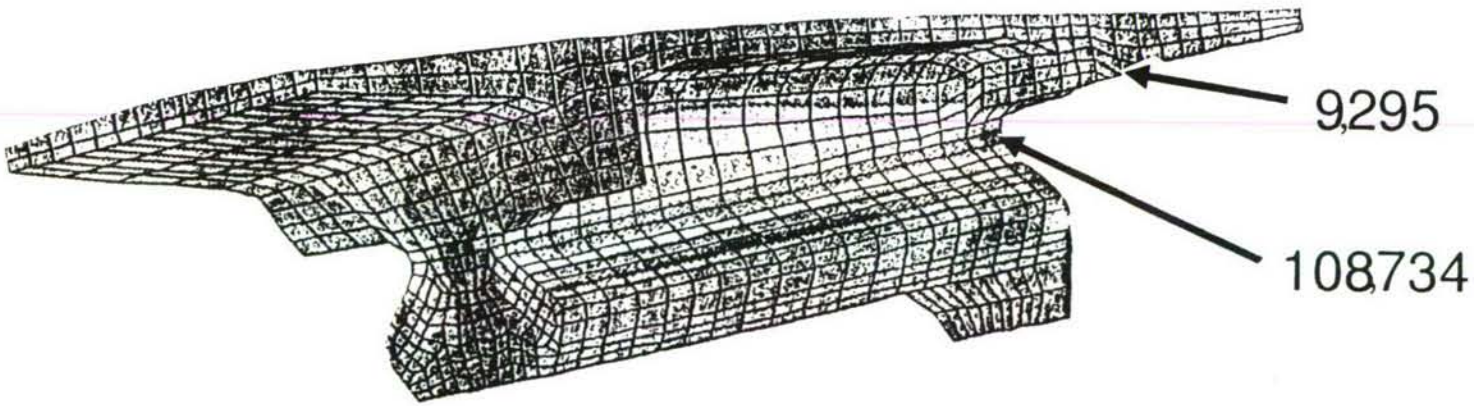
•ROTOR SPEED TRANSIENTS:



STEADY STATE STRESS DETERMINATION



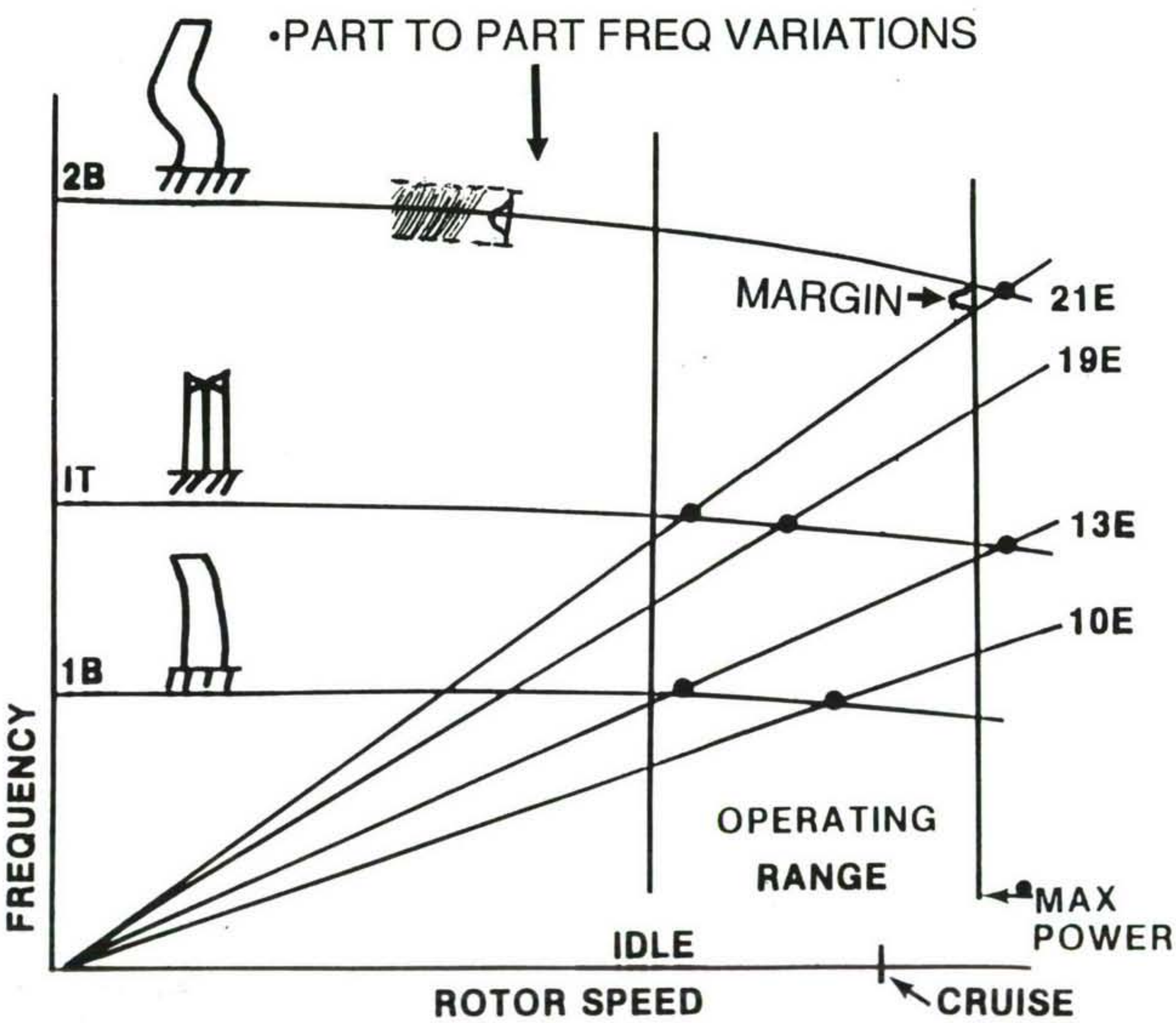
NO FRICTION



WITH FRICTION

DESIGN VARIABLES

TYPICAL BLADE RESONANCE DIAGRAM

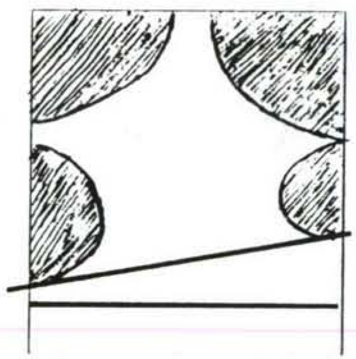


VARIABLES

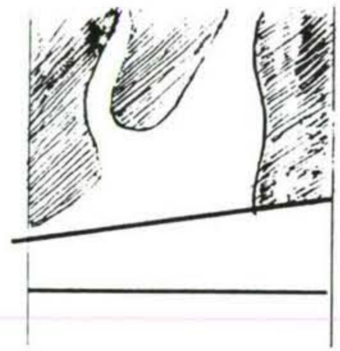
- COMPONENT STIFFNESS
- NO. OF EXCITERS
- STRESS MARGIN
- DAMPING

EXPERIENCE (CONTINUED)

- MODAL COMPRESSION
  - TREND IS TO WIDE CHORD (LOW ASPECT RATIO) BLADES
    - MORE MODES IN OPERATING RANGE
      - MORE COMPLEX MODES, HIGHER FREQUENCIES



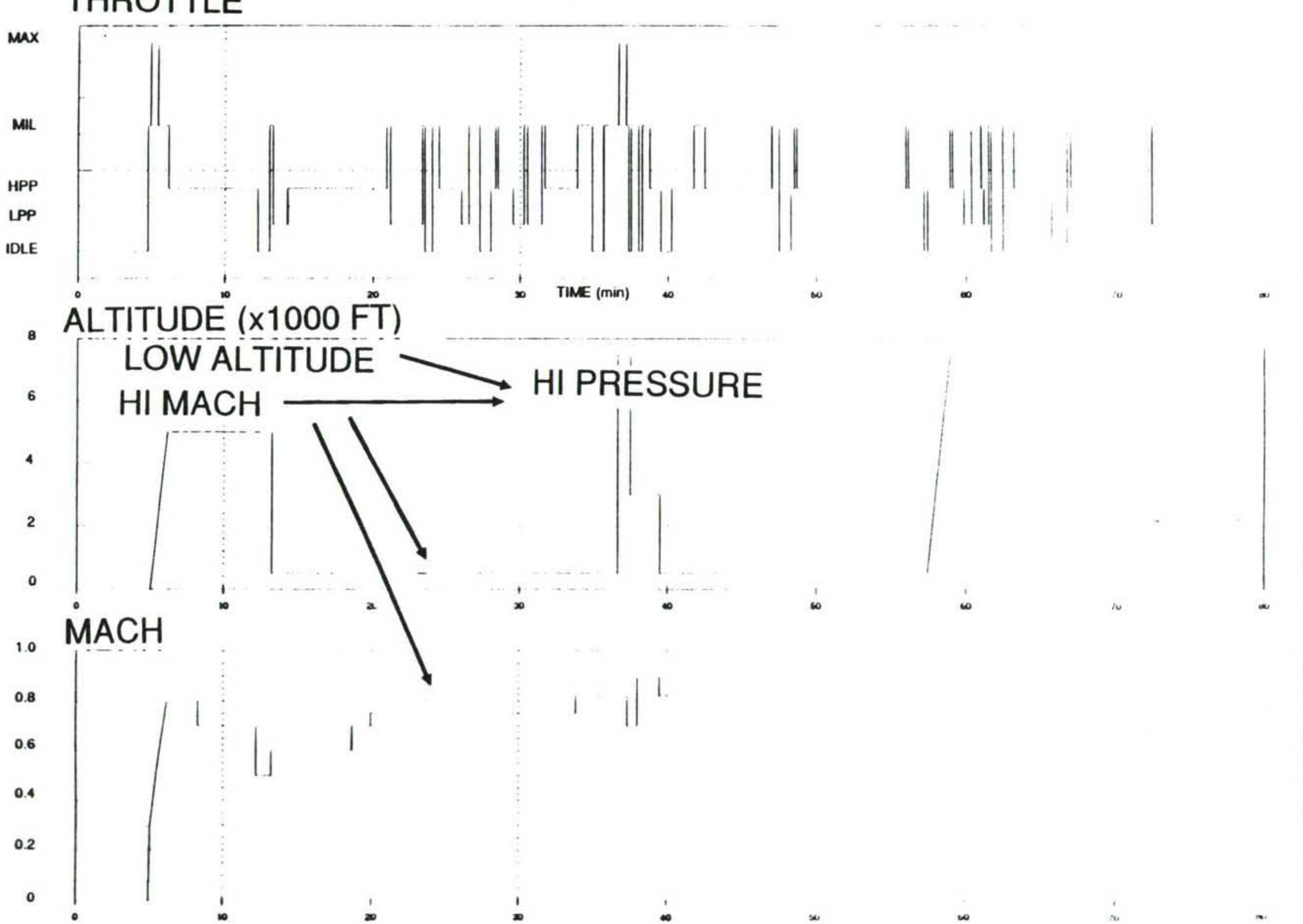
5840-6220HZ
8th ORDER MODE



6150 - 6470HZ
9th ORDER MODE

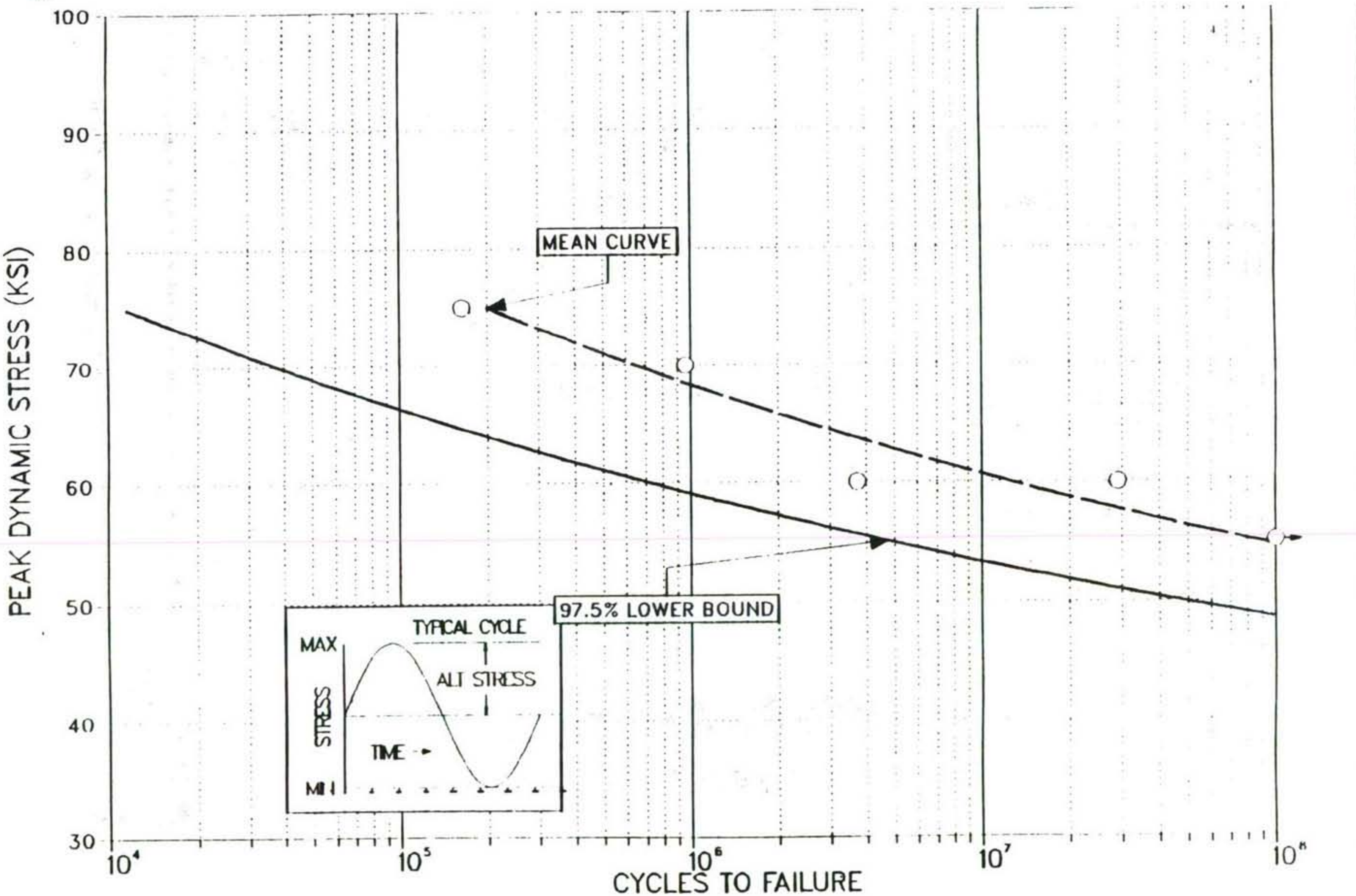
- CRACK CAN INITIATE AND GROW IN ONE MODE AND THEN SWITCH TO ANOTHER MODE PRIOR TO FAILURE. THIS CHANGES FAILURE PATTERN AND CAN ALTER FAILURE CONSEQUENCES.

IPE SURFACE ATTACK TACTICS



10

TI MATERIAL



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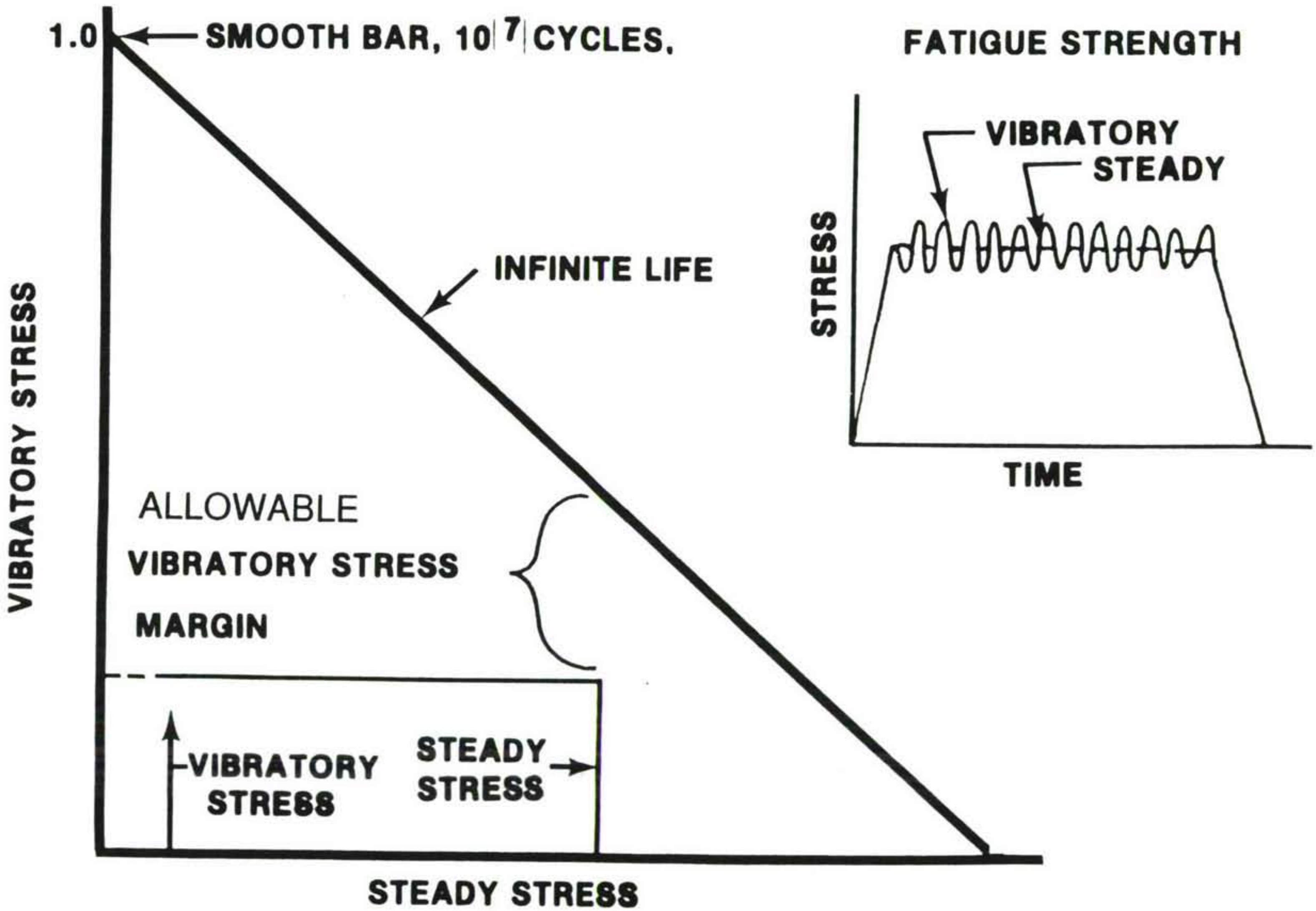
DESIGN VARIABLES
GOODMAN DIAGRAM



AEROMECHANICAL

WHAT WE ARE FACED WITH!! (CONT'D)

- ROTOR SPEED TRANSIENTS
- STEADY STATE STRESS DETERMINATION
- PART TO PART FREQ. VARIATIONS
- MODAL COMPRESSION
- PRESSURE EFFECTS - BEARING LOADS - VIB STRESS
- MATERIAL CHARACTERIZATION
  - ENDURANCE LIMITS - 10^9 VS 10^7
  - FREQ. EFFECTS ON ENDURANCE LIMITS
  - CYCLIC MEAN EFFECTS

- TEMPERATURE EFFECTS
- THERMAL STRESS EFFECTS
- INSTABILITIES
- AMB TEMP EFFECTS
- DAMPING VARIABILITIES
- WEAR
- GAGE RELIABILITY
- TELEMETRY: NOISE, G FORCES, 250°F TEMP LIMITS
- TIP RUBS
- NOISE
- SCREECH
- INLET DISTORTION
- BLEED
- DATA REDUCTION
- TEST FACILITY CAPABILITY
- TEST COST & SCHEDULE
- TEST FACILITY AVAILABILITY

ENSIP AEROMECHANICAL DESIGN REQ.

- ENGINE & COMPONENTS; PLUMB & ACCESSORIES......
  NO DETRIMENTAL RES IN OP RANGE

- HCF: STEEL - 10^7
  NON FERROUS - $3x10^7$

- ALLOWABLE: 40% OF MIN GOODMAN DIAGRAM

- FREQUENCY MARGIN AT 100% ROTOR SPEED: 10% MIN

- SHAFT CRITICAL MARGIN: 20% MIN

- FOD: K_t= 3 AND .030 IN L. E. CRACK FOR TWO
  INSPECTION PERIODS

ENSIP
AEROMECHANICAL VERIFICATION REQ
FORCED VIBRATION & INSTABILITIES

DESIGN
ANALYSIS

INSTRUMENTED
BENCH TEST

INSTRUMENTED
CORE TEST

INSTRUMENTED
ENGINE TESTS

AMT + HCF
STAIR STEP

ENSIP AEROMECHANICAL VERIFICATION REQ

•DYNAMIC ANALYSIS: ENGINE, COMPONENTS, PLUMB & ACC.

DEFINE →
- RESONANT FREQUENCIES
- MODE SHAPES
- CRITICAL SYSTEM MODES
- DRIVERS - CAMPBELL DIAGRAM
- ALLOWABLES - GOODMAN DIAGRAM

BENCH TESTS → RESONANT FREQUENCIES, MODE SHAPES

•MECHANICAL IMPEDANCE TEST: ALL PLUMB & ACC. (INST. ON ENGINE)

•VIBRATORY STRAIN SURVEY (CORE AND ENGINE):

EVAL →

- •SPEED TRANSIENTS
- •TEMP
- •PRESSURE
- •DISTORTION
- •STABILITY
- •STALL
- •PRIMARY & SEC CONTROL SYSTEM
- •BLEED
- •PWR EXT
- •ROTOR IMBALANCE
- •NOMINAL & OFF SCHEDULED VANES
- •AMBIENT TEMP

•MATERIAL CHARACTERIZATION: HCF (GOODMAN DIA) 10^7

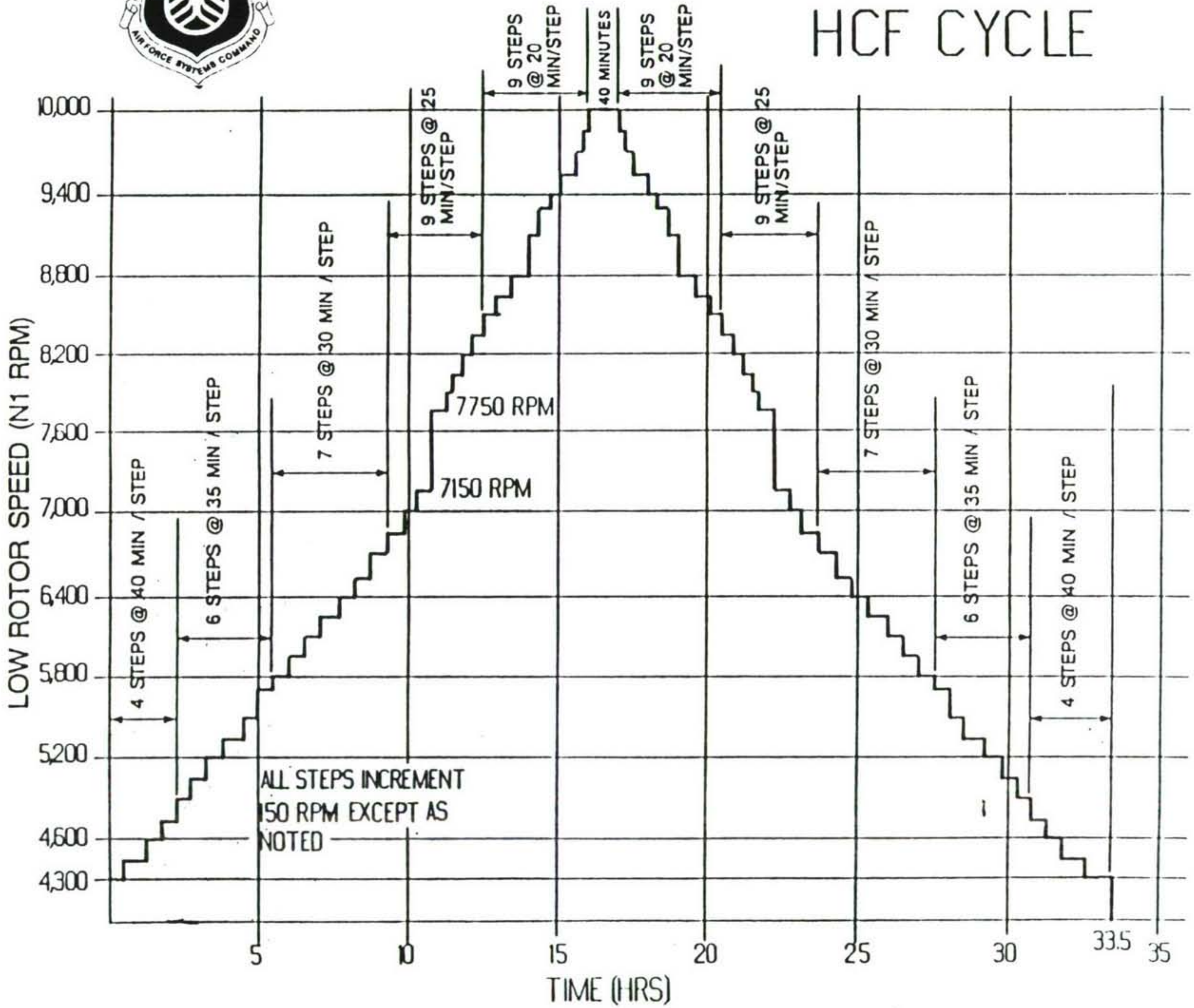
•AMT + HCF STAIRSTEPS (RAM INLET IF MISSION DICTATES)

•OVERSPEED TESTS: 115% FOR 5 MIN.

POTENTIAL CHANGES

- CONDUCT HCF TESTS AT RAM CONDITION
- USE STATISTICAL METHODS TO EVALUATE PART TO PART VARIATION VS ARBITRARY FACTOR
- MODAL ANALYSIS TO PRECEDE AEROMECHANICAL TESTS (PREDICT EXPECTED RESULTS)
- CHARACTERIZE AND SELECT TEST CONFIGURATION

HCF CYCLE


9 STEPS @ 20 MIN/STEP
40 MINUTES
9 STEPS @ 20 MIN/STEP
9 STEPS @ 25 MIN/STEP
9 STEPS @ 25 MIN/STEP
7 STEPS @ 30 MIN / STEP
7 STEPS @ 30 MIN / STEP
6 STEPS @ 35 MIN / STEP
6 STEPS @ 35 MIN / STEP
4 STEPS @ 40 MIN / STEP
4 STEPS @ 40 MIN / STEP
7750 RPM
7150 RPM
ALL STEPS INCREMENT
150 RPM EXCEPT AS
NOTED
LOW ROTOR SPEED (N1 RPM)
10,000
9,400
8,800
8,200
7,600
7,000
6,400
5,800
5,200
4,600
4,300
TIME (HRS)
5
10
15
20
25
30
33.5
35

POTENTIAL CHANGES

- SPIN CALIBRATE INSTRUMENTATION
- Ti HCF GOODMAN DIA ALLOWANCE
  - $10^{7} \longrightarrow 10^{9}$
  - CYCLIC MEAN + VIB?
- PLAN FOR BACK UP AEROMECHANICAL TEST
  - EST A MIN QUANTITY AND QUALITY OF DATA NECESSARY
- EVALUATE EFFECTS OF NEW DESIGN TECH ON INSTABILITIES:
  - VECTORED THRUST
  - IN-FLIGHT THRUST REVERSAL
  - POINT SHOOT
  - VARIABLE BY-PASS RATIO
  - ACTIVE CLEARANCE CONTROL
  - MORE COMPLEX CONTROLS

19

EMERGING TECHNOLOGIES

- COMPUTATIONAL FLUID DYNAMICS FOR UNSTEADY FLOW FIELDS IN CASCADE
    - PREDICT COMPLEX INTERACTIONS
    - ESTIMATE AERO DAMPING

- OPTICAL MEASUREMENT SYSTEMS
    - STRESSES
    - FLOW FIELDS
    - INSTABILITIES

EMERGING TECHNOLOGIES

- TELEMETRY PACKAGES WITH CAPABILITIES APPROACHING CORE ENGINES (IMPROVED TEMP CAPABILITY)

- SPECTRAL METHODS
  - CHARACTERIZE COMPLEX RESPONSES
  - ALLOW MIXES OF TEST AND ANALYSIS
  - INCLUDE VARIATIONS

SUMMARY

- AEROMECHANICS IS AREA FOR HIGH PAYOFF IMPROVEMENTS WITH ENSIP
- AF IS MAKING SOME CHANGES IN APPLICATION OF ENSIP TO THIS AREA TO IMPROVE PROCESS
- EMERGING TECHNOLOGY WILL PLAY A MAJOR PART IN UNDERSTANDING COMPLEX PHENOMENA IN AEROMECHANICS

RETIREMENT FOR CAUSE PERFORMANCE

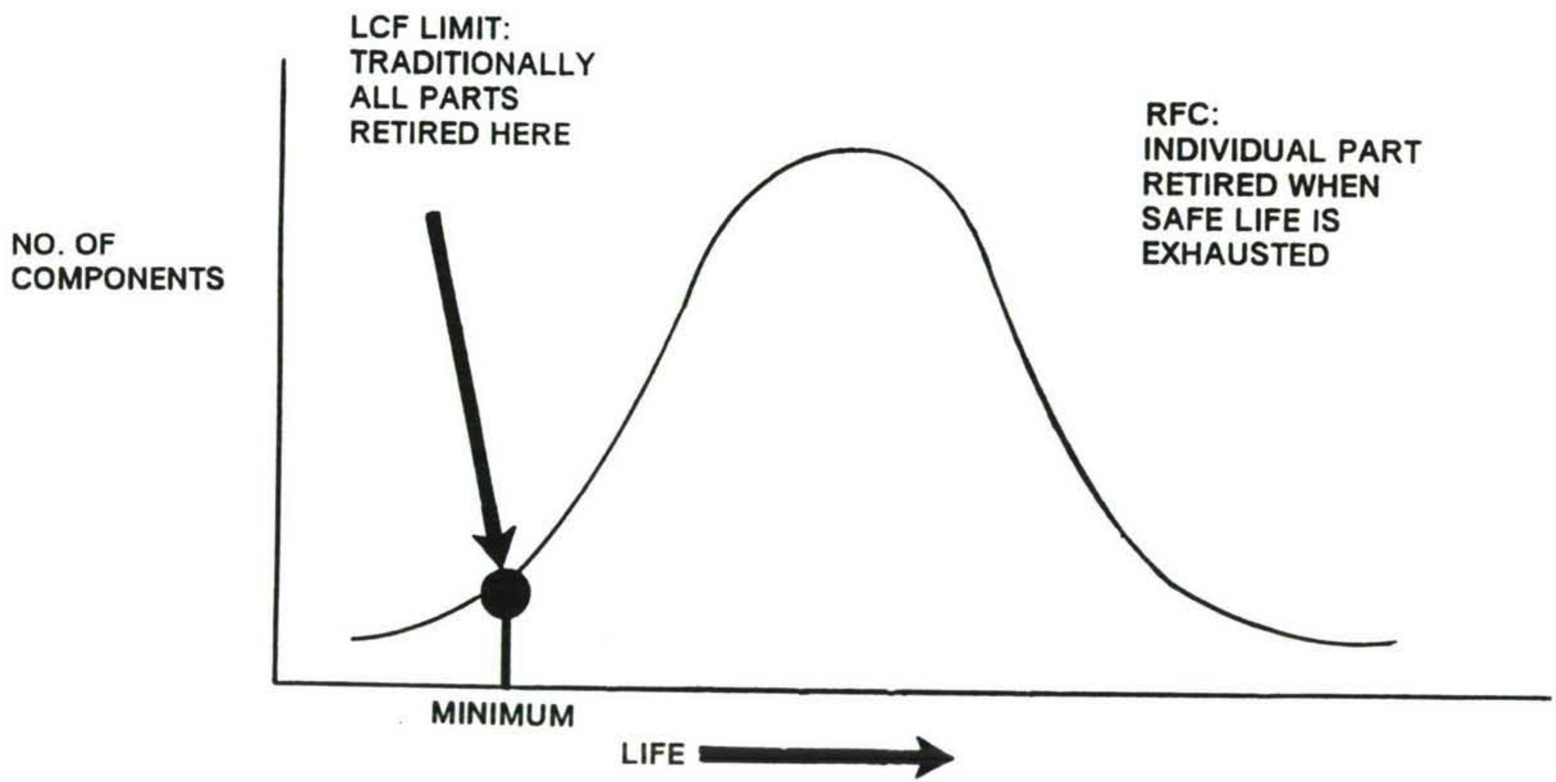
Predicted Vs. Actual

1989 USAF STRUCTURAL INTEGRITY
PROGRAM CONFERENCE
SAN ANTONIO, TEXAS
7 DECEMBER 1989

J.A. HARRIS, JR.

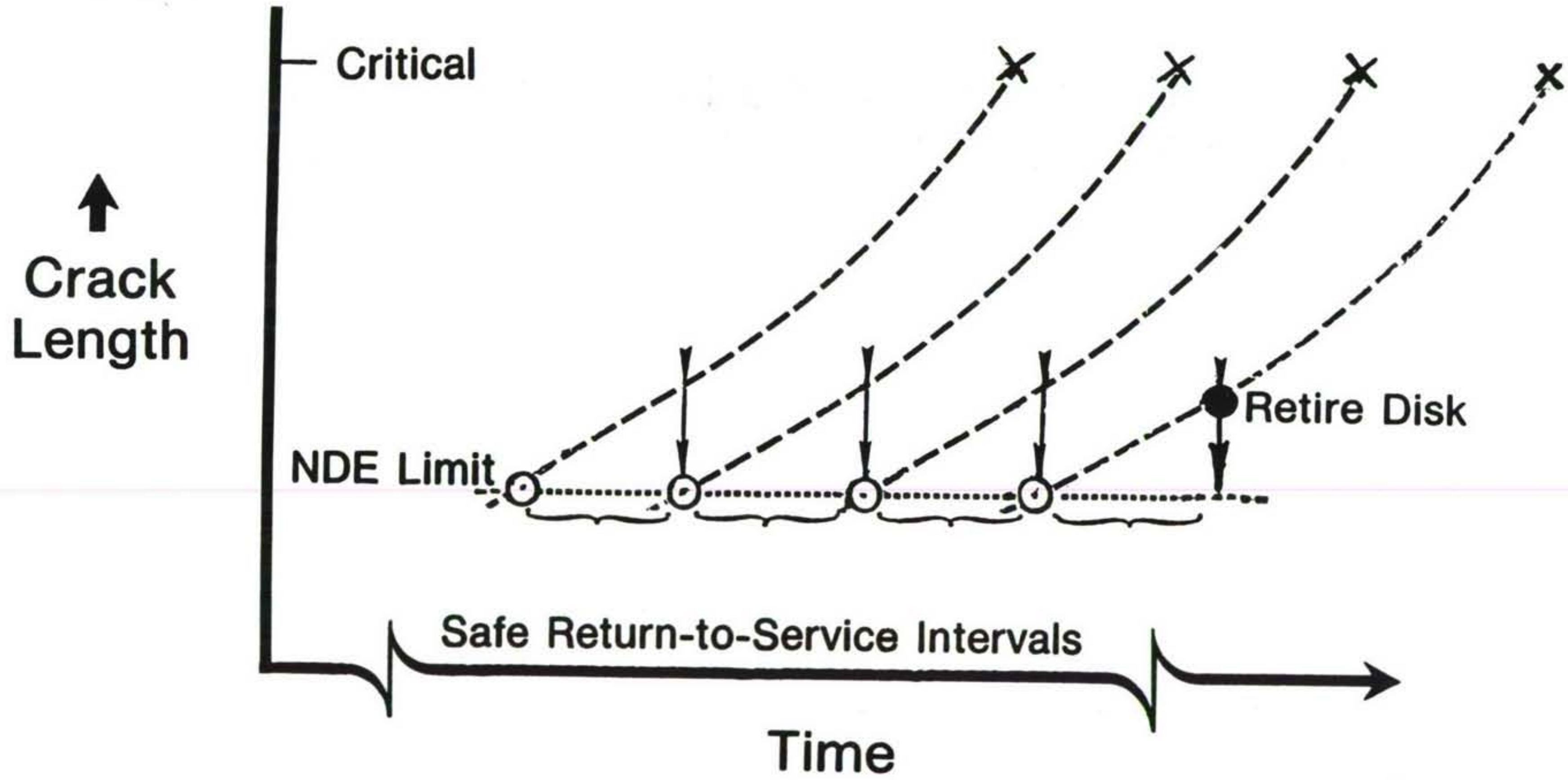
RETIREMENT FOR CAUSE

FRACTURE MECHANICS/NDE BASED PROCEDURE FOR MANAGEMENT OF LIFE LIMITED COMPONENTS



RETIREMENT FOR CAUSE CONCEPT

Inspect and Return to Service Until Cause for Retirement Is Found



AV 213446
810502
gn1-411

INTENT OF RETIREMENT FOR CAUSE

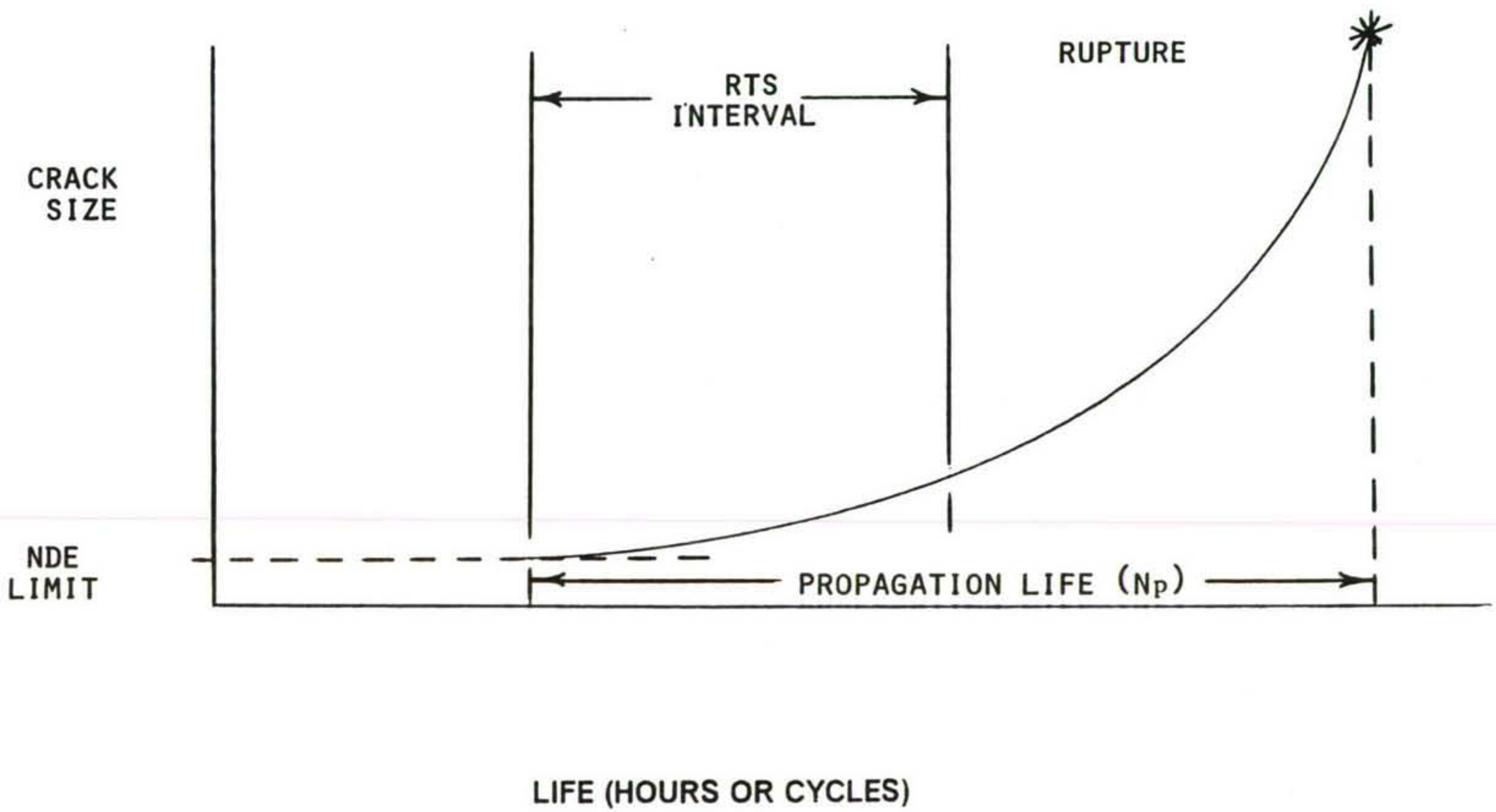
Safe Utilization Of Component Life

- ENABLES UTILIZATION OF ALL CRACK INITIATION LIFE AVAILABLE IN A COMPONENT POPULATION

- DOES NOT ADVOCATE USE OF CRACK PROPAGATION LIFE EXCEPT TO PROVIDE SAFETY MARGIN

PROPAGATION MARGIN

ASSURES SAFETY IN APPLICATION OF RFC



PROPAGATION MARGIN = PROPAGATION LIFE/RTS INTERVAL

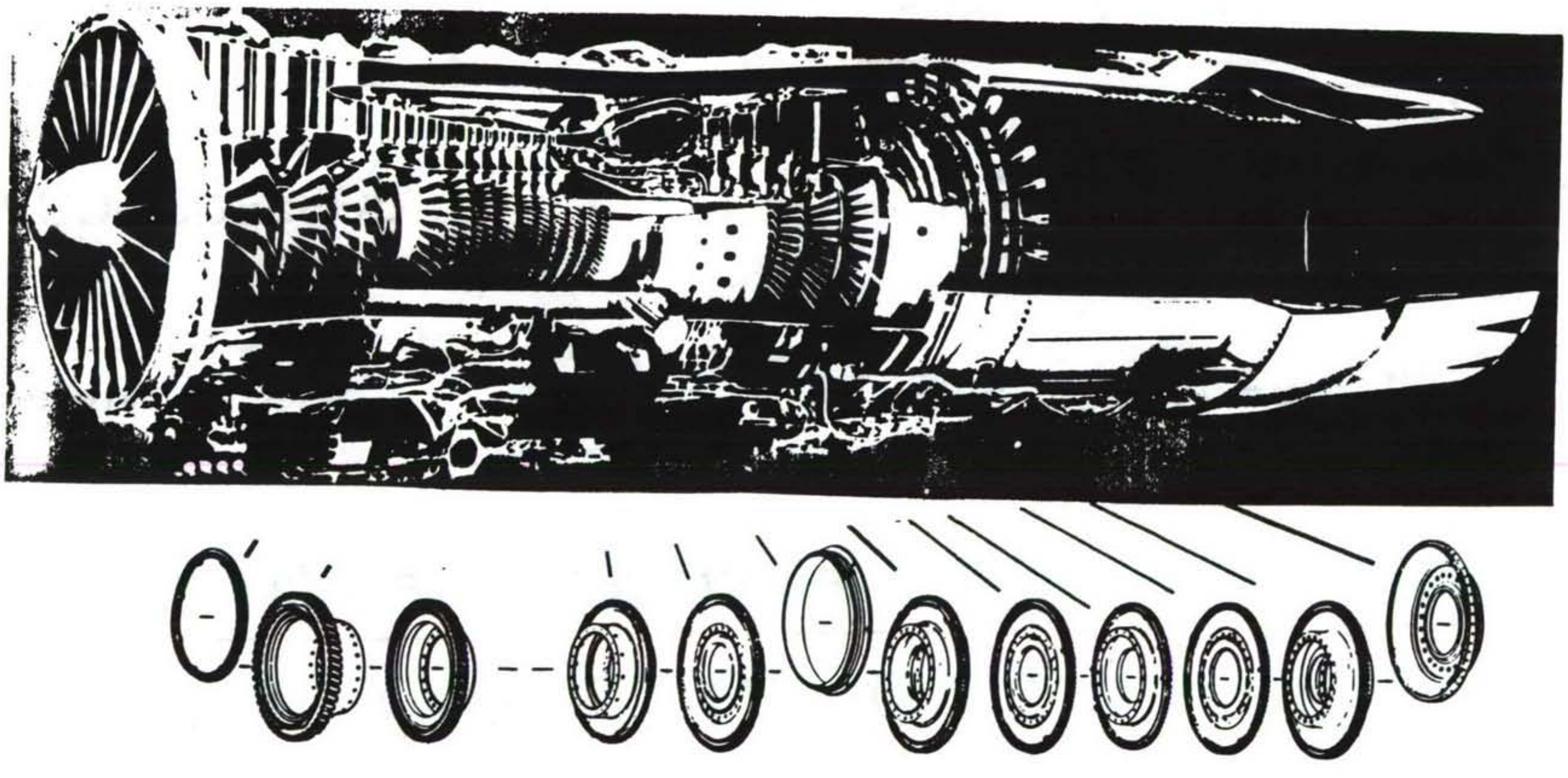
SELECTING COMPONENTS FOR RFC

Two Technical, One Economic Criteria

- COMPONENT LCF LIFE LESS THAN PROJECTED SYSTEM LIFE
- COMPONENT CRACK PROPAGATION LIFE GREATER THAN THE RETURN TO SERVICE INTERVAL WITH APPROPRIATE PROPAGATION MARGIN
- INSPECTION/REWORK COST LESS THAN COST OF NEW COMPONENT

USAF F100 TURBOFAN ENGINE

Fighter Engine With Typical Rotor Components



FOR THE F100 ENGINE

23 Rotor Components Meet Criteria

- FAN - 3 DISKS, 1 SPACER
- COMPRESSOR - 9 DISKS, 7 SPACERS
- FAN DRIVE TURBINE - 2 DISKS, 1 SPACER
- INSPECTION REQUIREMENTS DEFINED FOR EACH COMPONENT

IS RFC WORTH IT?

THE BOTTOM LINE FOR THE F100 ENGINE

RETURN:

- $966 MILLION PARTS COST SAVINGS
- $303 MILLION LABOR AND FUEL SAVINGS

INVESTMENT:

- $ 52 MILLION DEVELOPMENT/FACILITIZATION COST

<u>$1.2 BILLION NET LIFE CYCLE COST SAVINGS</u>

FIRST IMPLEMENTATION IN 1985 - 1986

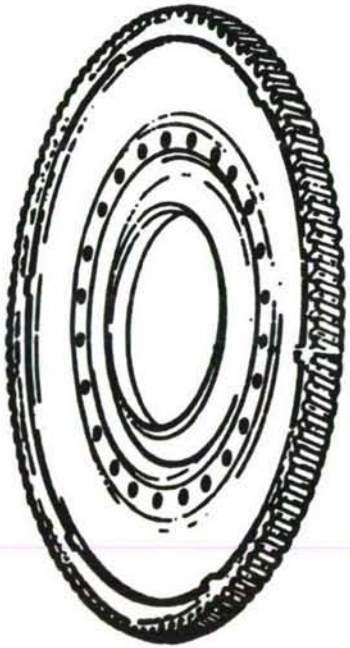
For Fan And Core Components

FAN
- DISKS VIA CRYOGENIC PROOF
- SPACER VIA EDDY CURRENT INSPECTION

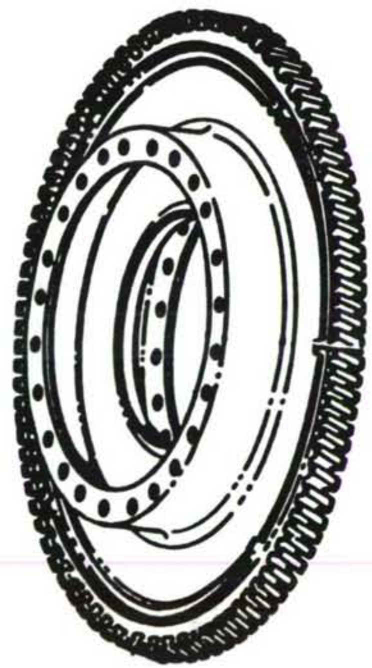
CORE-COMPRESSOR (HPC)
- DISKS AND SPACERS VIA EDDY CURRENT INSPECTION IN CONJUNCTION WITH CORE UPGRADE PROGRAM

HIGH PRESSURE COMPRESSOR HAS 10 DISKS

9 Disks Subject To RFC At Upgrade



4, 6*, 8, 10, 12th
STAGE

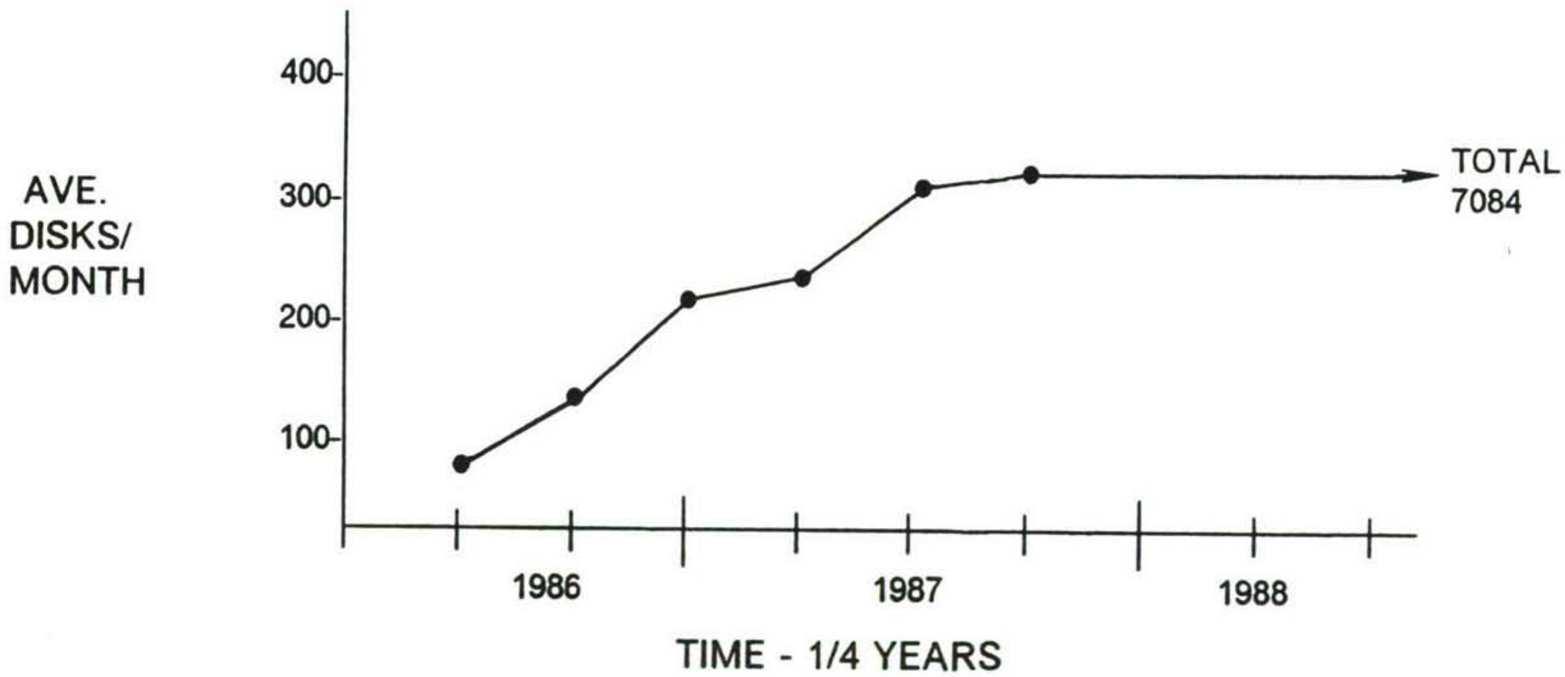


5, 7, 9, 11, 13th
STAGE

*NOT RFC.

INSPECTION RATE OF COMPRESSOR DISKS

For First 2 Years Of Implementation (5/86 - 6/88)



PREDICTED AND ACTUAL REJECT RATES

For HPC Disks At Upgrade Window (3600-5400 TAC)

DISK STAGE	PREDICTED % REJECT RATE	NUMBER INSPECTED	NUMBER REJECTED	ACTUAL % REJECT RATE
4th	0.8-5.0	687	21	3.1
5th	3.5	763	21	2.8
7th	11.8	736	89	12.1
8th	2.9	738	20	2.7
9th	11.4-18.0	841	91	10.8
10th	2.0	671	19	2.8
11th	4.2-10.8	678	51	7.5
12th	7.8	1003	48	4.8
13th	12.3-20.7	967	120	12.4
Aggregate	6.7-9.7	7084	480	6.8

ESTIMATED RFC COST SAVINGS

Returned To Service Instead of Replaced (1986 Base $)

DISK STAGE	COMPONENT COST AVOIDANCE	TOTAL LCC SAVINGS PREDICTED AT SYSTEM LIFE
4th	4.7	21.0
5th	5.9	44.4
7th	7.6	28.7
8th	4.7	25.9
9th	9.8	49.8
10th	4.4	19.3
11th	8.7	74.9
12th	6.9	36.2
13th	11.7	53.6
	$64.4 Million	$353.8 Million

SAVINGS COMPARISON FOR COMPRESSOR DISKS

Predicted Vs. Actual Thru June 1988 (1986 Base $)

- 790 ACTUAL OF 6500 PREDICTED OPPORTUNITIES = 12%
- 12% OF PREDICTED $353.8M SAVINGS = $43.0M
- ACTUAL SAVINGS: $64.4M - $2.7M COSTS = $61.7M

12% OF TOTAL OPPORTUNITIES YIELDS 17% OF SAVINGS

RFC PERFORMANCE FOR COMPRESSOR DISKS

Summary For First Two Years

- PROBABILISTIC LIFE ANALYSIS SYSTEM VERIFIED
- REJECT RATES AT OR BELOW PREDICTED
- SAVINGS GREATER THAN PREDICTED
- NO SERVICE INCIDENTS

THE USAF-ALC DAMAGE TOLERANCE ASSESSMENT PROGRAM

(An Informal Review of a Report prepared by the USAF-SAB)

by

H. Norman Abramson
Southwest Research Institute
San Antonio, Texas

A luncheon address
at the 1989 USAF Structural Integrity Program Conference

San Antonio, Texas
7 December 1989

I AM VERY PLEASED TO BE ABLE TO REPORT TO YOU TODAY, INFORMALLY, THE RESULTS OF A STUDY RECENTLY CONDUCTED AT THE REQUEST OF GENERAL HANSEN AT ALC HEADQUARTERS. THE REPORT WAS PREPARED BY A CROSS-MATRIX PANEL OF THE AIR FORCE SCIENTIFIC ADVISORY BOARD, OF WHICH I AM PLEASED TO BE A MEMBER. THE OTHER MEMBERS OF THIS PANEL INCLUDED BILL LEHMANN OF THE UNIVERSITY OF NEW MEXICO (SERVING AS CHAIRMAN); DICK GABRIEL FROM MCDONNELL-DOUGLAS; HOWARD W. LEAF, USAF RET.; SAM MUSA FROM E-SYSTEMS; JOHN HOUBOLT, RETIRED FROM NASA LANGLEY RESEARCH CENTER; LOU MONTULLI FROM BOEING AIRCRAFT COMPANY; AND MAX WILLIAMS, DEAN EMERITUS AT THE UNIVERSITY OF PITTSBURGH. THE PANEL WAS ASSISTED GREATLY BY JACK LINCOLN, WHOM YOU ALL KNOW, AND BY PHIL PANZARELLA, CHIEF SCIENTIST OF THE AIR FORCE LOGISTICS COMMAND.

I THINK THAT I WANT TO START OFF THE VERY FIRST THING BY TELLING YOU THAT THIS REPORT IS SOMEWHAT UNIQUE IN ONE VERY IMPORTANT RESPECT-IT WAS A REPORT PREPARED BY AN ADVISORY GROUP WHOSE ADVICE HAS ACTUALLY BEEN TAKEN! IN FACT, GENERAL HANSEN

TOOK IMMEDIATE AND POSITIVE ACTION TO ADDRESS THE RECOMMENDATIONS OF THE REPORT AND ORDERED THE DEVELOPMENT OF AN IMPLEMENTATION PLAN, WHICH HAS NOW BEEN COMPLETED, AND THEREFORE WE CAN ACTUALLY SEE SOME THINGS HAPPENING! THIS HAS BEEN A REMARKABLE RESPONSE TO AN ADVISORY COMMITTEE REPORT, PARTICULARLY SINCE EVERYTHING WAS PUT INTO MOTION WITHIN JUST A FEW SHORT WEEKS FOLLOWING SUBMITTAL OF THE REPORT.

I AM GOING TO GIVE YOU AN INFORMAL REVIEW OF OUR REPORT, FREELY INJECTING SOME OF MY OWN THOUGHTS AND PREJUDICES. BUT FIRST, LET ME GIVE YOU THE PRECISE ASSIGNMENT UNDER WHICH WE WORKED. WE WERE ASKED BY GENERAL HANSEN "TO UNDERTAKE A BROAD ASSESSMENT OF THE DAMAGE TOLERANCE ASSESSMENT (DTA) METHODOLOGY, PROCEDURES, POLICIES, AND THE EFFECTIVENESS OF DTA TECHNOLOGY AS APPLIED INTERNALLY WITHIN THE AIR FORCE LOGISTICS COMMAND." RATHER THAN A STRAIGHTFORWARD RECITATION OF CONCLUSIONS AND RECOMMENDATIONS, I AM GOING TO TRY TO SUMMARIZE THE REPORT IN A FEW PRINCIPAL TOPIC AREAS. FIRST OF ALL, THERE ARE THE INTERRELATIONSHIPS BETWEEN MANAGEMENT AND ENGINEERING

REQUIREMENTS; SECONDLY, THERE ARE THE TECHNICAL SKILLS (PERSONNEL) THAT ARE REQUIRED FOR SUCCESSFUL IMPLEMENTATION OF DAMAGE TOLERANCE ASSESSMENT; AND, FINALLY, THERE ARE SOME TECHNICAL ISSUES THAT ARE OF MAJOR CONCERN FOR THE FUTURE DEVELOPMENT AND IMPLEMENTATION OF THE DTA METHODOLOGY.

IN THE FIRST AREA, THAT OF INTERACTIONS BETWEEN MANAGEMENT AND ENGINEERING, THERE WERE SEVERAL SIGNIFICANT CONCLUSIONS DEVELOPED BY THE STUDY PANEL. FIRST OF ALL, WE VERY RAPIDLY CONCLUDED FROM A GREAT DEAL OF AVAILABLE EVIDENCE THAT BOTH THE PRESENT AIR FORCE AIRCRAFT INTEGRITY PROGRAM AND THE DAMAGE TOLERANCE ASSESSMENT PROGRAM ARE TECHNICALLY SOUND IN APPROACH. THEY PROVIDE ASSURANCE OF STRUCTURAL INTEGRITY, WHEN PROPERLY EXECUTED. UNFORTUNATELY, THIS EXECUTION IS NOT CARRIED OUT BY ALL OF THE SPM'S (SYSTEM PROGRAM MANAGER) WITH THE SAME DISCIPLINE AND COMPETENCE. FURTHERMORE, WE FOUND THAT THE AIR FORCE LOGISTIC COMMAND MECHANISMS FOR PROVIDING FOR FUNDING TECHNICAL PERSONNEL AND FOR THE ACQUISITION OF TECHNICAL EQUIPMENT DO NOT ALWAYS PROVIDE THE QUICK RESPONSE AND THE

FLEXIBILITY THAT IS REALLY NEEDED BY TECHNICAL ORGANIZATIONS. FOR EXAMPLE, EACH CENTER CARRIES OUT ITS DTA RESPONSIBILITIES INDEPENDENTLY. OUR PANEL OBSERVED THAT THERE IS A LOT OF VOLUNTARY COORDINATION, BUT THERE IS NO CENTRAL TECHNICAL AUTHORITY WITHIN THE COMMAND. WE ALSO NOTED THAT WHILE THE WARNER-ROBBINS DTA LABORATORY IS RESPONSIBLE FOR ENGINEERING SUPPORT TO ALL OF ITS SYSTEM PROGRAM MANAGERS AND, IN ADDITION, HAS BEEN DESIGNATED AS THE AIR LOGISTICS COMMAND TECHNOLOGY APPLICATIONS PROGRAM MANAGER FOR DAMAGE TOLERANCE ASSESSMENT, ITS RESPONSIBILITIES AND AUTHORITIES TO CARRY OUT THOSE FUNCTIONS ARE NEITHER WELL DEFINED NOR CLEARLY UNDERSTOOD BY ALL OF THE COMMAND DAMAGE TOLERANCE ASSESSMENT ACTIVITIES.

IN ORDER TO OVERCOME SOME OF THE PROBLEMS THAT THOSE CONCLUSIONS ILLUSTRATE, WE RECOMMENDED, THEREFORE, THAT SPECIFIC LOGISTIC COMMAND RESPONSIBILITIES BE ASSIGNED TO THE TECHNOLOGY APPLICATIONS PROGRAM MANAGER IN ORDER TO ASSURE THE DISCIPLINED EXECUTION OF AN OVERALL HIGH QUALITY DTA PROGRAM. THE TECHNOLOGY APPLICATIONS PROGRAM MANAGER SHOULD BE SPECIFICALLY CHARGED

TO BE RESPONSIBLE FOR THE FOLLOWING KINDS OF THINGS:

- TO ESTABLISH MEASURES AND CRITERIA FOR DTA PROCEDURES AND TECHNICAL SUPPORT TO THE SYSTEM PROGRAM MANAGERS.

- TO TEST AND PROTOTYPE NEW DTA PROCEDURES WITHIN THE LOGISTIC COMMAND.

- TO ASSIST ALL OF THE ALC'S IN IMPLEMENTING NEW TECHNOLOGY.

- TO DELIVER SPECIALIZED ENGINEERING ASSISTANCE TO THE LOGISTICS COMMAND AIRCRAFT STRUCTURAL INTEGRITY PROGRAM MANAGERS AS NEEDED.

- TO BE AN ADVOCATE FOR RESOURCES AND PROCEDURES TO IMPROVE THE DTA CAPABILITY THROUGHOUT THE LOGISTICS COMMAND.

- TO ASSIST THE USING COMMANDS TO IMPLEMENT THESE IMPROVED DTA MEASUREMENT AND INFORMATION TECHNIQUES.

- TO ESTABLISH PROCEDURES TO INCLUDE CORROSION ANALYSIS IN DTA.

- FINALLY, TO EVALUATE DTA EXECUTION FOR EACH WEAPON SYSTEM IN CONJUNCTION WITH AN ANNUAL AIRCRAFT STRUCTURAL INTEGRITY CONFERENCE.

FURTHERMORE, WE RECOMMENDED THAT THE COMMAND SEEK THE NECESSARY FUNDING AND PROCUREMENT MECHANISMS TO GIVE THE LOGISTICS COMMAND DTA PROGRAM THE SAME FLEXIBILITY THAT, SAY, 3600 FUNDING GIVES THE AIR FORCE RESEARCH DEVELOPMENT AND ACQUISITION ORGANIZATIONS.

A SECOND AREA COVERED BY THE REPORT HAD TO DO WITH CURRENT LEVELS OF TECHNICAL SKILLS. ONE OF OUR CONCLUSIONS WAS THAT THE LOGISTICS COMMAND NEEDS ABOUT 40 TO 50 ENGINEERS TO CARRY

OUT THE NECESSARY IN-HOUSE DTA FUNCTIONS, BUT IT HAS CURRENTLY ONLY ABOUT 25. FURTHERMORE, THE CURRENT LOGISTICS COMMAND GRADE STRUCTURE FOR DTA ENGINEERS IS NOT COMPETITIVE, WHICH LEADS IN PART TO THIS SHORTAGE, BUT ALSO RESULTS IN AN UNDESIRABLE SITUATION OVERALL WITH REGARD TO STAFFING. WE RECOMMENDED TO GENERAL HANSEN THAT 15-20 DTA ENGINEERS BE NEWLY EMPLOYED AND DEPLOYED ACROSS THE COMMAND. THESE ENGINEERS COULD BE AIR FORCE CIVILIANS, OR THEY COULD BE DEDICATED CONTRACTOR SUPPORT PERSONNEL. WE ALSO RECOMMENDED THAT THE COMMAND ESTABLISH IMPROVED PERSONNEL POLICIES FOR DTA ENGINEERS. FOR EXAMPLE, THE COMMAND SHOULD RECRUIT OR EDUCATE MORE ENGINEERS WITH ADVANCED GRADUATE EDUCATION IN DTA. IT SHOULD ALSO SEEK HIGHER TOP-GRADE LEVELS IN AN ATTEMPT TO ASSURE MORE RECRUITING FLEXIBILITY AND SHOULD DEVELOP CAREER PROGRESSION OPPORTUNITIES TO BE COMPETITIVE WITH BOTH INDUSTRY AND GOVERNMENT. ALSO, THE COMMAND SHOULD INITIATE A PROGRAM FOR INTERCHANGE OF ENGINEERS WITH THE AIR FORCE SYSTEMS COMMAND, INCLUDING CAREER PROGRESSION FROM THE SYSTEMS COMMAND TO THE LOGISTICS COMMAND. FINALLY, BUT NO LESS IMPORTANT, ALL DTA ENGINEERS WITHIN THE

COMMAND SHOULD HAVE EQUAL AND FULL ACCESS TO ALL OF THE STATE-OF-THE-ART TOOLS THAT MAKE DTA AN EFFECTIVE ACTIVITY.

I TURN NOW TO A FEW OF THE MORE IMPORTANT TECHNICAL ISSUES. FIRST OF ALL, THERE IS NO QUESTION THAT CORROSION IS A MAJOR CAUSE OF STRUCTURAL FAILURES, BUT IT IS NOT PRESENTLY FORMALLY INCORPORATED INTO THE DTA PROCESS. THIS IS VITALLY NECESSARY. WE ALSO NOTED THAT WHILE EACH SYSTEM PROGRAM MANAGER AGGRESSIVELY INVESTIGATES STRUCTURAL FAILURES, UNFORTUNATELY THERE APPEARS TO BE LITTLE TRANSFER OF "LESSONS LEARNED" BETWEEN THE ALCS. SURELY, THIS NEEDS TO BE IMPROVED. FURTHERMORE, FOR FUTURE SYSTEMS, THE LOGISTICS COMMAND DTA WILL NEED IMPROVED CAPABILITIES TO DEAL WITH NEW MATERIALS, NEW STRUCTURES, AND MODIFICATION PROCEDURES.

WE, THEREFORE, SPECIFICALLY RECOMMENDED THAT DTA BE EXPANDED TO INCLUDE CORROSION AS AN ESSENTIAL PART OF THE METHODOLOGY. WE ALSO ADVOCATE AN ENHANCED R&D PROGRAM FOR

DAMAGE TOLERANCE ASSESSMENT TO THE SYSTEMS COMMAND FOR BOTH EXISTING AND FUTURE AIRCRAFT SYSTEMS. THERE IS A GREAT DEAL THAT CAN AND NEEDS TO BE DONE TO ENHANCE THE METHODOLOGY, AND DOING IT TO THE POINT OF DEPLOYMENT WITHIN THE LOGISTICS COMMAND.

I THINK IT WORTHY OF NOTE, ONCE MORE, THAT GENERAL HANSEN AND HIS STAFF ACCEPTED THIS REPORT WITH GREAT ENTHUSIASM, ASKED FOR AN IMPLEMENTATION PLAN, AND HAS MOVED AGGRESSIVELY FORWARD WITH PUTTING OUR RECOMMENDATIONS INTO EFFECT. FOR THAT, WE ARE VERY GREATFUL, BUT EVEN MORE SO FOR THE ASSURANCE THAT DAMAGE TOLERANCE ASSESSMENT IS A VITAL PART OF LOGISTICS COMMAND RESPONSIBILITIES AND THAT THE COMMAND IS GOING TO DO EVERYTHING POSSIBLE TO MAKE IT AN EFFECTIVE AND USEFUL TOOL TO ENHANCE THE RELIABILITY OF ALL OF OUR AIR FORCE SYSTEMS.

THERE ARE SOME GENERAL LESSONS AND OTHER OBSERVATIONS OUT OF THIS STUDY THAT PROBABLY ARE WORTH BRINGING TO YOUR ATTENTION ALSO. FOR EXAMPLE, I FEEL VERY STRONGLY THAT THERE NEEDS TO BE A GREAT DEAL OF ADDITIONAL EMPHASIS PLACED ON

TRAINING AND EXPERIENCE WITH COMPOSITE STRUCTURES. MORE AND MORE OF OUR AIRCRAFT SYSTEMS CONTAIN COMPOSITE STRUCTURES AND, RATHER THAN JUST MORE RESEARCH AND DEVELOPMENT, WE NEED TO DO MUCH MORE HANDS-ON TRAINING AND TO DEVELOP A BODY OF EXPERIENCE IN UTILIZING THE TOOLS APPLICABLE TO THESE PARTICULAR STRUCTURES.

WITH RESPECT TO CORROSION, WHICH I HAVE ALREADY MENTIONED A COUPLE OF TIMES, THE NEED IS NOT ONLY TO INCORPORATE CORROSION ISSUES WITHIN THE METHODOLOGY, BUT TO HAVE QUALIFIED INSPECTORS AND NEW NON-DESTRUCTIVE INSPECTION TECHNIQUES AIMED SPECIFICALLY AT DETECTING AND ANALYZING THE DEPTH OF CORROSION DAMAGE. COMMERCIAL AVIATION INDICATES THAT CORROSION AND CORROSION-INDUCED MAINTAINANCE IS A VERY LARGE COST ITEM. THE NAVY AGREES. MORE ATTENTION MUST BE GIVEN TO CORROSION CONTROL IN THE DESIGN PROCESS, AS WELL AS IN MANUFACTURING AND MAINTENANCE. NEW ADVANCES ARE NEEDED IN CORROSION DETECTION BOTH BY VISUAL INSPECTION AND BY NEW NDI METHODS. THIS INCLUDES CORROSION INSPECTION UNDER PAINTS OR COATINGS AND THE DETECTION

OF CORROSION PHENOMENON NOT READILY APPARENT, SUCH AS HYDROGEN EMBRITTLEMENT AND STRESS CORROSION CRACKING.

IN CONNECTION WITH THE TRAINING OF NDI INSPECTORS WE HAVE A RATHER SERIOUS PROBLEM AS SO MUCH OF THIS DEPENDS ON EXPERIENCE AND WE HAVE FAIRLY LARGE NUMBERS OF PEOPLE WITH ONLY LIMITED EXPERIENCE. FURTHERMORE, INADEQUATE LEVELS OF COMPENSATION FOR THESE TYPES OF PERSONNEL HAVE GREATLY INHIBITED OUR ABILITY TO RECRUIT SKILLED AND CAPABLE INDIVIDUALS. TO PERFORM EDDY-CURRENT INSPECTIONS ON ALL FASTENER HOLES IN AN AIRCRAFT IS A TEDIOUS JOB. WE ARE FREQUENTLY LOOKING FOR SMALL CRACKS OVER LARGE SURFACE AREAS, AND INSPECTIONS ARE MORE THAN LIKELY DONE UNDER TIME (SCHEDULE) CONSTRAINTS AND UNDER DIFFICULT CONDITIONS (COLD, HOT, LIMITED ACCESS). THERE IS TREMENDOUS OPPORTUNITY IN THE HUMAN FACTORS AREA TO INCREASE THE RELIABILITY AND SPEED OF INSPECTIONS.

FINALLY, I THINK THAT I SHOULD EMPHASIZE THE VERY IMPORTANT ROLE OF MULTI-SITE DAMAGE. DTA IN ITSELF IS OF VITAL IMPORTANCE,

BUT WITHOUT CONSIDERATION OF WHAT WE HAVE SEEN RECENTLY IN CERTAIN COMMERCIAL AIRCRAFT, MULTI-SITE DAMAGE IS GOING TO REVEAL PROBLEMS HERETOFORE UNSUSPECTED. BOTH THE FAA AND NASA HAVE ACTIVE PROGRAMS TO STUDY MULTI-SITE DAMAGE AND TO DEVELOP ANALYSIS PROCEDURES FOR DESIGN AND REPAIR.

ONE MORE AREA THAT NEEDS TO HAVE SOME ATTENTION HAS TO DO WITH BETTER REPAIR PROCEDURES AND THE NECESSITY FOR EVALUATING REPAIRS PERIODICALLY. IT IS PROBABLY WORTH NOTING THAT IN THE COMMERCIAL AIRCRAFT FIELD, REPAIRS ARE MADE ON THE BASIS OF STATIC STRENGTH RATHER THAN FATIGUE AND DAMAGE TOLERANCE. EVEN SO, THERE ARE QUITE A FEW AREAS OF EXPERIENCE FROM COMMERCIAL AVIATION THAT WE CAN USE TO OUR ADVANTAGE IN THE MILITARY FIELD.

I PERSONALLY BELIEVE THAT A GREAT DEAL COULD BE GAINED IN ADDRESSING THE AGING AIRCRAFT AND STRUCTURAL INTEGRITY ISSUES BY HAVING MORE INTERACTIONS, BOTH FORMAL AND INFORMAL, BETWEEN THE FAA, THE AIR FORCE, THE NAVY, NASA, AND COMMERCIAL AVIATION

COMPANIES. AN ENGINEER EXCHANGE PROGRAM AMONGST THESE AGENCIES COULD BE BENEFICIAL TO ALL CONCERNED. WHILE THE AIR FORCE HAS ITS OWN AGING AIRCRAFT PROBLEMS, AND ALTHOUGH FIGHTER AIRCRAFT STRUCTURAL INTEGRITY ISSUES ARE DIFFERENT FROM THOSE OF COMMERCIAL AIRCRAFT, THERE IS STILL MUCH TO BE GAINED FROM EACH OTHER. THERE MAY BE SOME PEOPLE IN THESE VARIOUS ORGANIZATIONS WHO GET LOCKED INTO THE NARROW PERSPECTIVE OF THEIR OWN PROBLEMS AND ARE NOT VERY RECEPTIVE TO OPPORTUNITIES OR SOLUTIONS THAT ARE AVAILABLE IN OTHER SEGMENTS OF THE TECHNICAL COMMUNITY, BUT WE NEED TO TRY TO BRING THESE PEOPLE TOGETHER WHENEVER AND WHEREVER POSSIBLE. ONE THING IS FOR SURE; WE DON'T WANT AIRCRAFT FALLING OUT OF THE SKY - IRRESPECTIVE OF WHETHER THEY ARE COMMERCIAL OR MILITARY. IN SOME AREAS, THE MILITARY MAY SHOW THE WAY TO BETTER AIRCRAFT STRUCTURAL INTEGRITY; IN SOME AREAS THE COMMERCIAL SECTOR MAY SHOW THE WAY - IN EITHER CASE, WE BOTH HAVE THE SAME OBJECTIVE AND WE CAN LEARN FROM EACH OTHER.

IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

7 DECEMBER 1989

DR. T. CHRISTIAN AND W.O. GREENHAW
WARNER ROBINS AIR LOGISTICS CENTER

N.K. SNEAD
LOCKHEED AERONAUTICAL SYSTEMS COMPANY- GEORGIA

IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

The assessment of multiple repair Damage Tolerance effects was initiated because of the increasing number of high time aircraft. These aircraft often require structural repairs for damage resulting from corrosion, load induced cracking, and foreign object damage.

Further, the probability of multiple damage in close proximity, requiring multiple repairs, increases with accumulating A/C age.

Previous DTA analysis has indicated that standard wing structural repairs often significantly reduce aircraft safety limits resulting in more stringent inspection requirements.

The question to be answered is whether structural repairs in close proximity to other repairs increases crackgrowth rates and results in increases in inspection requirements. This paper attempts to answer this question and define the scope of the problem.

IMPACT OF WING MULTIPLE REPAIRS

BACKGROUND

- o GENERAL CONCERN FOR AGING AIRCRAFT
- o AIRCRAFT WITH STRUCTURAL REPAIRS
  MULTIPLE REPAIRS IN CLOSE PROXIMITY
- o PREVIOUS DAMAGE TOLERANCE ASSESSMENT
  OF SINGLE STANDARD REPAIRS

IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

In 1988 WR-ALC placed Lockheed-Georgia under contract to study the effects of multiple repairs for wing standard repairs as defined in the C-130 T.O.-3 manual. The scope of analysis discussed in this paper in large measure resulted from the requirements of that contract.

Since the C-130 T.O.-3 contains a large number of standard wing repairs, a major portion of the study addressed selection of a limited set of repairs for assessment. This included surveying current T.O.-3 limits on repair proximity and the development of a repair selection criteria.

Many multiple repair patterns are possible (spanwise, chordwise, etc.) and all could not be assessed in detail for a large number of T.O.-3 repairs. As a result, preliminary analyses were performed to evaluate and select the most critical multiple repair patterns.

Likewise considerable effort was required to define criteria for the downselection of TO-3 repairs for analysis. Once the criteria was defined, the selection of specific repairs for analysis from the large number shown in the TO-3 was completed.

Once the specific T.O.-3 repairs and MR patterns were defined, a conventional DTA analysis, with supporting finite element analysis, was performed to provide data to evaluate inspection requirements and recommend changes in MR limits for the T.O.-3.

IMPACT OF WING MULTIPLE REPAIRS

SCOPE

- o REVIEW C-130 REPAIR MANUAL
- o DEFINE CRITICAL MULTIPLE REPAIR PATTERNS
- o SELECT STANDARD REPAIRS FOR ANALYSIS
- o PERFORM DAMAGE TOLERANCE ASSESSMENT (DTA)
- o RECOMMENDATIONS

IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

A survey of the existing C-130 repair manual showed generally uniform restrictions on multiple spanwise repairs in the center wing. In most cases the spacing is limited to 15″ minimum or none is defined. As the analysis will show, however, the limitations are not always consistent with sensitivity to Damage Tolerance. In some cases, the current restrictions may be overly restrictive, while in others, not restrictive enough.

The survey of outer wing repairs revealed the complete absence of restrictions on spanwise MR. As the DTA results in this paper will show the absence of limits is not supported on a damage tolerance basis.

Limitations on chordwise repairs are generally applied to most repairs (but not all) in both center and outer wings. There is considerable variation in the limits from repair to repair. For example, one repair might be restricted to one per wing panel while another might be restricted to every other riser.

IMPACT OF WING MULTIPLE REPAIRS

MULTIPLE REPAIR SPACING LIMITATIONS SUMMARY
STANDARD REPAIR MANUAL (T.O. 1C-130A-3)

o SPANWISE:
- MINIMUM SPACING = 15″ FOR MOST CENTER WING SURFACE PANEL REPAIRS
- NO SPACING LIMITS FOR OUTER WING SURFACE PANEL REPAIRS

o CHORDWISE:
- LIMITATIONS ON SPACING FOR SOME SURFACE PANEL REPAIRS, IN TERMS OF CONFIGURATION DIMENSIONS (FOR EXAMPLE, ONE RISER PER PANEL)

IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

The first order of business prior to performing the DTA was to determine what types of MR patterns are most critical. Generally, repairs may be installed in essentially an infinite number of positions relative to each other. To simplify the study, three basic forms of MR configurations were considered; spanwise, chordwise, and simple diagonal.

Two basic assumptions were used in all analysis performed.

(1) All external loading was assumed spanwise axial only. C-130 wing surface loading affects on DTA are predominately inboard-outboard axial.

(2) All analysis was performed for homogeneous repair mixes. In other words all repairs in a MR pattern were assumed identical. This assumption is conservative for the worst repair but unconservative on the best when placed in a "mixed" multiple repair pattern. For purposes of qualitative assessment of the types of multiple patterns, this is not important.

These types of MR configurations were most simply assessed using simplified finite element models for generic repair configurations. This allowed quick computer turnaround and assessment of many variables (such as repair spacing, width, length, etc.) in a short time.

IMPACT OF WING
MULTIPLE REPAIRS

DEFINE CRITICAL MULTIPLE REPAIR PATTERNS

- o DEFINE PATTERN GEOMETRIES FOR STUDY
  - SPANWISE
  - CHORDWISE
  - DIAGONAL
- o STRUCTURAL IDEALIZATION ASSUMPTIONS
  - EXTERNAL LOADING SPANWISE ONLY
  - ALL REPAIRS IDENTICAL IN EACH PATTERN
- o PERFORM FINITE ELEMENT MODEL (FEM) PARAMETRIC STUDY

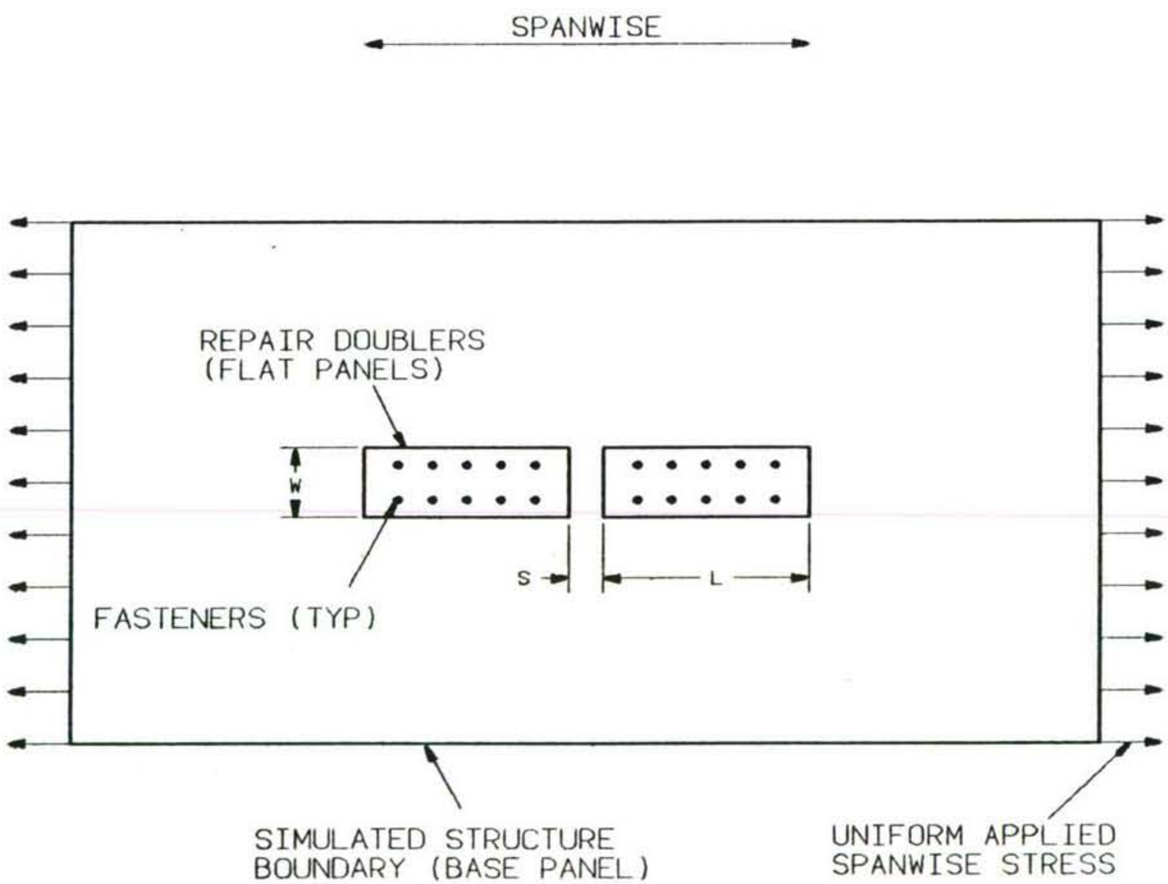
IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

This sketch schematically shows the typical structure used for the **FEM** evaluation of the various **MR** patterns under study. The base structure is a flat, finite width plate with one or more doublers attached via mechanical fasteners. The panel is loaded along its long axis with a far field stress of 10 **KSI**. Each **FEM** served as a simple internal loads model providing repair fastener load distributions and stress peaking at the critical location. For each configuration, the **FEM** maximum fastener load and base panel stress are tabulated and compared for **MR** pattern effects.

The particular pattern shown here corresponds to dual Spanwise Repairs. Each doubler is rectangular (WxL) and spaced at Ss end to end.

IMPACT OF WING MULTIPLE REPAIRS

PARAMETRIC STRUCTURE IDEALIZED CONFIGURATION DUAL SPANWISE REPAIRS

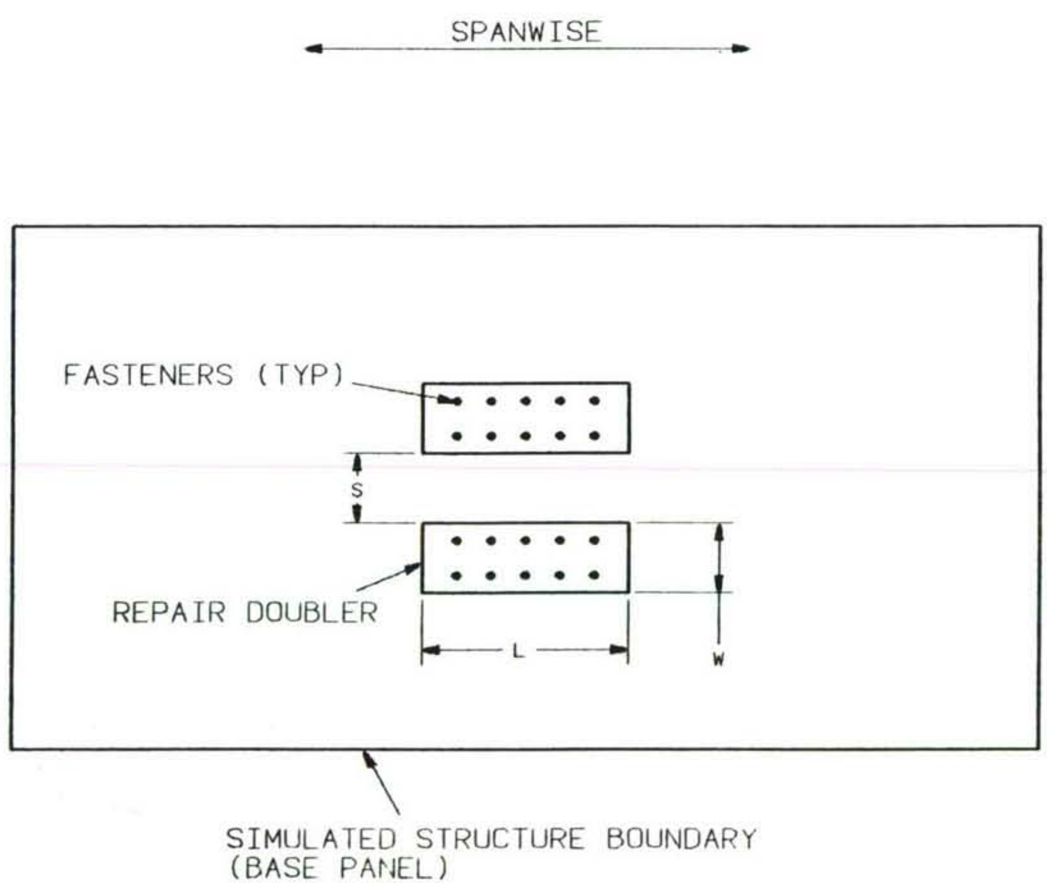


IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

This pattern corresponds to dual Chordwise Repairs. Each doubler is rectangular (WxL) and spaced at Sc edge to edge.

IMPACT OF WING MULTIPLE REPAIRS

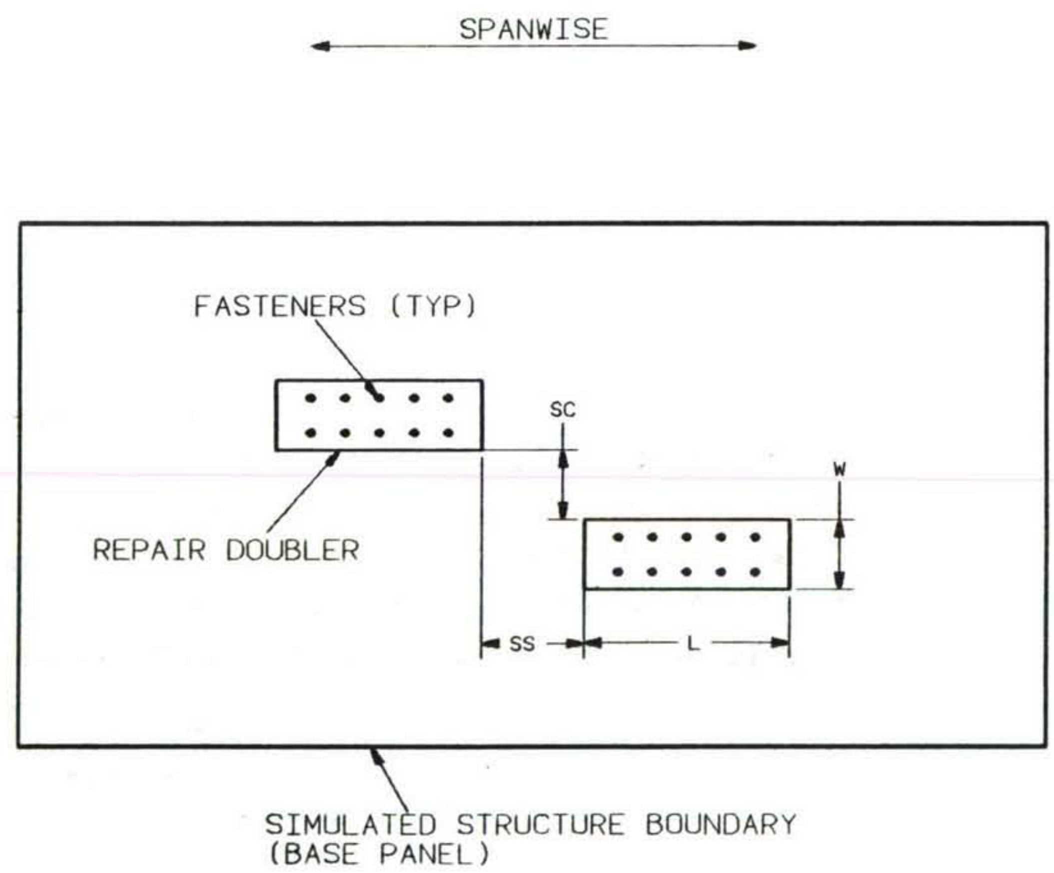
PARAMETRIC STRUCTURE IDEALIZED CONFIGURATION DUAL CHORDWISE REPAIRS



This pattern corresponds to dual Diagonal Repairs. Each doubler is rectangular (WxL) and spaced at Ss (spanwise end to end) and Sc (chordwise edge to edge).

IMPACT OF WING MULTIPLE REPAIRS

PARAMETRIC STRUCTURE IDEALIZED CONFIGURATION DUAL DIAGONAL REPAIRS



IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

Shown here is the list of parameters studied in the parametric **FEM** analysis. They include the doubler rectangular dimensions (L - length and W - width), the spanwise (Ss) and chordwise (Sc) spacing, the doubler to base panel thickness ratio (where base panel tp = 0.100 in. for all cases). The effect of numbers of doublers in a pattern (from one to infinite) was also included.

IMPACT OF WING
MULTIPLE REPAIRS

PARAMETRIC ANALYSIS - FINITE ELEMENT MODEL (FEM) VARIABLES

- o RECTANGULAR REPAIR DOUBLER LENGTH (L)
- o RECTANGULAR REPAIR DOUBLER WIDTH (W)
- o DOUBLER SPACING (S) FOR MULTIPLE REPAIRS
  - SPANWISE SPACING (Ss), CHORDWISE SPACING (Sc)
- o REPAIR DOUBLER TO PANEL THICKNESS RATIO (Td/Tp)
  - PANEL THICKNESS Tp = 0.1″ IN ANALYSIS MODEL
- o NUMBER OF REPAIR DOUBLERS (n)

IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

This sample plot compares peak average local stresses among several generic spanwise multiple repairs. It shows the percent change in average stress, relative to a single repair, near the critical fastener. It is plotted against doubler spanwise end-to-end spacing.

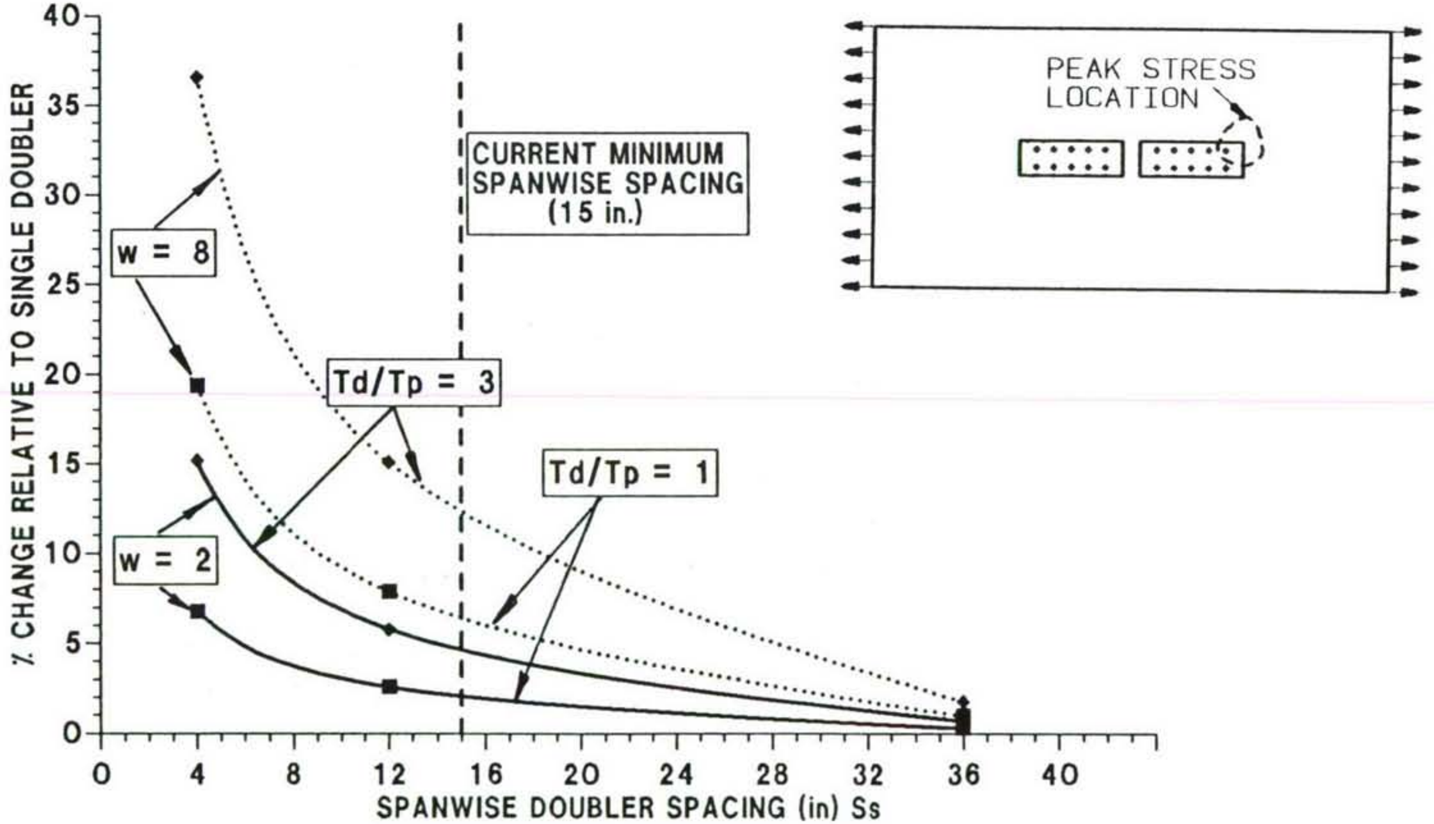
The geometric parameter values represented here are similar to those for many of the repairs defined in the C-130 T.O.-3. Data are shown for doubler parameters:

- length (l) = 36″
- width (w) = 2″ and 8″
- doubler to panel thickness ratio = 1 and 3
- number of doublers = two

Substantial increases in local stress occur as spanwise repairs are placed closer together and as the repairs become increasingly stiff relative to the base structure. Since crackgrowth increases significantly with local stress (and fastener loads), these results can translate to substantially increased crackgrowth rates in real repairs as they are placed in close proximity to one another in a spanwise pattern.

IMPACT OF WING MULTIPLE REPAIRS

PARAMETRIC RESULTS - DUAL 36 IN. LONG SPANWISE REPAIRS PANEL CRITICAL SKIN STRESS INCREASE VS. SPACING

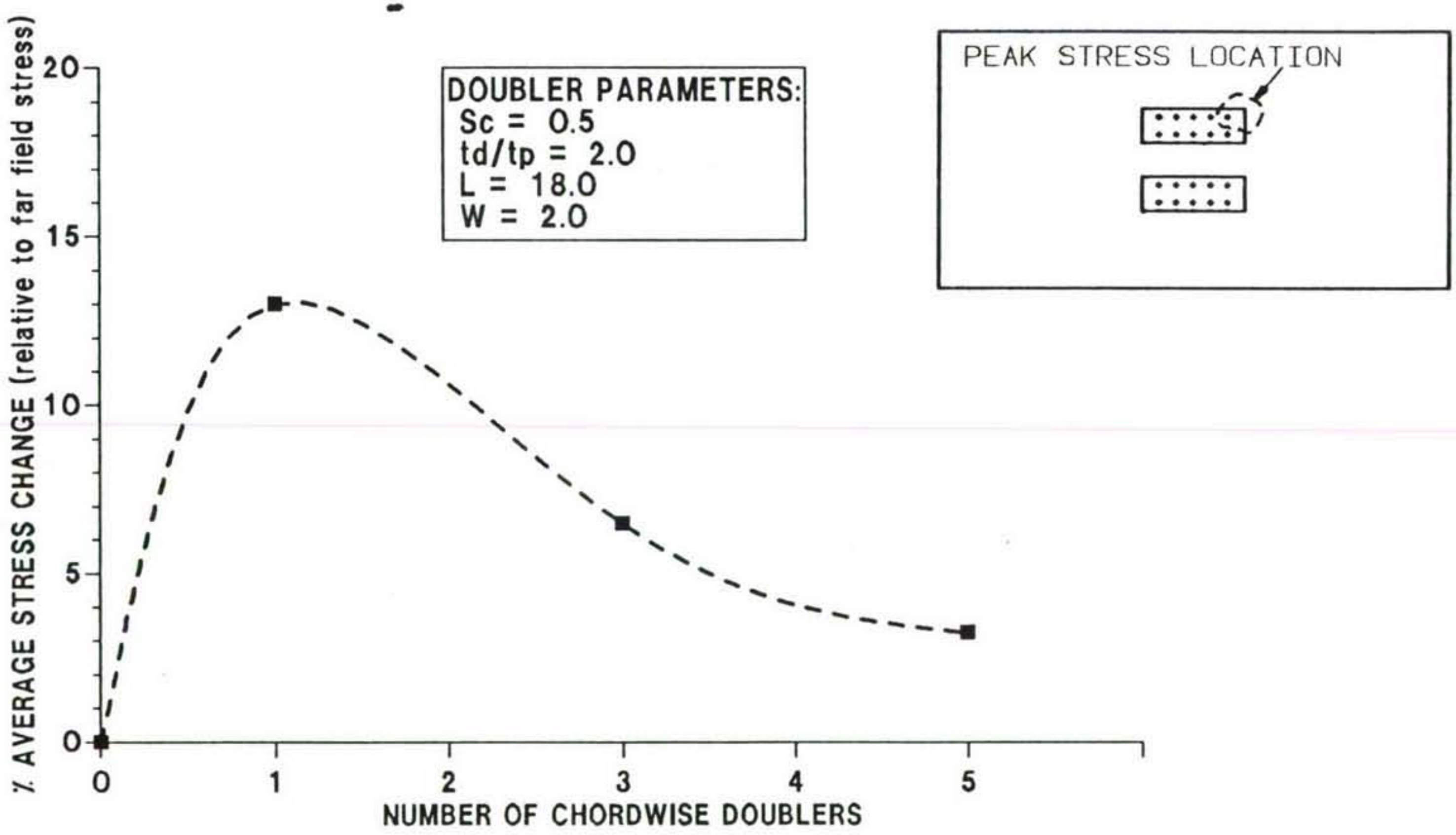


IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

This curve shows the effect of placing multiple doublers side by side in a chordwise pattern. Here the peak average local stress change relative to farfield is plotted against the number of doublers in the pattern. Note the significant increase for the first repair and subsequent decrease as doubers are added. The data shown here is for a selected set of parameters but the same trend occurs for all patterns of this type.

IMPACT OF WING MULTIPLE REPAIRS

PARAMETRIC RESULTS - MULTIPLE IDENTICAL CHORDWISE REPAIRS
PANEL CRITICAL SKIN STRESS CHANGE VS. NUMBER OF REPAIRS



IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

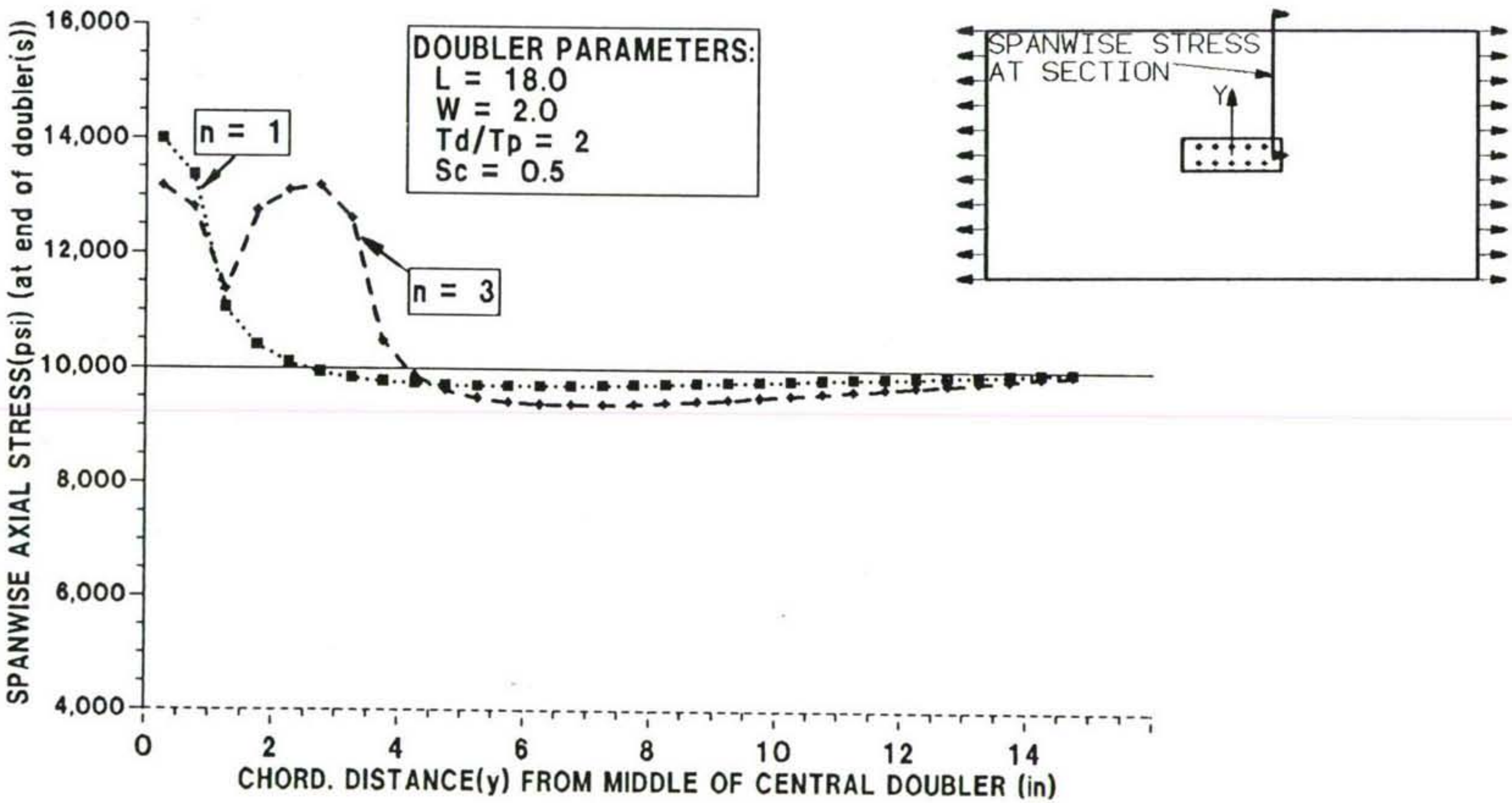
This curve also compares single to chordwise doublers. It shows the base panel stress distribution as it varies chordwise at the end of the doubler(s). This particular distribution corresponds to doubler(s) eighteen inches long by two inches wide. Two curves are shown representing:

(1) Single doubler
(2) Three chordwise doublers

Note that the peak stress always occurs at the outside fastener of the outside doubler. In each case, however, addition of chordwise doublers results in lower peak stress relative to the single doubler.

IMPACT OF WING
MULTIPLE REPAIRS

PARAMETRIC RESULTS - MULTIPLE IDENTICAL CHORDWISE REPAIRS
CHORDWISE DISTRIBUTION OF STRESS FOR FIXED SPACING



IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

Comparison of doubler peak fastener loads and base panel stresses for a wide variety of parameters (thickness, width, length, spacing, and number of repairs) provided an insight into the effects of the various MR configurations.

In all cases, in-line spanwise MR increased fastener loads and stresses when compared to single repairs.

Chordwise MR, on the other hand, always generated lower fastener loads and stresses when compared with single repairs.

Diagonal repairs generated higher stress and fastener loads compared to single repairs. Increases were much less than those shown for spanwise MR for comparable doubler stiffness and spacings.

IMPACT OF WING MULTIPLE REPAIRS

PARAMETRIC RESULT SUMMARY
PEAK STRESS AND FASTENER LOADS

- SPANWISE - INCREASED RELATIVE TO SINGLE REPAIR
- CHORDWISE - NO HIGHER THAN SINGLE REPAIR
- DIAGONAL
  - HIGHER THAN SINGLE REPAIR
  - MUCH LESS THAN MULTIPLE SPANWISE REPAIRS

maintaining homogeneity within each MR, a direct comparison between repair types could be made regarding relative sensitivity to multiple repair conditions.

world damage may occur in any random order requiring placement of these repairs in an infinite number of configurations. As the parametric analysis showed, however, the worst case pattern, for predominately uniaxially loaded structures, was always simple inline spanwise repairs with progressively reduced effects as the doublers are moved to simple chordwise positions.

As a result, analysis of directly inline spanwise repairs were emphasized in this portion of the assessment. Each repair was analyzed for a minimum of two spacings to provide indications of sensitivity to repair spacings and as guidance to recommendations for TO-3 changes.

NOTE: ONE CHORDWISE CONFIGURATION WAS ANALYZED TO CORRABORATATE THE PARAMETRIC RESULTS (FIG 2-26, THE REPAIR MOST SENSITIVE TO SPANWISE MR). RESULTS WERE ENTIRELY CONSISTENT WITH THE PARAMETRIC ANALYSIS. AS ADDITIONAL CHORDWISE REPAIRS WERE ADDED TO THE EXISTING TO-3 REPAIRD DAMAGED AREA, BOTH LOCAL STRESSES AND CORRESPONDING SAFETY LIMITS WERE MARGINALLY REDUCED.

As in the parametric analyses, all TO-3 repairs analyzed in multiple damage patterns were assumed identical. This was an assumption driven by analytical necessity and by the need to understand the sensitivity of each repair independent of other repair types. Selection of mixed multiple repairs and the resulting FEM complexity were not considered cost effective or within the scope of the program. Furthermore, by

IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

Once the general effects of each MR type was understood, selection of specific wing standard repairs for analysis proceeded.

The selection of specific T.O.-3 repairs for analysis was based upon four primary inputs:

(1) Wing locations where spanwise loading predominates.

(2) Repair stiffness - across section of repair stiffnesses with emphasis on the high end.

(3) Single repair DTA results - repairs were generally selected which previously had been shown to be DTA critical for single repairs.

(4) Parametric FEM results - Basic T.O.-3 repair parameters (length, width, thickness, etc.) were used with parametric FEM results to roughly predict stress and fastener load effects for multiple repairs and thereby help in the downselect process.

Using these inputs, a reasonable sample of wing standard repairs was selected for analysis.

Once the specific TO-3 repairs were selected for analysis, definition of the multiple repair pattern(s) to investigate was required. In the real

IMPACT OF WING
MULTIPLE REPAIRS

SELECTION OF MULTIPLE STANDARD REPAIRS CONFIGURATIONS FOR ANALYSIS

- o SELECTION CRITERIA (EIGHT ANALYSED)
  - LOCATIONS WHERE SPANWISE LOADS PREDOMINATE
  - REPAIR-TO-PANEL RELATIVE STIFFNESS
  - SINGLE REPAIR DTA RESULTS
  - PARAMETRIC FINITE ELEMENT MODEL RESULTS

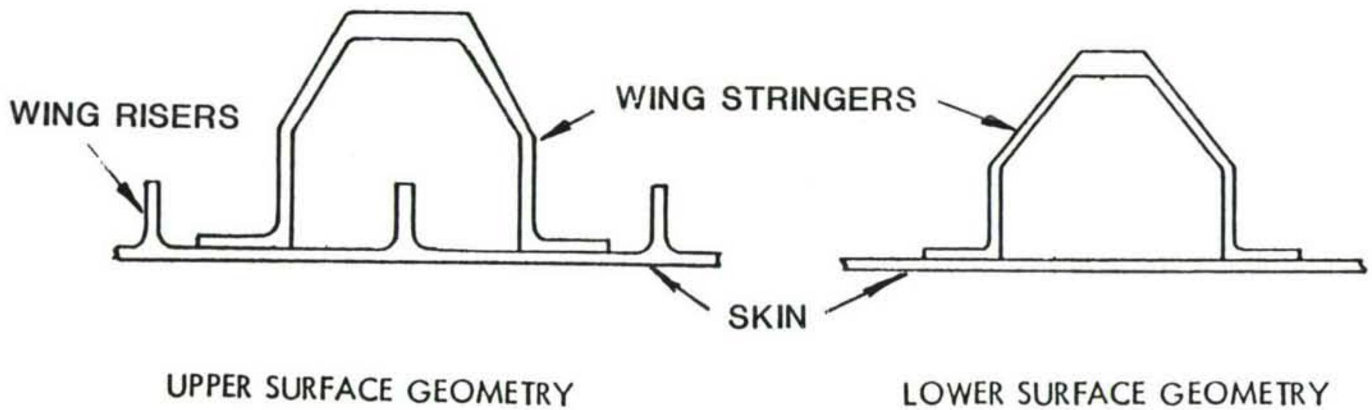
- o MULTIPLE REPAIR PATTERN
  - CHORDWISE REPAIRS NOT CRITICAL FOR DTA CRACKGROWTH
  - ANALYZE SPANWISE REPAIRS FOR MINIMUM OF 2 REPAIR SPACINGS
  - ALL STANDARD REPAIRS IN EACH PATTERN IDENTICAL

IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

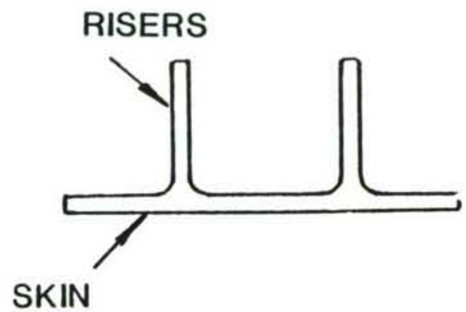
This figure shows the basic C-130 wing surface structure for both the center and outer wings. The center wing plank consists of the skin stiffened with hat section stringers (upper and lower) and interal risers (upper only). The outer wing surface consists of an integrally stiffened skin.

IMPACT OF WING MULTIPLE REPAIRS

TYPICAL C130 WING SURFACE CROSS SECTIONS



CENTER WING



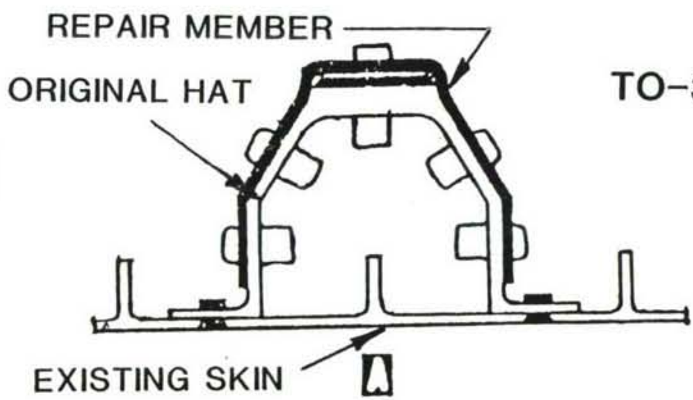
OUTER WING

IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

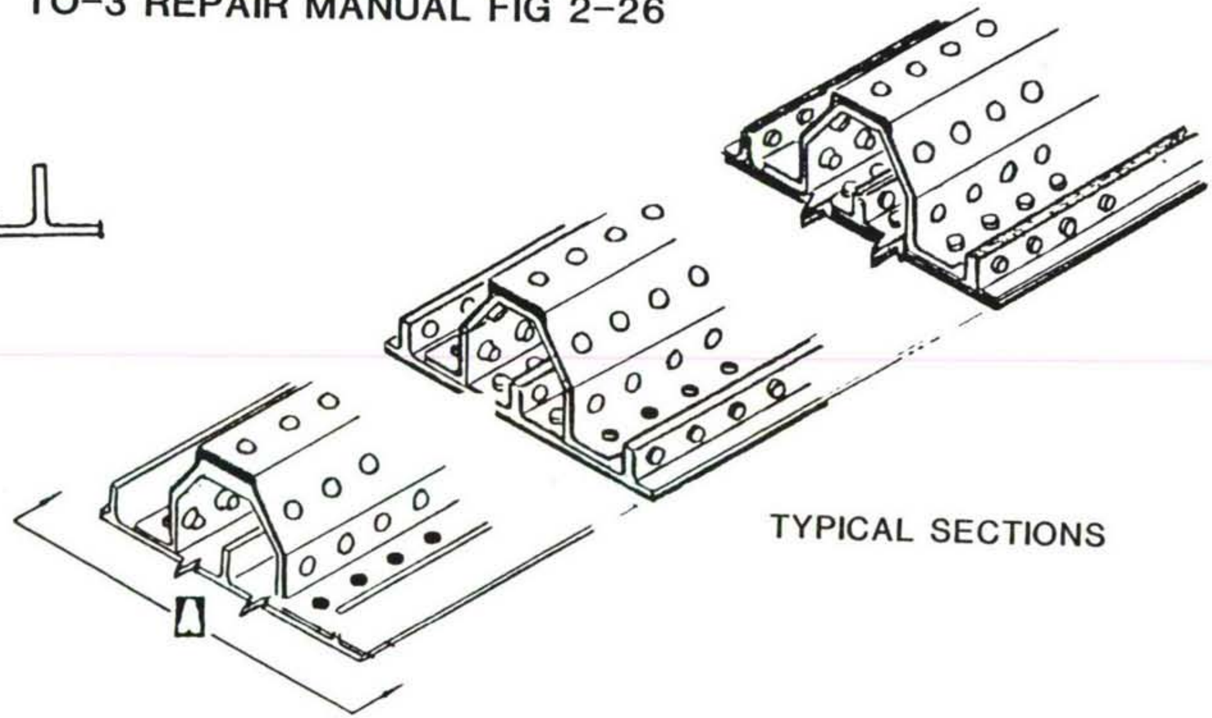
This is a sample T.O.-3 center wing repair selected for analysis. It represents the most extensive center wing compound damage and repair addressed in the C130 TO-3. In this case the damage includes an entire segment of the wing surface: one stringer, three risers, and the adjacent skin all severed.

IMPACT OF WING
MULTIPLE REPAIRS

CENTER WING DAMAGE STANDARD REPAIR
COMPOUND REPAIR - UPPER SURFACE PANEL & STRINGER DAMAGE



TO-3 REPAIR MANUAL FIG 2-26



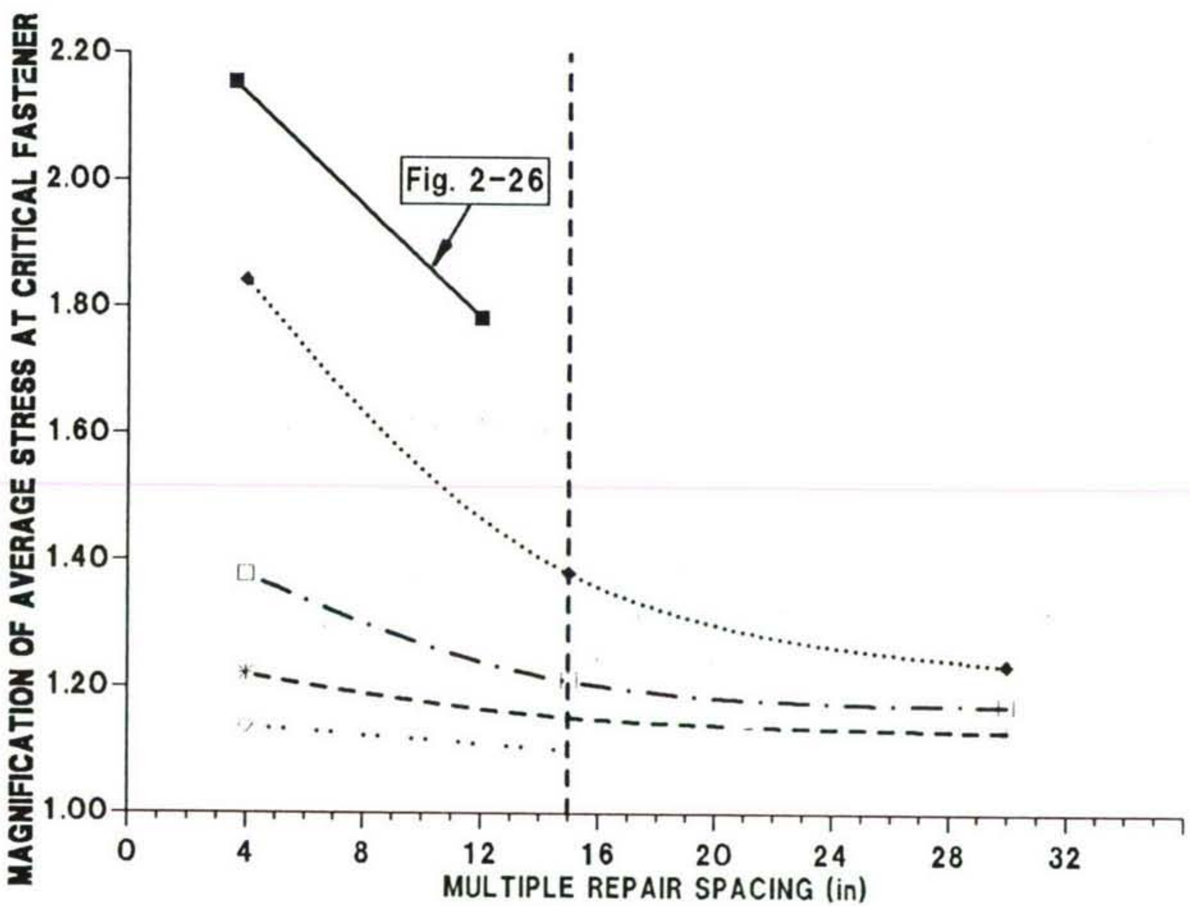
TYPICAL SECTIONS

IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

This plot shows the peak average stress variation (relative to far field stress) with spanwise spacing for each of the T.O.-3 center wing repairs studied.

IMPACT OF WING
MULTIPLE REPAIRS

CENTER WING FEM RESULTS
CRITICAL SKIN STRESS MAGNIFICATION VS. REPAIR SPACING

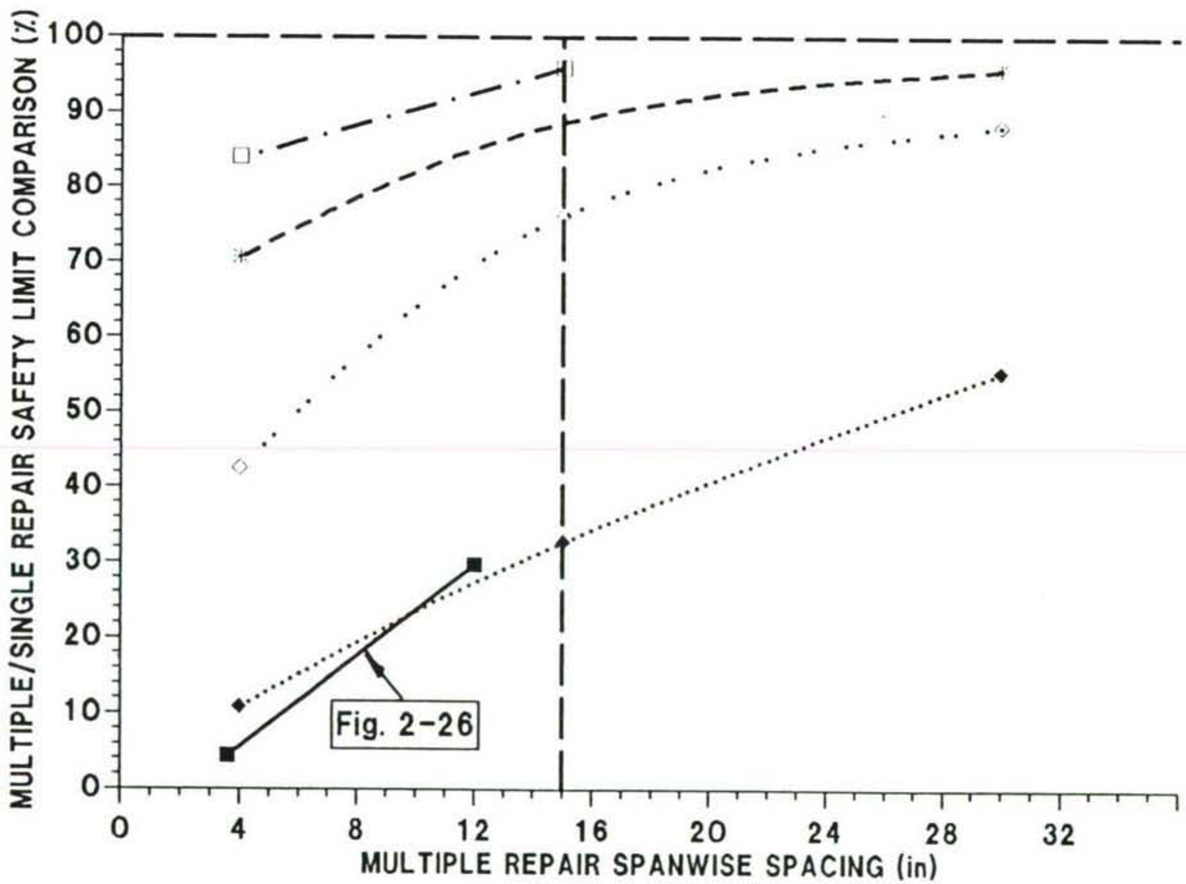


IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

This plot shows the ratio of spanwise multiple repairs safety limit to single repair safety limit for each of the T.O.-3 center wing repairs studied and as a function of multiple repair spacing. Note the significant effect of multiple repairs as the spacing decreases. These decreases can be very important where single repair Safety Limits are already low. Note also the significant reductions in safety limit for several of the repairs at the current common spacing limit of 15 inches.

IMPACT OF WING MULTIPLE REPAIRS

C130-A CENTER WING DTA RESULTS
SAFETY LIMIT CHANGE VS. REPAIR SPANWISE SPACING



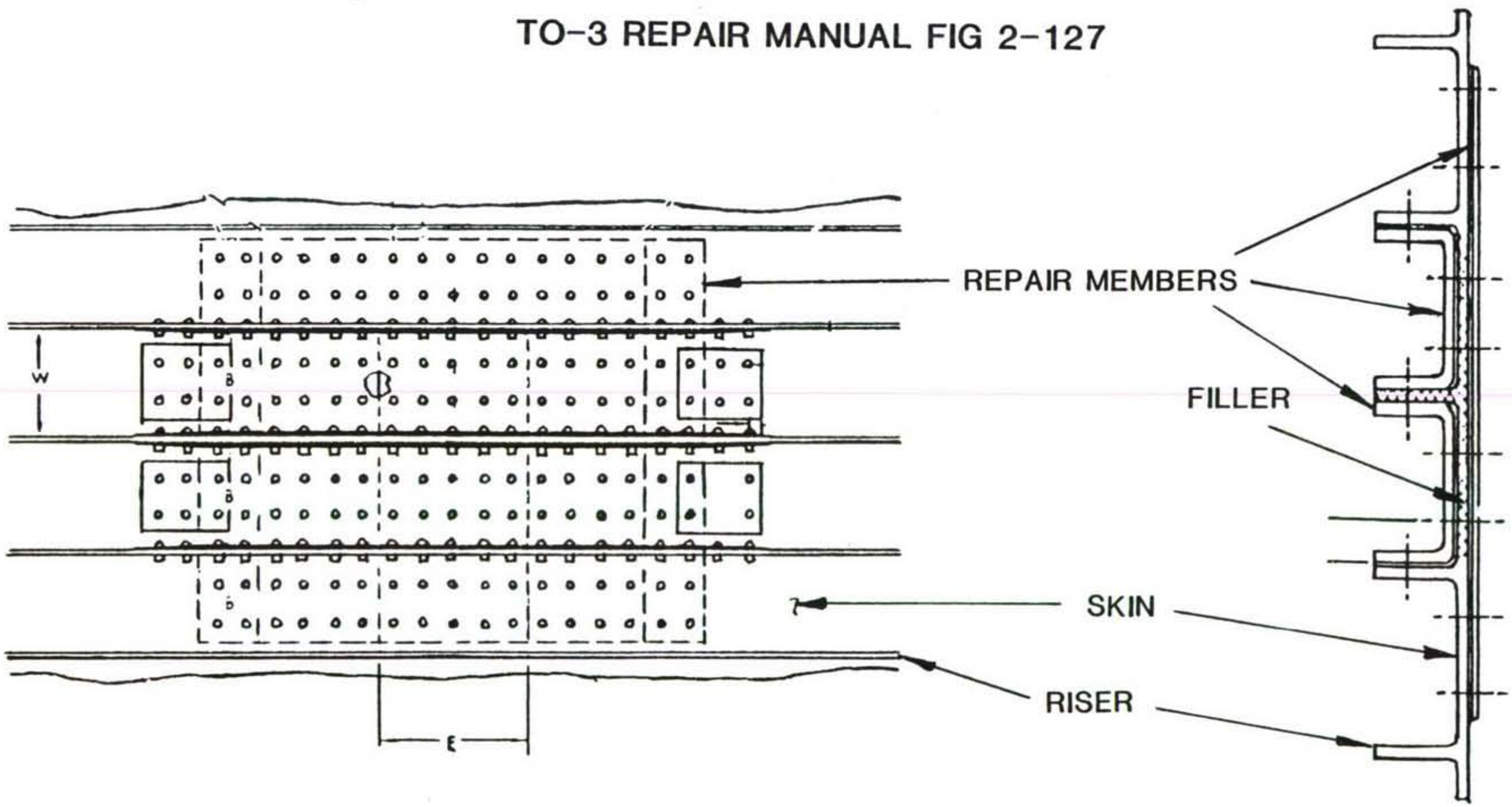
IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

This is a sample T.O.-3 outer wing repair selected for analysis. It represents the most extensive outer wing compound damage and repair addressed in the C130 TO-3. In this case the damage includes an entire segment of the wing surface: one riser and the skin bays on both sides are severed.

IMPACT OF WING
MULTIPLE REPAIRS

OUTER WING DAMAGE STANDARD REPAIR
COMPOUND REPAIR - SURFACE PANEL INTEGRAL SKIN AND RISER DAMAGE

TO-3 REPAIR MANUAL FIG 2-127

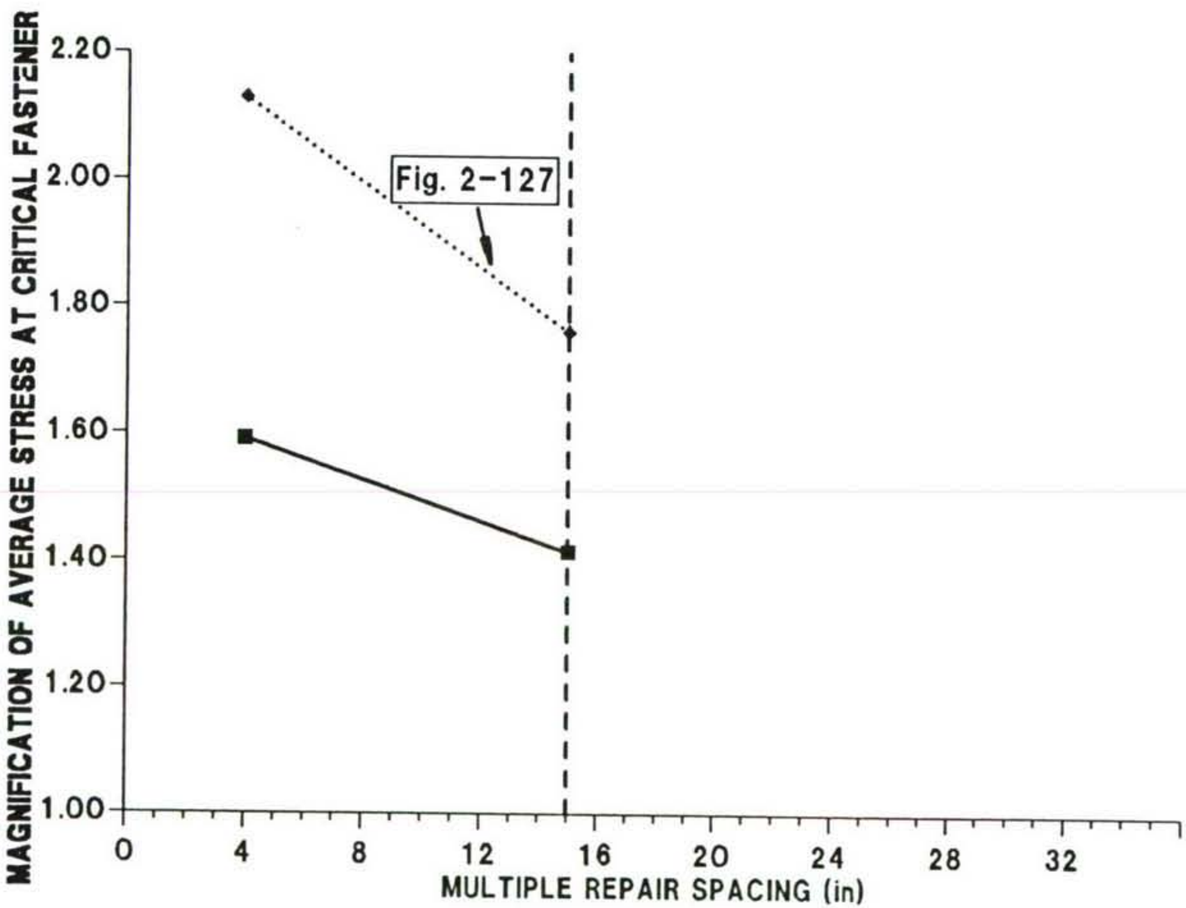


IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

This plot shows the peak average stress variation (relative to far field stress) with spanwise spacing for each of the T.O.-3 outer wing repairs studied.

IMPACT OF WING MULTIPLE REPAIRS

OUTER WING FEM RESULTS
CRITICAL SKIN STRESS MAGNIFICATION VS. REPAIR SPACING

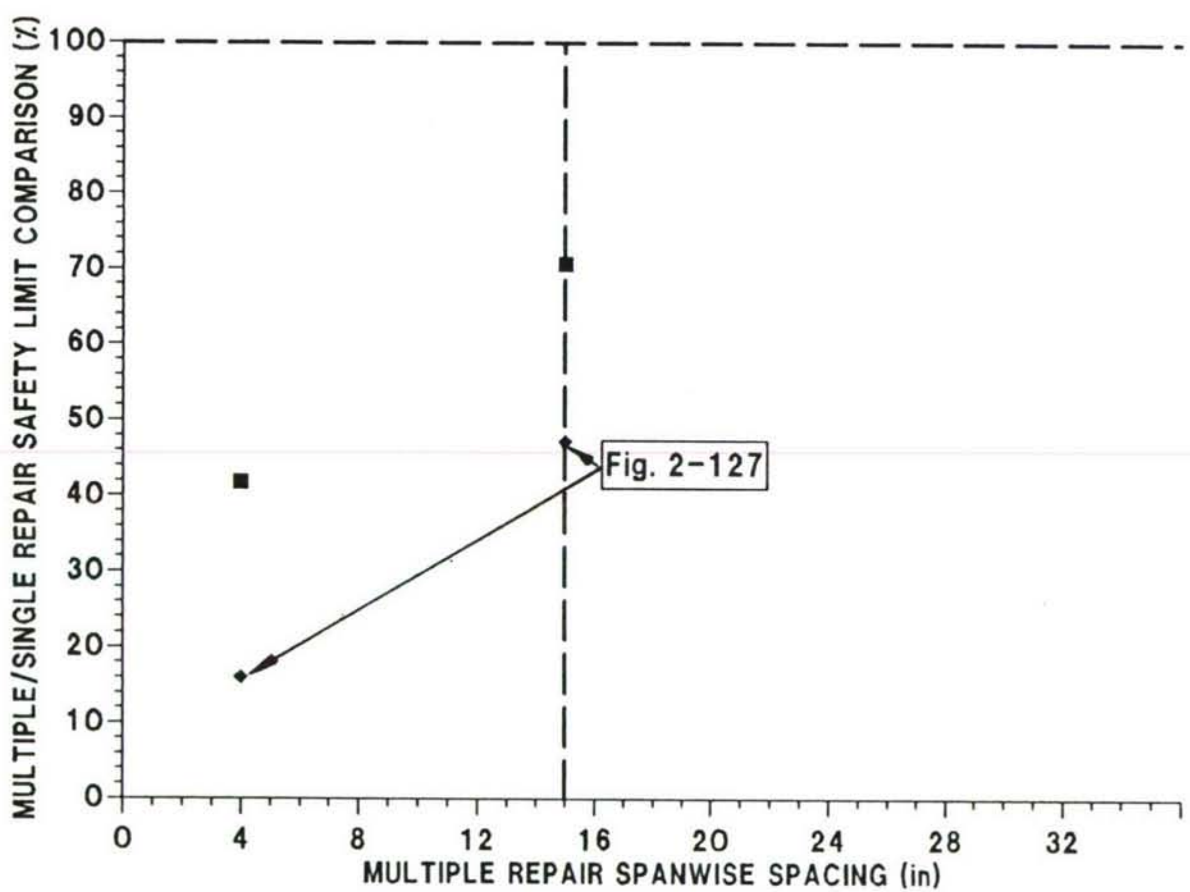


IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

This plot shows the ratio of spanwise multiple repairs safety limit to single repair safety limit for each of the T.O.-3 outer wing repairs studied and as a function of multiple repair spacing. Note the significant effect of multiple repairs as the spacing decreases. These decreases can be very important where single repair Safety Limits are already low.

IMPACT OF WING
MULTIPLE REPAIRS

C130-A OUTER WING DTA RESULTS
SAFETY LIMIT CHANGE VS. REPAIR SPANWISE SPACING

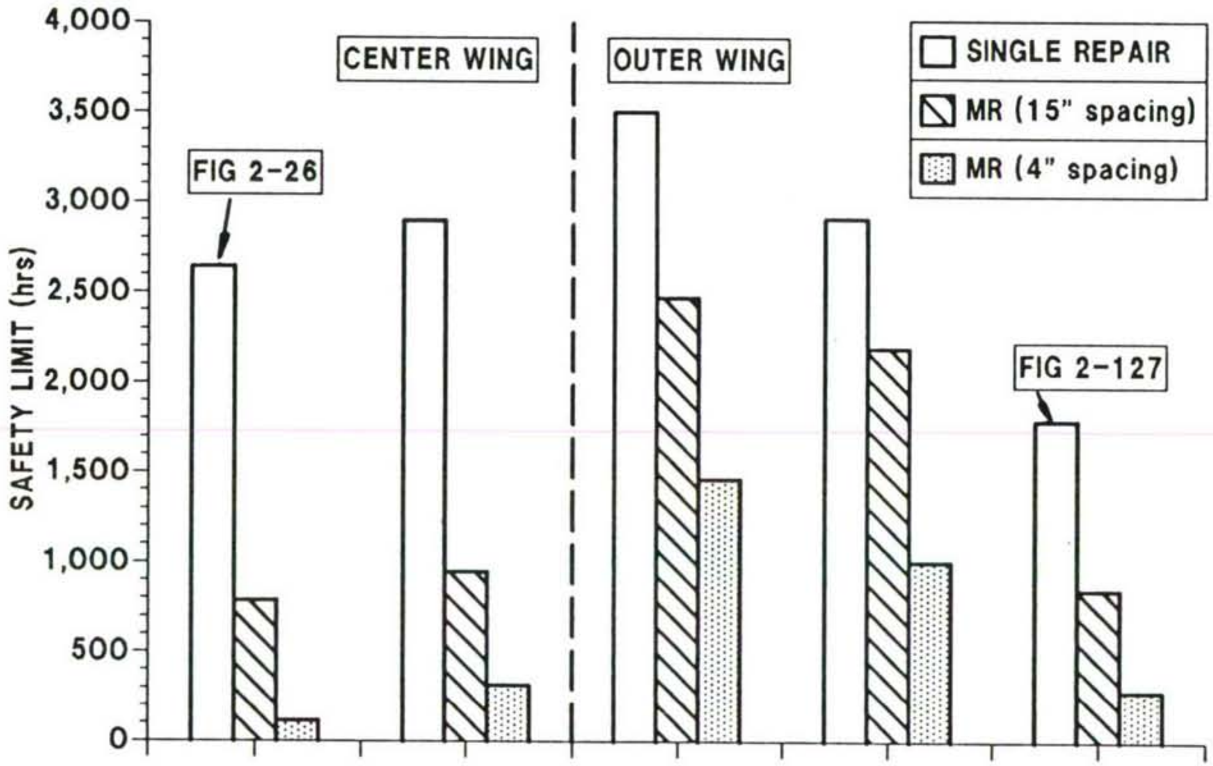


IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

Shown here ae the calculated safety limits for each single and multiple repair configuration analyzed (C-130A models). Even for the single repair configurations, these values are very low compared to the base structure without repairs and imply very short initial inspection intervals. In each case the safety limits are significantly decreased when repairs are placed in tandem spanwise even at the currently common 15″ limit.

IMPACT OF WING
MULTIPLE REPAIRS

SUMMARY OF DAMAGE TOLERANCE ASSESSMENT
C-130A WING REPAIRS SAFETY LIMIT

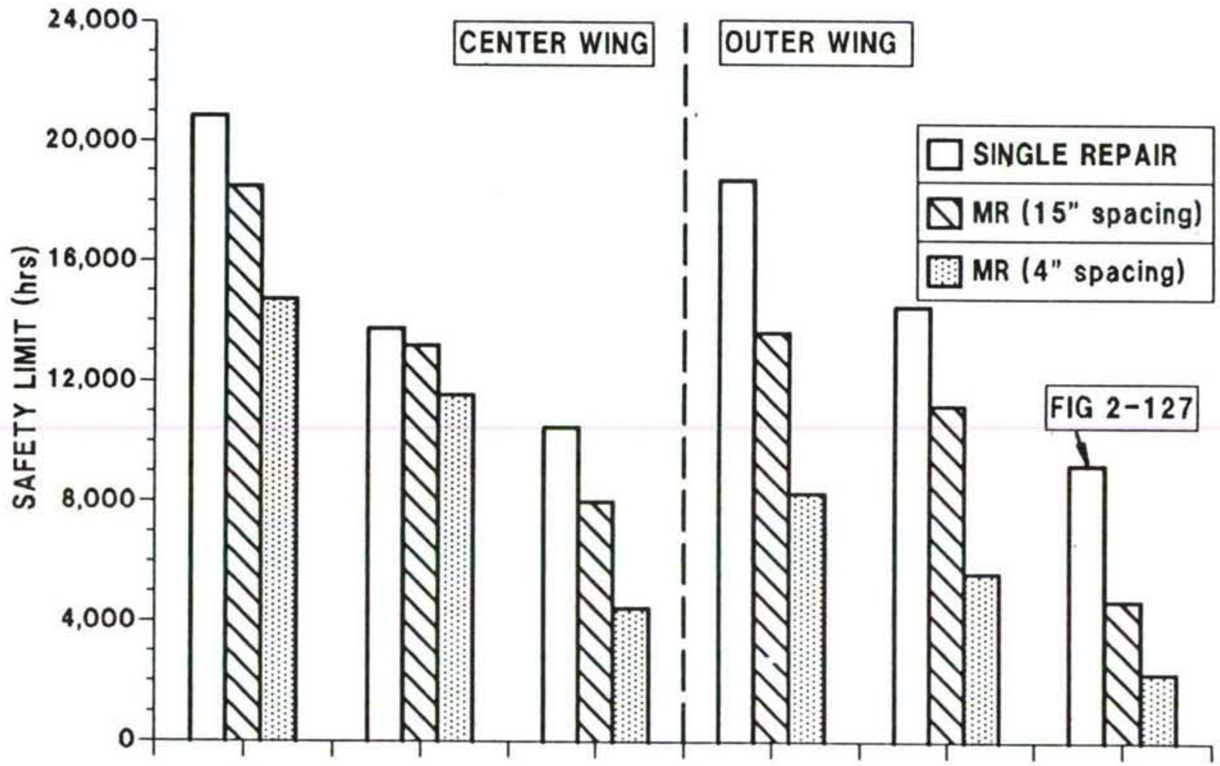


IMPACT OF
WING MULTIPLE REPAIRS
ON DAMAGE TOLERANCE

Shown here are the calculated safety limits for each single and multiple repair configuration analyzed for C-130H models. Again the multiple repair configurations often significantly reduced values compared to single repair. In the worst case, for repair 2-127, multiple repairs placed at 15″ end-to-end spacing reduced the single repair safety limit by almost 50%. (Note that the outer wing repairs are not currently restricted by the T.O.-3 relative to spanwise multiple repairs).

IMPACT OF WING MULTIPLE REPAIRS

SUMMARY OF DAMAGE TOLERANCE ASSESSMENT C-130H WING REPAIRS SAFETY LIMIT



IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

It is evident that multiple repairs placed directly in line spanwise on a structure like the C-130 wing can significantly decrease Damage Tolerance of the base structure. On the other hand, these same repairs may be placed side by side (chordwise) without significantly affecting crackgrowth rates.

As a result, the multiple repair limits currently defined in the C-130 T.O.-3 result in reduced safety limits compared to single repairs.

IMPACT OF WING MULTIPLE REPAIRS

CONCLUSIONS

- MULTIPLE SPANWISE REPAIRS CAN INCREASE CRACKGROWTH RATES COMPARED TO SINGLE REPAIRS
- MULTIPLE CHORDWISE REPAIRS NOT LIKELY TO INCREASE CRACKGROWTH RATES COMPARED TO SINGLE REPAIRS
- MULTIPLE DIAGONAL REPAIRS INCREASE CRACKGROWTH RATES RELATIVE TO SINGLE REPAIRS BUT NOT AS SEVERELY AS SPANWISE
- CURRENT C-130 REPAIR MANUAL MULTIPLE REPAIR SPACING LIMITS RESULT IN REDUCED SAFETY LIMITS WHEN COMPARED TO SINGLE REPAIRS

IMPACT OF WING MULTIPLE REPAIRS ON DAMAGE TOLERANCE

The increase in crack growth rates caused by installation of wing spanwise repairs is a fact that cannot be avoided short of complete replacement of damaged base structure. Therefore, these effects can at best be minimized by increasing the spanwise spacing limits currently defined in the T.O.-3 manual and/or by redesign of the most sensitive repairs. Such a revision should result in greater compatibility and consistency of restrictions between repairs defined in the repair manual.

Since any repair (whether it be a single repair or multiple repairs) can significantly reduce damage tolerance, comprehensive repair tracking and inspection programs are being developed for both standard and non-standard repairs.

As noted earlier, all analysis performed here assumed a mix of identical repairs in multiple repair patterns. Of course, such uniform conditions are likely to be the exception rather than the rule. Since the direct DTA analyses of such non homogeneous multiple repairs is impractical in most cases, a simple conservative approach can be taken to resolve this problem. That is, that for inspection interval definition purposes, the worst case repair, homogeneous results should be used. And further, when an inspection is performed, all repairs within given spacing limits should be inspected.

IMPACT OF WING MULTIPLE REPAIRS

RECOMMENDATIONS

- o INCREASE SPACING LIMIT FOR SPANWISE MULTIPLE REPAIRS IN DAMAGE TOLERANT CRITICAL AREAS
- o REDESIGN STANDARD REPAIRS AS REQUIRED
- o REVISE REPAIR MANUAL FOR GREATER COMPATABILITY BETWEEN REPAIRS
- o INCORPORATE MULTIPLE REPAIR EFFECTS IN CURRENT TRACKING AND INSPECTION PROGRAMS

AN UPDATE ON SOFI:
NAVY'S STRUCTURAL ON-LINE FATIGUE INFORMATION SYSTEM

by

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Abstract

Structural On-line Fatigue Information (SOFI) is a comprehensive system under development by the U.S. Navy to improve structural fatigue tracking for the Navy's older aircraft. SOFI is comprised of three subsystems for usage data: Acquisition, Processing and Dissemination. SOFI's Acquisition Subsystem, the Structural Data Recording Set (SDRS) with multiparameter measurement capability, will replace the outdated and limited g exceedance measuring system currently in use. SOFI's Acquisition and Processing Subsystems will together optimize the record-to-report cycle using state-of-the-art communication hardware, structural fatigue software and relational database software. SOFI's Dissemination Subsystem will complete the loop by providing up-to-date and accurate usage information in a "user-friendly" format. A related paper presented at last year's conference introduced SOFI in terms of its three subsystems.

Components of the Acquisition Subsystem, both hardware and software, have been assembled and developed for the purpose of gathering and transmitting raw data to a Central Processing facility. The Processing Subsystem has been developed to receive the raw data from the Acquisition Subsystem, perform quality control checks, compute Fatigue Life Expended (FLE) or safe life values, and finally, to update the relational database with the processed fatigue life information stored in monthly and cumulative tables. An on-line interface has been developed to provide instant access to the updated data for daily operational and long-range strategic planning. This

along with the hard copy reports published quarterly constitutes the Dissemination Subsystem.

Representative data from recent SDRS flight tests at NAS, Patuxent River, MD, were used to evaluate the three subsystems independently. Results of the evaluation and the program for fielding an operational system are discussed.

Introduction

A successful aircraft fleet management program can be defined as one which makes optimal usage of each of its individual aircraft. An accurate estimate of the structural damage, and therefore, of the remaining structural life of an aircraft is an important factor in determining the effectiveness of the Navy aircraft fleet management system. The Navy recognizes this fact and has implemented a successful aircraft life management program to monitor and manage its aircraft fleet (Figure 1). Among the foremost objectives are the safety of the crew, structural integrity of the aircraft and operational readiness of the fleet. An effective life management program also ensures the availability of a variety of planning options, principally flexibility with respect to establishing force levels and mixes.

In order to achieve these objectives, the Navy's approach can be categorized as twofold in that (Figure 2):

a. Structural life limits, in terms of flight hours, catapults, arrestments, etc., are established for all fatigue critical components and locations of the airframe for every T/M/S aircraft.

b. Every individual aircraft and its components are tracked to ensure their usage within the established life limits.

The structural life limits are determined by extensively fatigue testing the full-scale aircraft, its major assemblies, components, elements and coupons. For all of the Navy's fixed-wing aircraft, the structural life limits are published by NAVAIR through Reference 1.

The above approach to Navy's aircraft life management is implemented through NAVAIR's Aircraft Structural Life Surveillance (ASLS) program (Figure 3). The program is comprised of three major life cycle phases: continuous tracking (SAFE), timely assessment (SLAP) and necessary extension (SLEP). More details on each of these subsystems can be found in Reference 2. In a major endeavor to upgrade individual aircraft structural tracking, the Navy has initiated the SOFI system under its ASLS program. In the early stages, SOFI development will be carried out under SAFE, but eventually when all aircraft are inducted into SOFI, its will replace SAFE in its entirety.

SOFI has been described as an information system designed to gather, process and channel the flow of structural fatigue-related information on Navy aircraft to the users, owners, planners and maintainers. From the standpoint of functional clarity, SOFI has been divided into three distinct but interfacing subsystems (Figure 4):

Acquisition Subsystem
Processing Subsystem
Dissemination Subsystem

The Acquisition Subsystem pertains to gathering of structural usage data from all Navy aircraft and encompasses various onboard recording systems. The Navy is currently transitioning to multiparameter onboard recording and computer-based ground stations for all their aircraft, replacing outdated load factor exceedance counters. The Processing Subsystem deals with analyzing the usage data obtained via the Acquisition Subsystem. The processing software is being developed as a common algorithm with modular interfaces to accommodate the data format from the various recording systems of the Acquisition Subsystem. The Dissemination Subsystem makes the information from the Processing Subsystem readily and conveniently available to the authorized activities. An on-line, random access graphic software package is being designed and developed for this purpose.

The principal objectives of SOFI are listed in Figure 5. As stated above, the essence of SOFI is upgrading the fatigue life/usage tracking

methodology for each individual Navy aircraft. In addition to improving the accuracy of NAVAIR's ASLS Program by incorporating changes such as multiparameter tracking instead of existing single parameter tracking, etc., SOFI will also reduce the record-to-report cycle time. SOFI will eventually include the fatigue/usage tracking of all Navy aircraft and components. Finally, SOFI is designed to use state-of-the-art fatigue prediction methods and to have the capability for easy incorporation of future technological improvements. The approach adopted for meeting the above listed SOFI objectives is highlighted in Figure 6.

Figure 7 pictorially presents the flow of information through the three SOFI subsystems. The Acquisition Subsystem consists of both ground-based and onboard systems. The aircraft usage data are obtained from an onboard recorder via a ground station unit, whereas the configuration data (that is, information regarding structural modifications or component interchanges) originate with the maintenance personnel and are detailed from organic or depot maintenance sites. The Processing Subsystem maintains a log of all data received, performs quality control checks on the data, computes the fatigue life expended for all critical components and then updates the database. The Dissemination Subsystem makes all processed information available to fleet activities via an on-line computer network and hard copy reports.

<u>SOFI Elements Status</u>

The status of the various elements of SOFI in terms of its three subsystems is shown in Figure 8. The Acquisition Subsystem, which includes the recording systems from various aircraft types, e.g., F-18, F-14, etc., is currently functional. A new multiparameter recording system consisting of the airborne Structural Data Recording Set (SDRS) and a ground station Recorder Reproducer (RR) for downloading and transmitting the recorded data has been specifically designed and developed for SOFI. The SDRS/RR system has been successfully flight tested and qualified and is currently functional for the A-6 aircraft. Details about methodology for processing and dissemination of data have been worked out and comprehensive computer software codes have been generated as part of the Processing and Dissemination Subsystem development. Detailed updates on each of the three subsystems are given below.

Acquisition Subsystem

An overview of the Acquisition Subsystem is shown in Figure 9. The Acquisition Subsystem covers both aircraft usage data and aircraft configuration data. As mentioned earlier, this subsystem includes the recording systems from various aircraft types but this paper will focus only on the SDRS/RR based system developed specifically for SOFI.

The system is comprised of an airborne unit and a ground based Recorder Reproducer. The airborne unit consists of a recorder converter, motional transducer and memory unit (Figure 10). The Recorder Converter (RC) processes all the signals monitored by the motional transducer and other instruments, such as strain gages, and stores them in the Memory Unit (MU) based on a predetermined recording algorithm. Data from the memory unit are then downloaded onto a 3 1/2" diskette using the recorder reproducer. A detailed description of the SDRS-based system can be found in Reference 2. Figure 11 shows an RC/MU unit installed on an A-6E aircraft.

Some of the salient features of the individual units of the SDRS system are outlined in Figures 12 through 14. The recorder converter is a generic multiparameter instrument with capability to record 12 analog channels, eight discrete triggers, and elapsed time in addition to interfacing with the two pressure transducers for airspeed and altitude. The recorder gives the user flexibility to choose from any of the three predetermined recording algorithms, viz., peak/valley, usage matrix, and fixed band. The recorder can sample at a maximum rate of 512 samples/sec. with capability to switch sampling configuration based on certain predetermined flight conditions, e.g., sample longitudinal acceleration at a higher rate during landing. The motional pick-up transducer is a high accuracy 3-axis servo instrument for measuring the acceleration (N_z, N_x, and roll) signals. The memory unit is easily accessible and portable, has a non-volatile memory, and has the capacity to store aircraft usage data for about a month. The memory unit stores information on aircraft identification and maintains cumulative counts of the total number of flights, catapults and arrestments on the airframe. The memory unit has indicators for system malfunction and memory unit full.

The recorder reproducer is an off-the-shelf Zenith lap-top computer with an interface for the memory units. The reproducer downloads data from the memory unit onto 3 1/2" disks and also interprets data for maintenance actions, such as the need for airframe over-stress inspections. A unique file name containing the aircraft Bureau Number and a five digit Julian data is automatically created during data downloading. Additionally, the reproducer is also used for maintaining backup data on the hard disk and for reconfiguring the recorder converter.

SDRS system installations are currently under way for the A-6E aircraft with two aircraft already operational (Figure 15). The data collected to date from these two aircraft totalling 127 flights and 154 hours are very consistent and meet all expectations. The measured analog channels include N_z, N_x, roll acceleration, roll rate, Mach number, altitude and two strain channels. The monitored discrete records are the flap up/down, gear up/down, hook up/down, and weight-on/off-wheels. Some problems were realized early in the program with respect to missed catapults and arrestments and the weight-on/off-wheel switch activation. These problems have since been traced to inappropriate threshold settings and corrective remedial actions have already been successfully implemented.

A system has been designed to obtain aircraft configuration data (Figure 16) consisting of information on the remove/install records of all life limited, interchangeable aircraft components. This new system is to replace or augment the existing laborious form-based configuration tracking. The new configuration tracking system will be a PC-based on-line "logbook" system connected by a network of PCs and operates on a very user-friendly menu-driven software.

Processing Subsystem

The main items of the Processing Subsystem are listed in Figure 17. The rationale and the software for these items are in various phases of development with a basic working model already in place. Details on each of these items of the Processing Subsystem are given in Figures 18 through 23.

The usage data can be received either via mail on 3 1/2" disks or directly by modem, whereas the configuration data are received through an on-line PC-based network. A computerized log of all data receipts is maintained with the capability to automatically generate messages for missing data. A batch procedure has been developed to convert the data to engineering units and upload to the mainframe. A sample of the data file is shown in Figure 19. Quality control checks (Figure 20) are performed on the data and all errors are classified as salvageable/unsalvageable. The first step of quality control is performed on the data file header information which contains information on cumulative flight number, catapults, arrestments, etc. The remaining quality checks can be classified as data validity checks and data consistency checks. While the data validity checks are with respect to the established recording criteria such as data bounds, rise-fall/deadband criteria, etc., the data consistency checks focus on checking the average of the recorded values against expected/established trends. The quality control checks also monitor possible drift in strain gage zero values.

Even though the planning and design of the SDRS/RR system have been done with reasonable care and foresight, it is realized that occasionally some of the data will be lost due to a temporary malfunction of the system. A gap-fill procedure has been defined to address such situations (Figure 21). The procedure includes developing a multiparameter regression equation for strains and usage statistics based on groupings of fleet, squadron, and individual aircraft.

Incremental fatigue damage is calculated following quality control and gap-filling (Figure 22). A sequence accountable fatigue algorithm, incorporating the Navy's philosophy of damage tracking to fatigue initiation, has been developed for this purpose. This general purpose program can track damage to multiple locations and can be used for processing N_z generated, N_z actual or actual recorded strain spectra. In each case, the history of the airframe, in terms of residual stress, is accounted for in calculating the fatigue damage. The final step of the Processing Subsystem involves archiving the data in a readily retrievable form and preparing/updating the database to be used for information dissemination purposes (Figure 23).

Dissemination Subsystem

An on-line system has been designed to disseminate aircraft pertinent information to all the designated users in a fast, reliable and efficient manner (Figure 24). The Dissemination Subsystem addresses all aircraft-related information such as the fatigue life expended, configuration data, usage data, component service life limits, technical directives issued, etc. The users of the Dissemination Subsystem are designated by NAVAIR and will include squadrons, functional wings, administrative and supply activities, etc. An important feature of the Dissemination Subsystem is its capability to limit the accessibility of any user and to set up different accessibility hierarchy levels based on the user. The hardware for the system is comprised of a central computing center connected to the various squadrons with a network of PCs. The software for the system is based on a relational database and an on-line graphic interface. A sample of the graphic screen is shown in Figure 25. As can be seen from the figure, the software has random access capability and one can easily retrieve the desired information by simply clicking a mouse on the available menus on the screen.

SOFI Future Efforts

The updates presented in this paper are evidence of considerable progress that has been made in the design, development and implementation of the SOFI system. However, as with any new system, especially one of this magnitude, there still exists a few loose ends to be tied and a few new features to be added. A list of foreseen future activities is given in Figure 26. While the SDRS system is currently in operation on only A-6E aircraft, it will soon be installed in other aircraft like the F-14, E-2, C-2, etc. The SDRS system will also be adapted for use with rotary wing aircraft such as the H-46, AH-1W, etc. The software of the recorder reproducer will be upgraded to provide preliminary usage information during data downloading by the squadron.

Concluding Remarks

This paper has presented an update on Navy's SOFI system in terms of its background, objectives and status (Figure 27). It is seen that the various elements of SOFI have each been successfully developed, tested and implemented and are currently in various phases of upgrading. The future effort is mainly towards integrating all these subsystems and all the various Navy aircraft into one centralized system.

Acknowledgments

All aspects of SOFI are under the sponsorship of the Structures Branch (AIR-5302) of the Naval Air Systems Command in Washington, D.C.

Reference

1. NAVAIR Instruction 13120.1B, "Fixed Wing Aircraft Structural Life Limits," published by Commander, Naval Air Systems Command, Washington, D.C.

2. "SOFI: Navy's Structural On-line Fatigue Information System," Elchuri, V., et al., Proceedings of the 1988 Structural Integrity Program Conference, editors T. D. Cooper and J. W. Lincoln, Wright-Patterson AFB, OH, May 1989.

FIGURE 1

NAVAIR APPROACH

ESTABLISH LIMITS

TESTING

- FATIGUE TESTING DEMONSTRATES SPEC LIFE

ENSURE LIMITS

TRACKING

- FLEET USAGE TRACKING ENSURES SAFE STRUCTURAL LIFE LIMITS ARE NOT EXCEEDED

FIGURE 2

NAVY'S AIRCRAFT LIFE MANAGEMENT PROGRAM

NAVAIR OBJECTIVES

- ENSURE FLIGHT SAFETY - CREW & AIRCRAFT
- ENSURE OPERATIONAL READINESS
- PRESERVE OPNAV PLANNING OPTIONS
- ENSURE OPNAV FORCE LEVEL/MIX FLEXIBILITY
- ASSESS FUTURE NEEDS

NAVAIR APPROACH

- ESTABLISH STRUCTURAL LIFE LIMITS - AIRFRAME & COMPONENTS
- ENSURE USAGE AGAINST THESE LIMITS

FIGURE 3

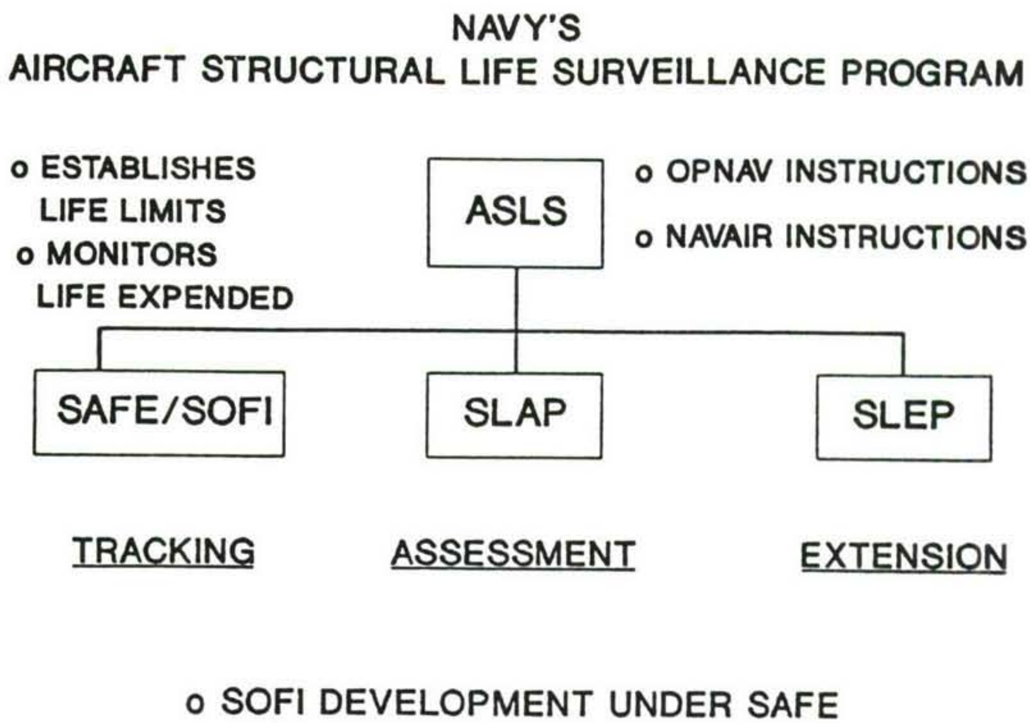

FIGURE 4

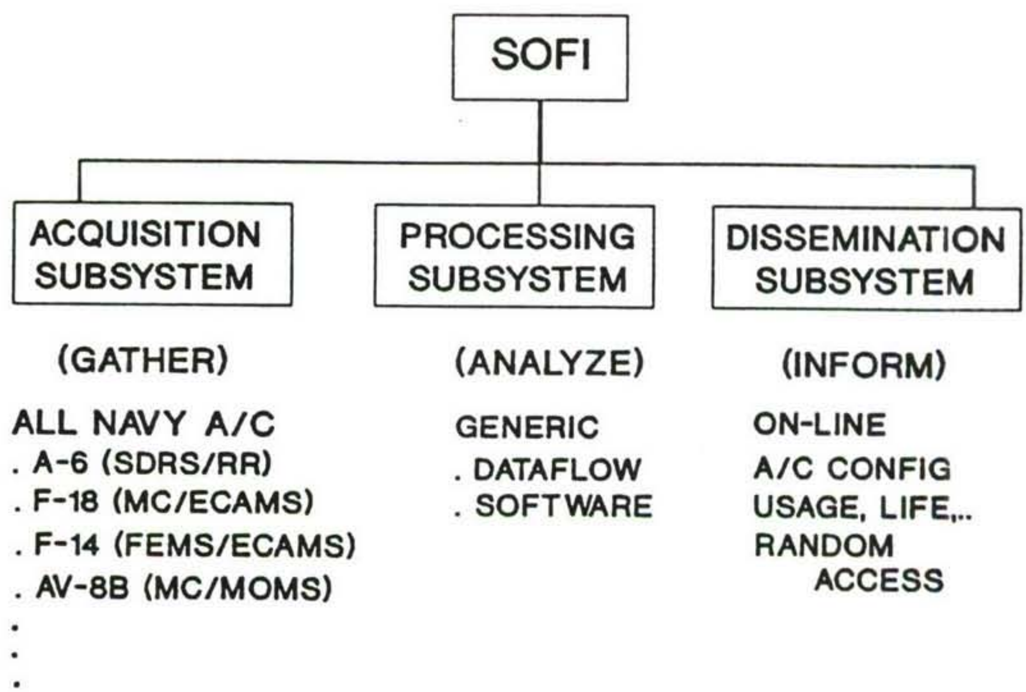

FIGURE 5

SOFI OBJECTIVES

- IMPROVED ACCURACY
- REDUCED RECORD-TO-REPORT CYCLE TIME
- ALL NAVY AIRFRAME AND COMPONENT FATIGUE TRACKING
- STATE-OF-THE-ART FATIGUE PREDICTIONS

FIGURE 6

SOFI APPROACH

- IMPROVED ACCURACY
  - . SINGLE PARAMETER --> MULTI-PARAMETER
  - . LOGBOOK LNDGS/CATS --> ON-BOARD RECORDER
  - . Nz EXCEEDANCES --> Nz/STRAIN CYCLES
  - . CONSERVATIVE FLEs --> REALISTIC FLEs
- REDUCED RECORD-TO-REPORT TIME
  - DATA TRANSMISSION
    - . MESSAGES, COMTRAK --> ONLINE NETWORK
    - . POSTCARDS, MAIL --> DISK, MODEM, MAIL
  - DATA PROCESSING
    - . LABOR INTENSIVE --> AUTOMATED
  - DATA REPORTING
    - . HARD COPY --> HARD COPY, ONLINE
- ALL AIRFRAME AND COMPONENT FATIGUE TRACKING
  - . SDRS FOR A/C WITH CAG
  - . SDRS FOR A/C WITH NO SYSTEM
  - . AUTOMATED CONFIGURATION TRACKING
- STATE-OF-THE-ART FATIGUE PREDICTION
  - SEQUENCE ACCOUNTABLE FATIGUE INITIATION ALGORITHM
  - COMMONALITY
    - . FATIGUE ALGORITHM
    - . GAP-FILL PROCEDURE
  - MODULARITY
    - . FACILITATES TECHNOLOGY UPDATES

FIGURE 7

SOFI SYSTEM UPDATES

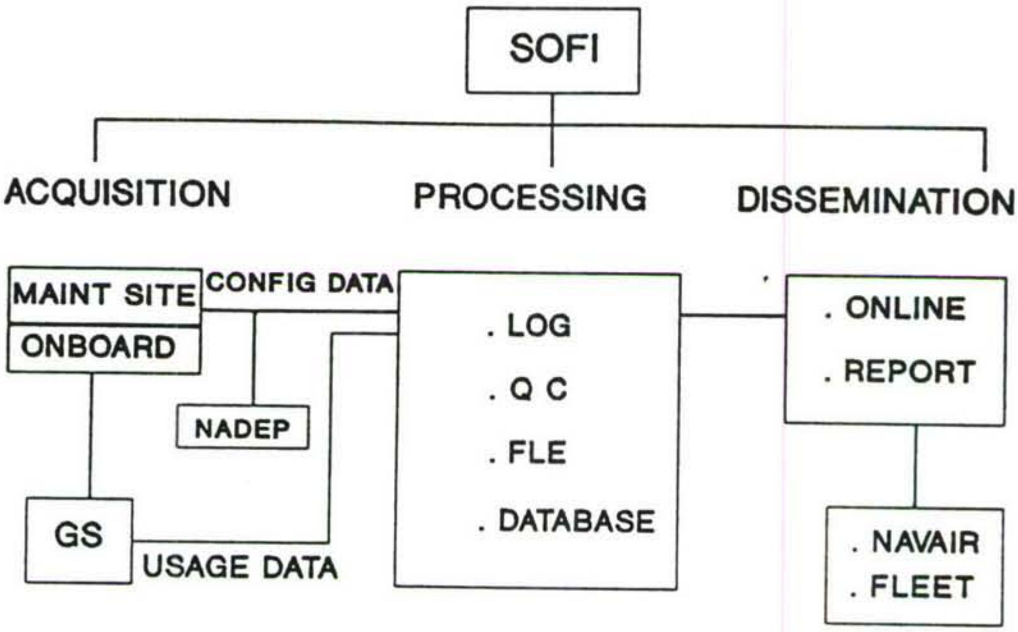

FIGURE 8

SOFI ELEMENTS STATUS

- ACQUISITION SUBSYSTEM
  - F/A-18 (MC/ECAMS) FUNCTIONAL
  - AV-8B (MC/MOMS) UNDER DEVELOPMENT
  - F-14 (FEMS/ECAMS) FUNCTIONAL
  - A-6 (SDRS/RR)
    - SDRS FLIGHT TESTED, QUALIFIED, IN-PRODUCTION
    - FUNCTIONING UNITS IN A-6 AIRCRAFT
  - COUNTING ACCELEROMETER GROUP FUNCTIONAL
    - SDRS SLATED TO REPLACE CAG
- PROCESSING SUBSYSTEM
  - AUTOMATED LOG
  - QUALITY CONTROL SOFTWARE
  - FATIGUE COMPUTATION
  - AUTOMATED MESSAGE GENERATION
  - GAP FILLING
- DISSEMINATON SUBSYSTEM
  - PC-BASED NETWORK
  - RANDOM ACCESS SOFTWARE

FIGURE 9

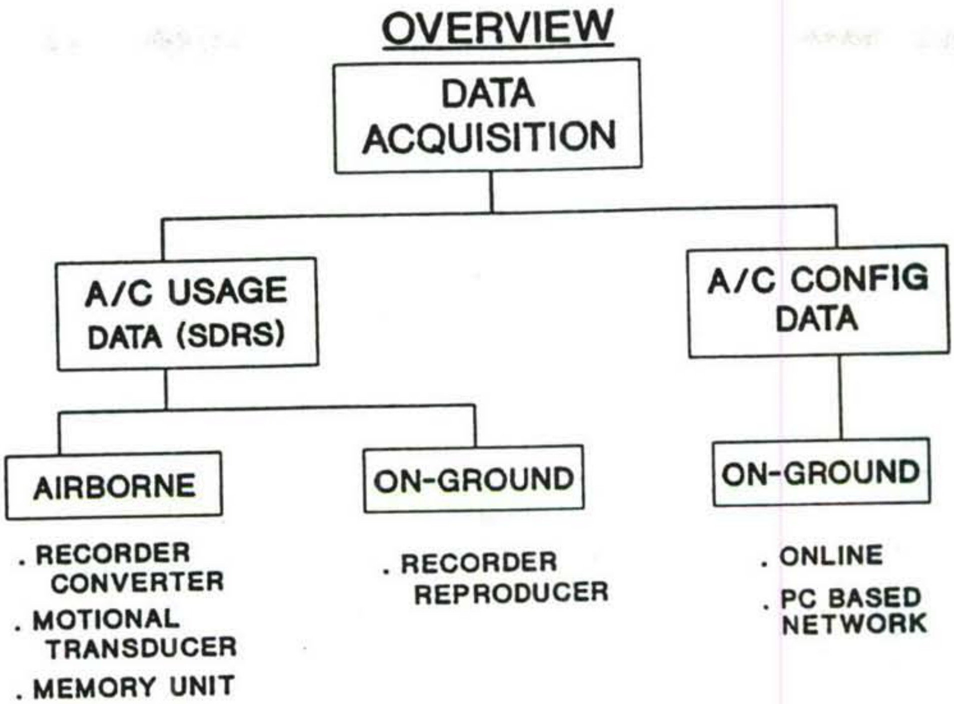

FIGURE 10

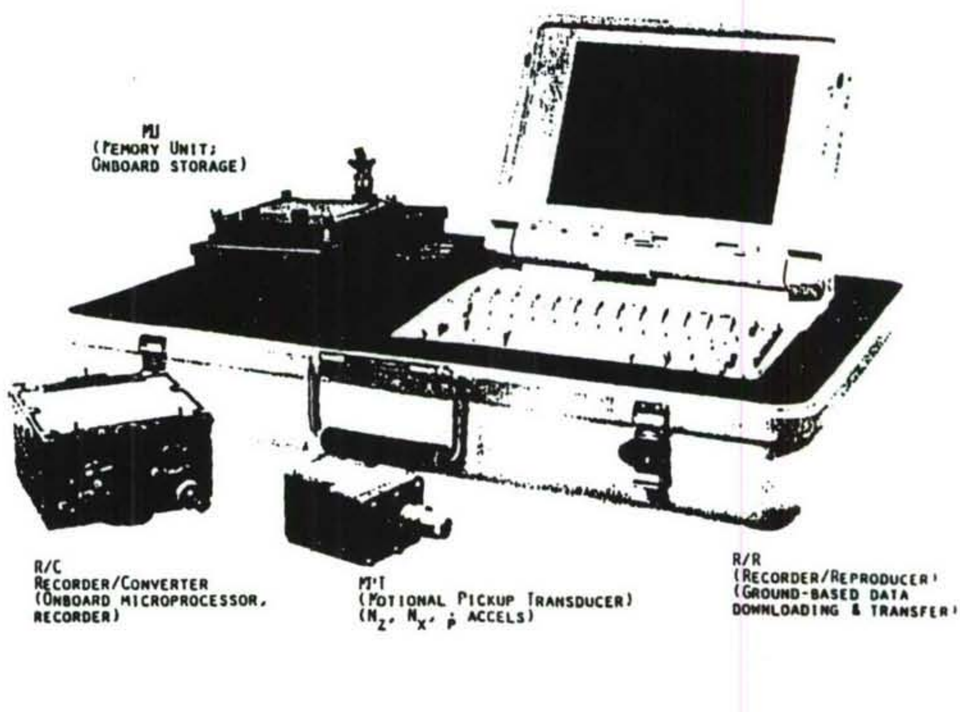

FIGURE 11

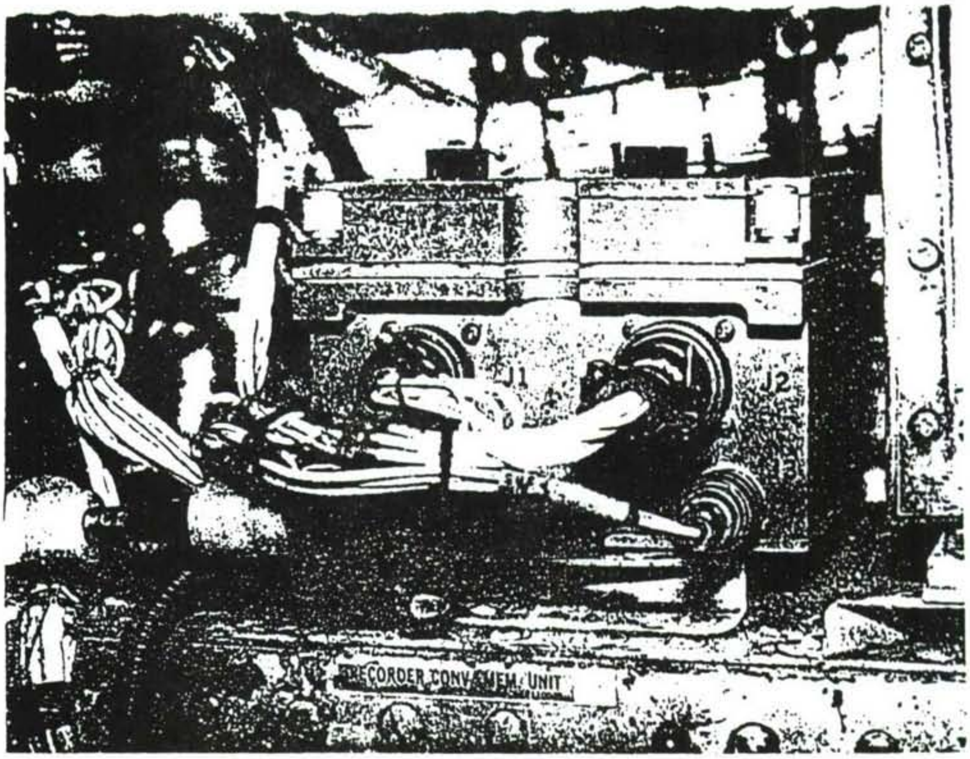

FIGURE 12

. <u>RECORDER CONVERTER, RO-601/ASH-37</u>

- **SIGNAL CONDITIONING AND MULTIPLEXING**
  - . 12 ANALOG CHANNELS
  - . 8 DISCRETE TRIGGERS
  - . ELAPSED TIME
- **TWO PRESSURE TRANSDUCERS**
  - . AIRSPEED
  - . ALTITUDE
- **THREE RECORDING ALGORITHMS**
  - . PEAK/VALLEY
  - . USAGE MATRIX
  - . FIXED BAND
- **SAMPLING RATE OF 512 SAMPLES/SEC**
- **POWER REQUIREMENTS : 28 VDC @ 10 WATTS**

FIGURE 13

MOTIONAL PICKUP TRANSDUCER/MEMORY UNIT

MOTIONAL PICKUP TRANSDUCER (MPT) TR-354/ASH-37

- 3 AXIS SERVO TYPE ACCELEROMETER (Nz, Nx, AND ROLL ACCELERATION)
- ACCURACY 0.5% FULL SCALE TYPICAL

MEMORY UNIT (MU) MU-983/ASH-37

- REMOVABLE AND NON VOLATILE
- 100 K BYTES (EXPANDABLE TO 200 K BYTES)
- BIT AND MEMORY USAGE INDICATORS
- AIRCRAFT IDENTIFICATION
- CUMULATIVE COUNTS OF FLTs, CATs, TRAPs

FIGURE 14

. RECORDER REPRODUCER, RD-608/ASH-37

- DATA DOWNLOAD FROM MEMORY UNIT
- INTERPRETS DATA FOR MAINTENANCE ACTIONS
- DISPLAY DATA TRANSFER GOOD/BAD
- INDICATE MAX Nz, NO OF CATs AND TRAPs
- AUTOMATIC FILE NAMING
- CONFIGURE RECORDER CONVERTER
- BACKUP DATA ON HARD DISK

FIGURE 15

EXAMPLE (A-6 FLIGHT DATA)

- MEASURED PARAMETERS
  - . Nz, Nx, P., P, STRAIN1, STRAIN2
  - . MACH NO, ALT, FLAP UP/DOWN
  - . GEAR UP/DOWN,HOOK UP/DOWN, CATS, TRAPS
- DATA RECORDED BUNOS 152945/155703
  - . HOURS 154
  - . FLIGHTS 127
  - . CATS 55
  - . TRAPS 55
- PROBLEMS TO DATE
  - . MISSED CATS/TRAPS
  - . WEIGHT-ON-WHEEL SWITCH ACTIVATION

FIGURE 16

AIRCRAFT CONFIGURATION DATA

- RECORD OF LIFE LIMITED, INTERCHANGEABLE COMPONENTS
- REPLACE/AUGMENT CURRENT PAPER-BASED CONFIGURATION TRACKING
- PC-BASED ONLINE SYSTEM
- MENU DRIVEN SOFTWARE

FIGURE 17

PROCESSING SUBSYSTEM

- DATA RECEIPT/LOG
- QUALITY ASSURANCE
- GAP FILLING
- FATIGUE COMPUTATIONS
- ARCHIVING / DISSEMINATION DATABASE

FIGURE 18

DATA RECEIPT/LOG

- MAIL, MODEM, CONFIG DATA ONLINE
- COMPUTERIZED LOG
  . AUTOMATIC MESSAGE GENERATION
- DISKETTE COPY AND RECYCLE
- ENGINEERING UNITS
- DATA UPLOADING TO MAINFRAME
- SAMPLE DATA FILE

FIGURE 19

Recorder Serial Number	>	103
Configuration Number	>	0
Flight Number	>	124
Recorder Serial Number	>	103
Configuration Number	>	0
Number of Cats	>	2
Number of Traps	>	2
Maximum Nz	>	5.07
BIT Status	>	00000800000000
Bureau Number	>	155703
Naval Air Station	>	VA42
Total Flight Time	>	04969:657
Data File Name	>	3c538920.200
RR Serial Number	>	102
Program Version Number	>	1
Date of Printout	>	09/14/1989

#####	Time	Accelerations			Mach	Strain Gages			Roll		
	SSSSS:mmm	Nz	Nx	Roll	Number	1	2	ALT.	Rate	Temp	DS
00001	00000:000 R	-2.66	-3.82	9.40	-0.578	-457	-572	14366	-362.50	41	FC
00002	00000:000 Z	0.99	-0.09	0.10	0.446	762	648	-501	-71.88	41	FC
00003	00000:000	1.09	-0.29	0.70	0.183	1600	1296	-533	9.38	51	D F6
00004	00009:993	1.09	-0.29	0.10	0.203	1715	1372	-371	-3.13	51	D F7
00005	00099:661	1.19	-0.29	0.10	0.274	1677	1258	1991	0.00	51	D F5
00006	00168:094	1.58	-0.39	0.10	0.335	P 1982	1524	4986	0.00	51	F5
00007	00168:098	1.58	-0.39	0.10	0.335	1982	P 1524	1986	0.00	51	F5
00008	00174:313	0.50	-0.19	0.10	0.335	V 1372	915	5254	6.25	51	F5
00009	00174:317	0.50	-0.19	0.10	0.335	1372	V 915	5254	6.25	51	F5

FIGURE 20

QUALITY ASSURANCE

- DATA QUALITY CONTROL CHECKS
  - o ERRORS
    - . SALVAGEABLE/UNSALVAGEABLE
  - o HEADER
    - . FLT NO, CATS, TRAPS, BIT, BUNO, etc.
  - o DATA VALIDITY
    - . DATA BOUNDS
    - . RISE-FALL/DEADBAND CRITERIA
  - o DATA CONSISTENCY
    - . EXCEEDANCE VALUES
    - . NO OF Nz AND STRAIN CYCLES
    - . DATA FROM NAVFLIR
- STRAIN GAGE ZERO STUDY

FIGURE 21

GAP-FILLING

- REGRESSION ANALYSIS FOR STRAINS
  - . MULTIPLE PARAMETERS
  - . INDICATOR OF STRAIN GAGE PERFORMANCE
  - . USEFUL FOR ESTIMATING MISSING DATA
- USAGE STATISTICS
  - . FLEET, SQUADRON
- AUTOMATIC MESSAGE GENERATION

FIGURE 22

FATIGUE COMPUTATIONS

- CRACK INITIATION
- SEQUENCE ACCOUNTABLE ALGORITHM
- MULTI-LOCATION
- MULTIPLE APPROACH FOR SMOOTH TRANSITION
  - . Nz GENERATED SEQUENCE
  - . Nz ACTUAL SEQUENCE
  - . STRAIN SEQUENCE
- SPECTRUM GENERATION
- PREVIOUS HISTORY ACCOUNTED FOR

FIGURE 23

. ARCHIVING/DISSEMINATION DATABASE

- DATA ARCHIVING
  - . PROCESSED
  - . RAW
  - . GAP FILLED
- PREPARE/UPDATE DATABASE FOR DISSEMINATION

FIGURE 24

DISSEMINATION SUBSYSTEM

- ONLINE UP-TO-DATE INFORMATION ON
  - . FATIGUE LIFE EXPENDED
  - . CONFIGURATION DATA
  - . USAGE DATA (CATS/TRAPS/LNDGS)
  - . SERVICE LIFE LIMITS
  - . TECHNICAL DIRECTIVES
- USERS
  - . SQUADRONS/WINGS/NAVAIR/OPNAV/...
- HARDWARE
  - . CENTRAL COMPUTING SUBSYSTEM
  - . SQUADRONS NETWORKED WITH PCS
- SOFTWARE
  - . DATABASE MANAGEMENT SYSTEM
  - . PC-BASED ONLINE GRAPHIC INTERFACE
  - . RANDOM ACCESS
  - . MINIMAL TRAINING

FIGURE 25

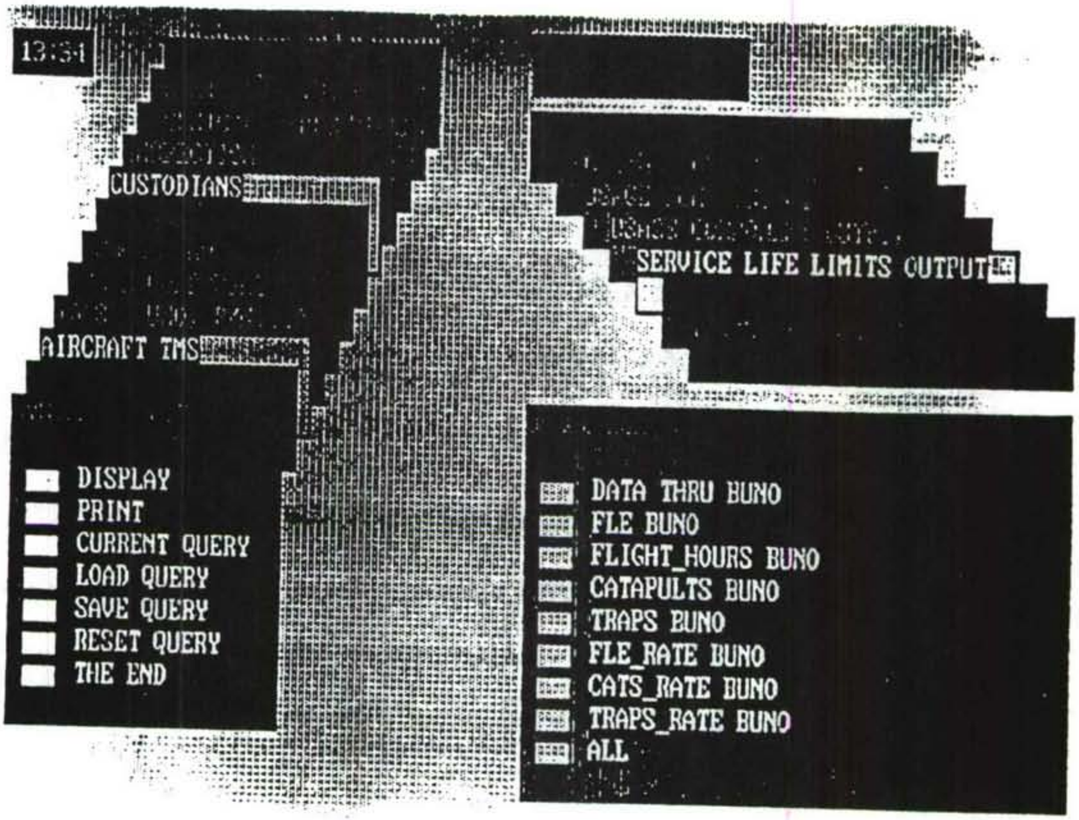

FIGURE 26

SOFI FUTURE EFFORTS

- SDRS SLATED FOR INSTALLATION IN OTHER A/C
  . FIXED WING AIRCRAFT (F-14, E-2, C-2 ..)
  . ROTARY WING AIRCRAFT (H-46, AH-1W..)
- DEVELOP/IMPLEMENT ONLINE CONFIG TRACKING
- PROVIDE PRELIMINARY USAGE DATA DURING DOWNLOADING
- TRANSITION TO COMMON METHODOLOGY AND ALGORITHM FOR ALL NAVY AIRCRAFT
- INTEGRATE ALL SUBSYSTEMS / ALL AIRCRAFT

FIGURE 27

CONCLUDING REMARKS

- BRIEF UPDATE ON SOFI
- STEADY PROGRESS ON ALL SUBSYSTEMS
- SDRS QUALIFIED, A-6 INSTALLATIONS UNDERWAY
- PROCESSING SOFTWARE DEVELOPED/TESTED
- DISSEMINATION SOFTWARE DEFINED/DEVELOPED
- FUTURE EFFORTS --- INTEGRATION

RISK ANALYSIS

AND

PROBABILISTIC DESIGN

339-1317-57

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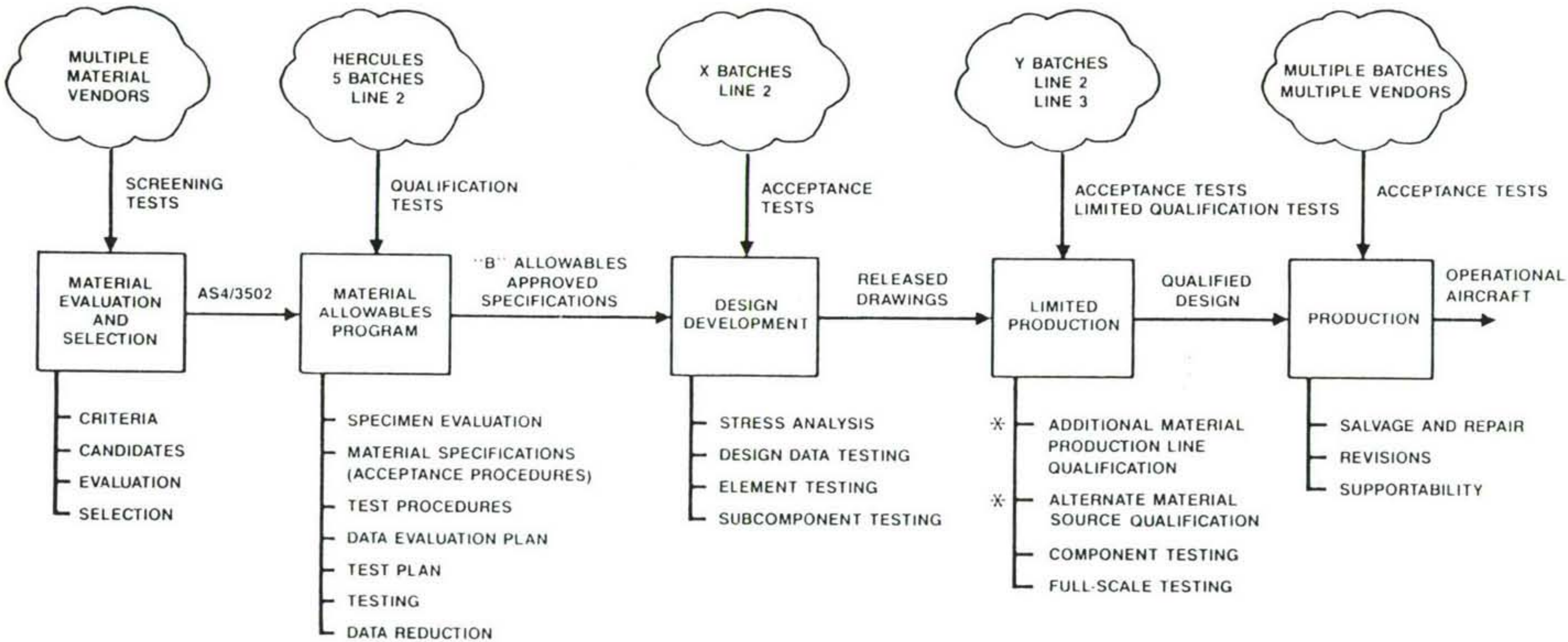
COMPOSITE MATERIAL EVOLUTION ON A TYPICAL AIRCRAFT PROGRAM



* APPARENTLY LOWER COMPRESSION PROPERTIES

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ISSUES

- Material allowables definition
  - Decisions
  - Compression testing
- Material acceptance criteria
- What is aircraft risk if material has degraded?
  - Operational usage
  - Static test
- Design for reliability

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RISK ANALYSIS

DESIGN ALLOWABLES RISK ANALYSIS

- Operational aircraft
  - Loads developed from exceedance data
  - 180°/wet properties
  - Sensitivity to mechanical properties

- Static test article
  - Ultimate loads
  - Room temperature, dry properties
  - Property distributions developed from actual material used

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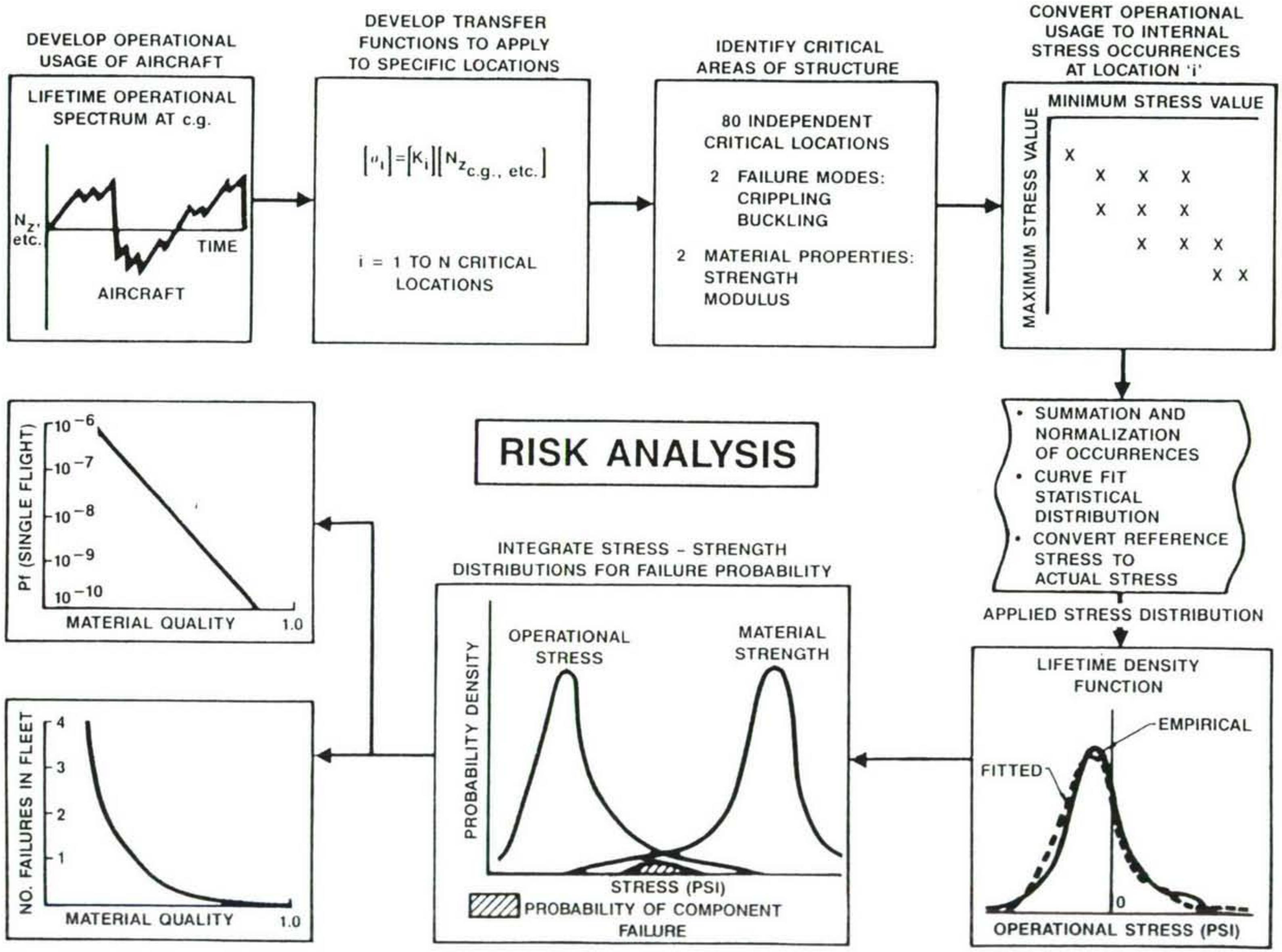

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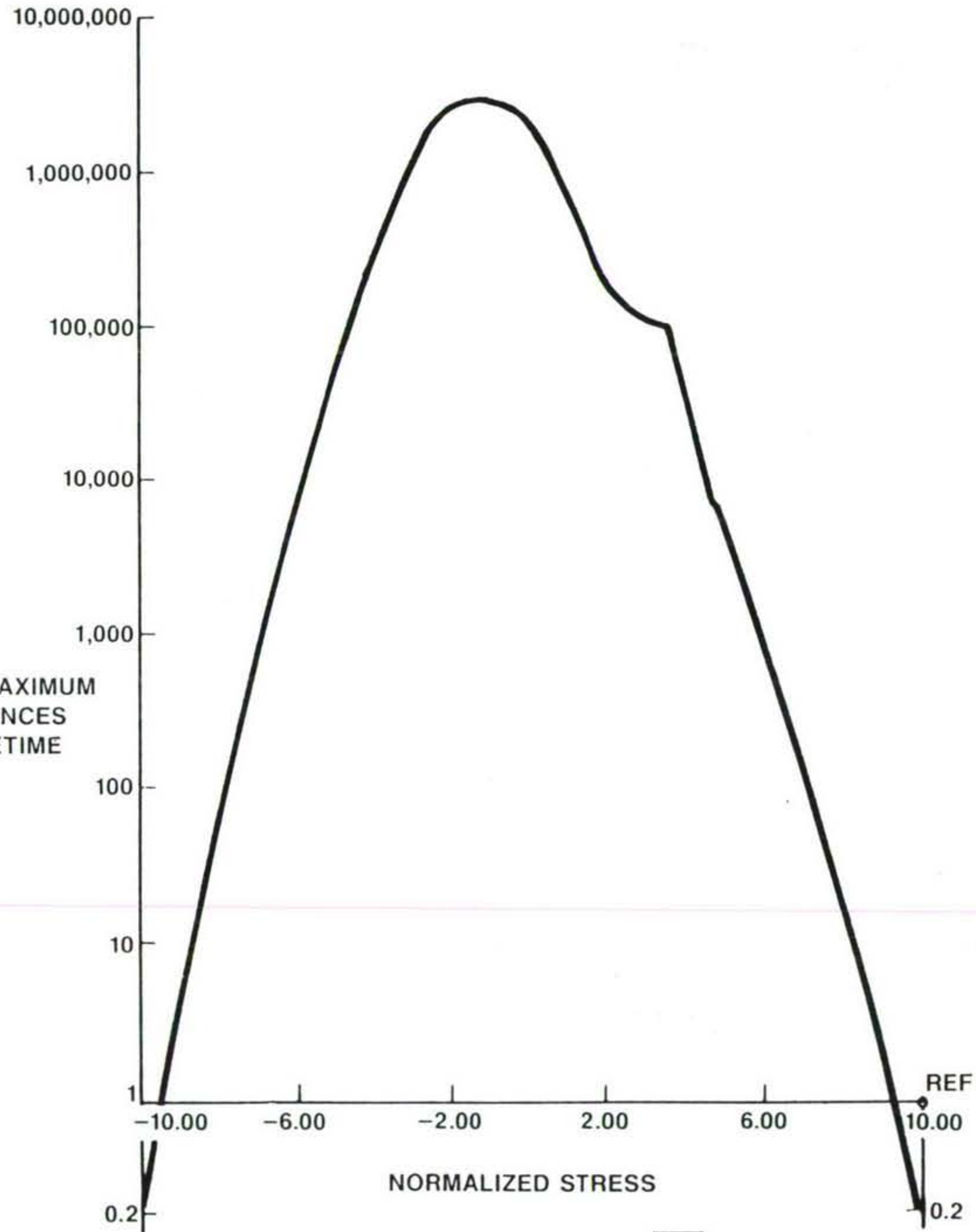
OPERATIONAL SPECTRUM
LIMIT LOAD OCCURS
ONCE PER 5 LIFETIMES



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VARIABLES AND ASSUMPTIONS

Stress

Mission mix, Weight, Load factor, Frequency	Covered by operational usage data
Stress prediction	Conventional methods
Temperature	Constant, worst case
Moisture	Constant, worst case
Effect of damage	1/4-inch diameter hole considered to exist at all areas

Strength

Batch variation	In allowables
Processing	Normal
Environment	Worst case
Manufacturing flaws/discrepancies	
Detected	Repaired to original capability
Undetected	Covered by 1/4-inch hole criteria

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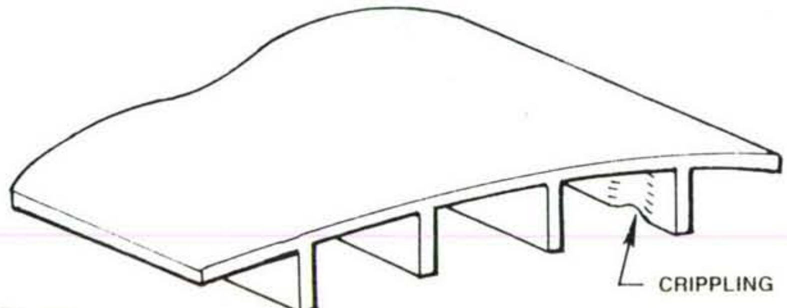
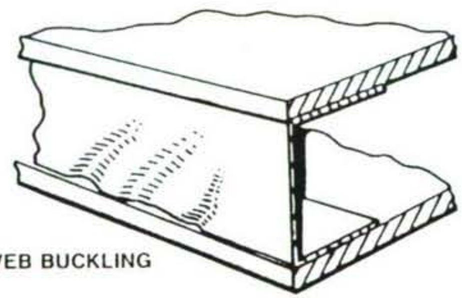
LIST OF CRITICAL MARGINS

	Location/Part No.	MS	Failure Mode
1.	Upper skin, No. 12 blade	0.02	Crippling
2.	Upper skin, No. 4 blade	0.06	Column buckling
3.	Rear spar web, 210 O/B	0.02	Buckling (combined loads)
4.	Lower skin	0.04	Bearing
5.	Upper skin	0.04	Buckling
A.	Rear spar cap, titanium	0.02	

6-40 used lowest probability of failure value from first 5





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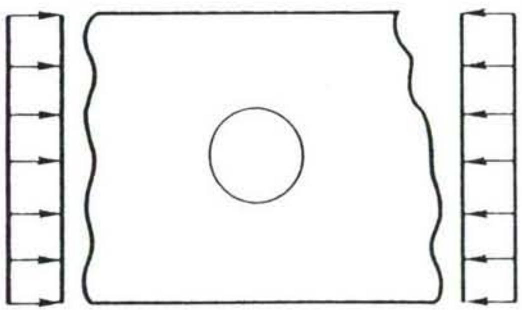
STRESS SHIFT FOR FAILURE MODE

HYPOTHESIS: CRIPPLING AND OPEN HOLE COMPRESSION ARE LINEARLY PROPORTIONAL TO MATERIAL COMPRESSION STRENGTH

APPLICATION: STRUCTURAL MARGINS-OF-SAFETY CAN BE USED TO CALCULATE APPLIED STRESS DISTRIBUTION.

EXAMPLE:



MATERIAL COMPRESSION ALLOWABLE = 181,000 PSI

BOLT HOLE COMPRESSION ALLOWABLE = 60,000 PSI

APPLIED STRESS = 57,100 PSI AT ULTIMATE

FACTOR-OF-SAFETY = 1.05

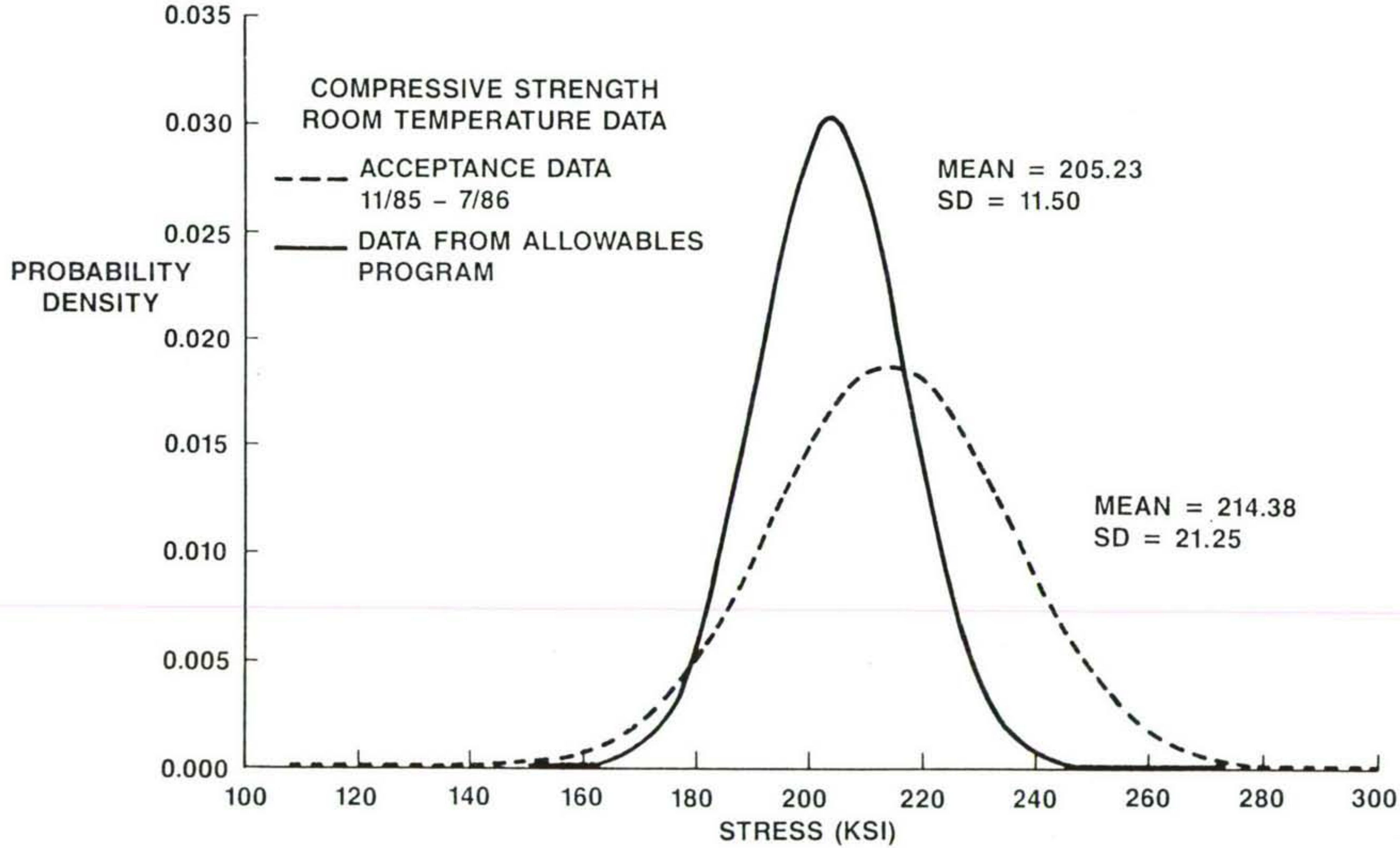
MARGIN-OF-SAFETY = +0.05

OPERATING STRESS IS REFERENCED TO LIMIT LOAD

$$\text{STRESS AT LIMIT LOAD} = \frac{\text{MATERIAL ALLOWABLE}}{\text{FACTOR OF SAFETY} \times 1.5} = \frac{181{,}000}{1.05 \times 1.5} = 115{,}000 \text{ PSI}$$

339-1317-51

AS4/3502 STRENGTH COMPARISON



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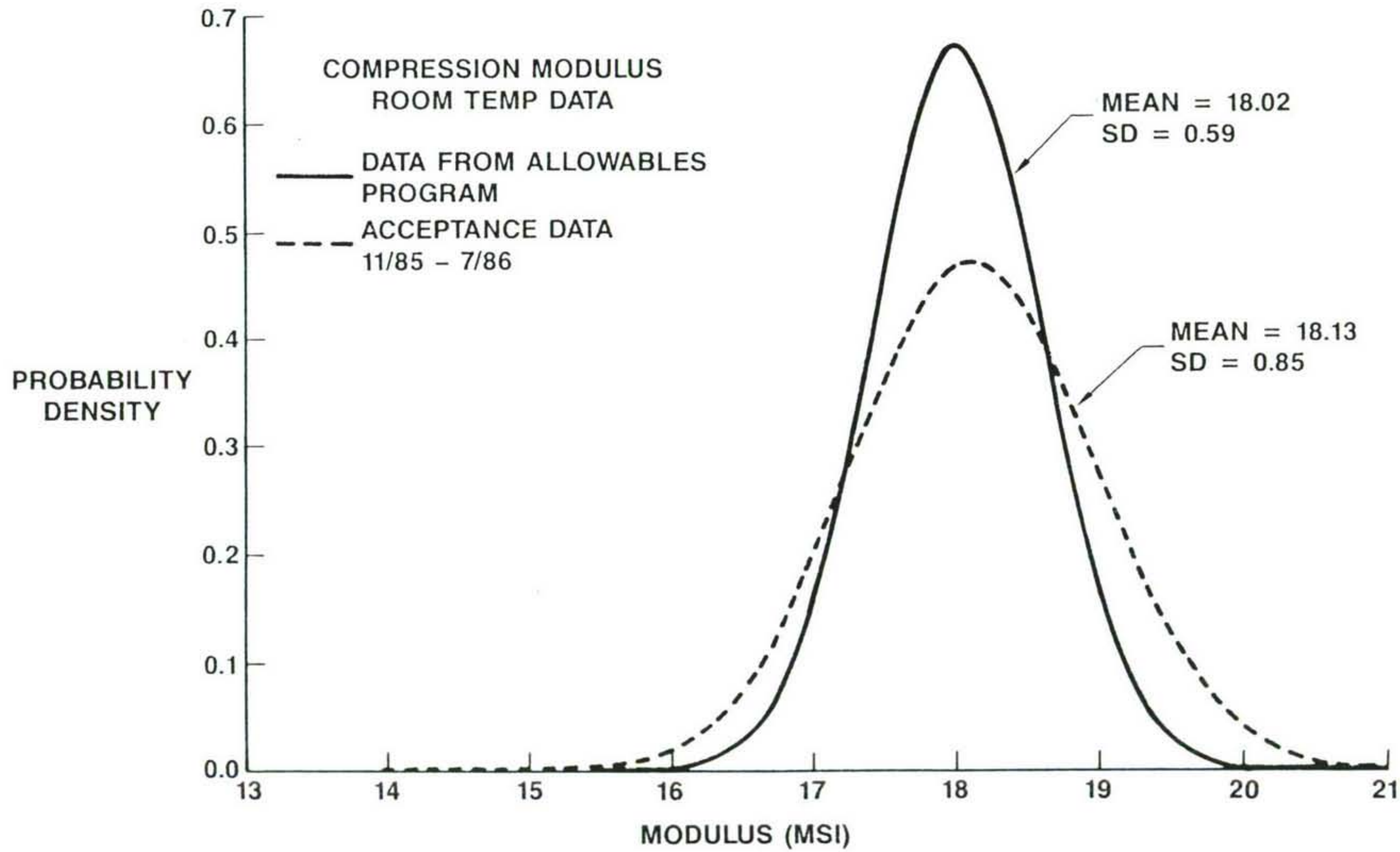
AS4/3502 MODULUS COMPARISON



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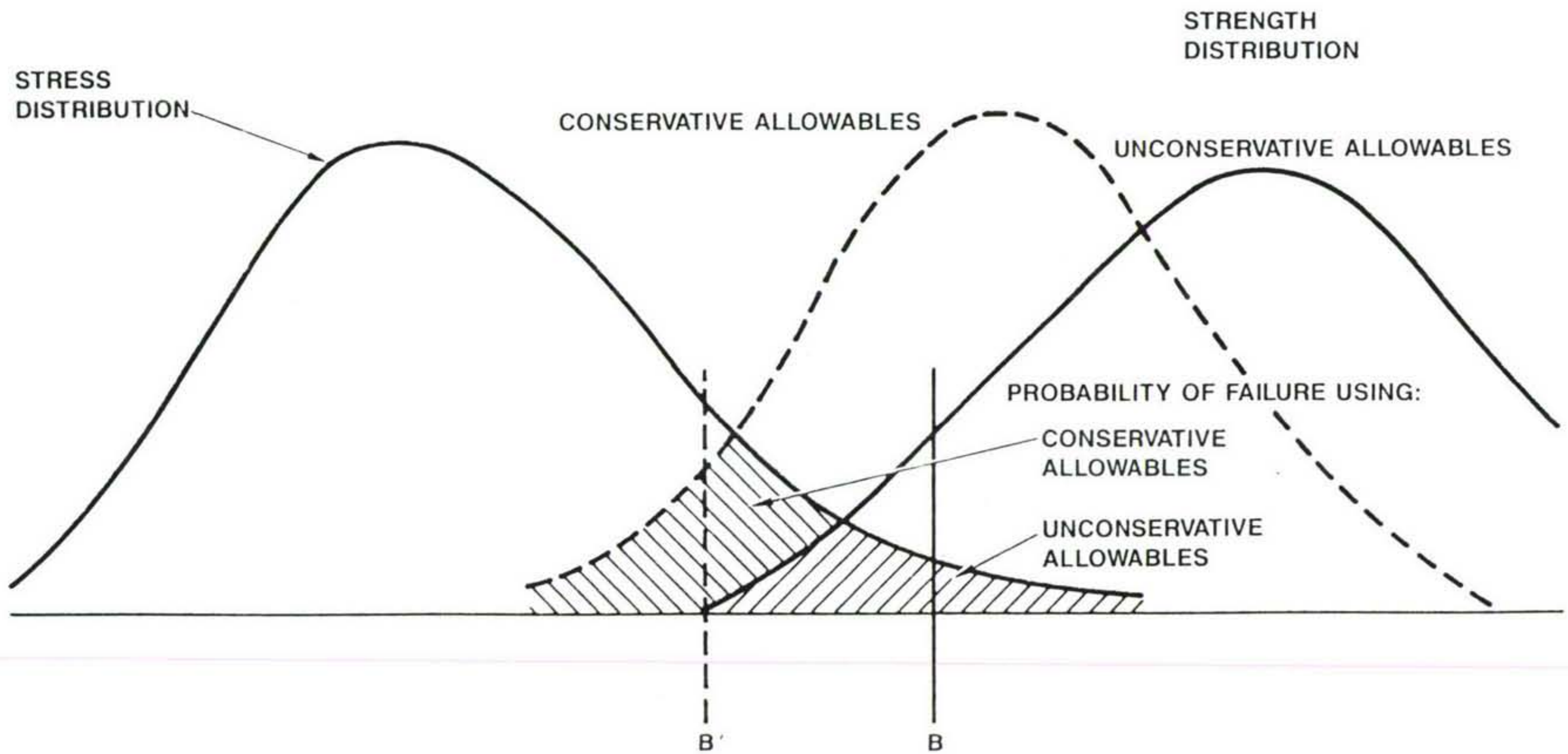
RISK ANALYSIS



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RISK ASSESSMENT

- Definition of acceptable risk level
  - "The first criteria is that risk of structural failure in a single flight should be no greater than the risk routinely accepted by the public. The second criterion is that the risk should be controlled such that the expected number of losses due to structural failure in the lifetime of the fleet should be less than one. The application of these principles indicate that the single flight probability of failure should be 10^{-7} or less."

Dr. John W. Lincoln

- Risk assessment of the F-16 wing
  - Left wing failed at 128% limit load
  - Wing redesigned for acceptable risk
  - Corresponded to FS = 1.45

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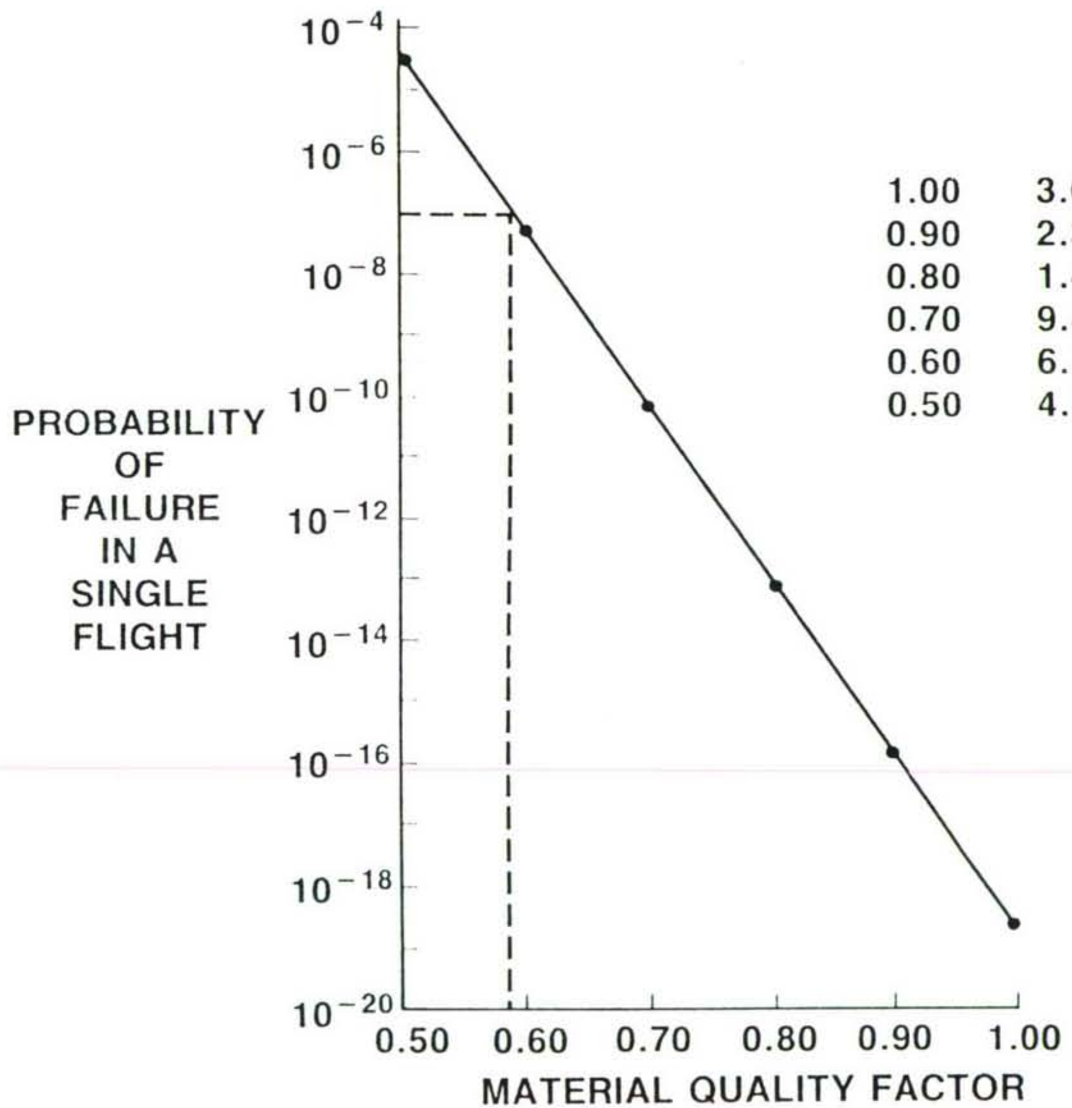
MATERIAL QUALITY SENSITIVITY



1.00	3.0×10^{-19}
0.90	2.3×10^{-16}
0.80	1.4×10^{-13}
0.70	9.4×10^{-11}
0.60	6.1×10^{-8}
0.50	4.5×10^{-5}

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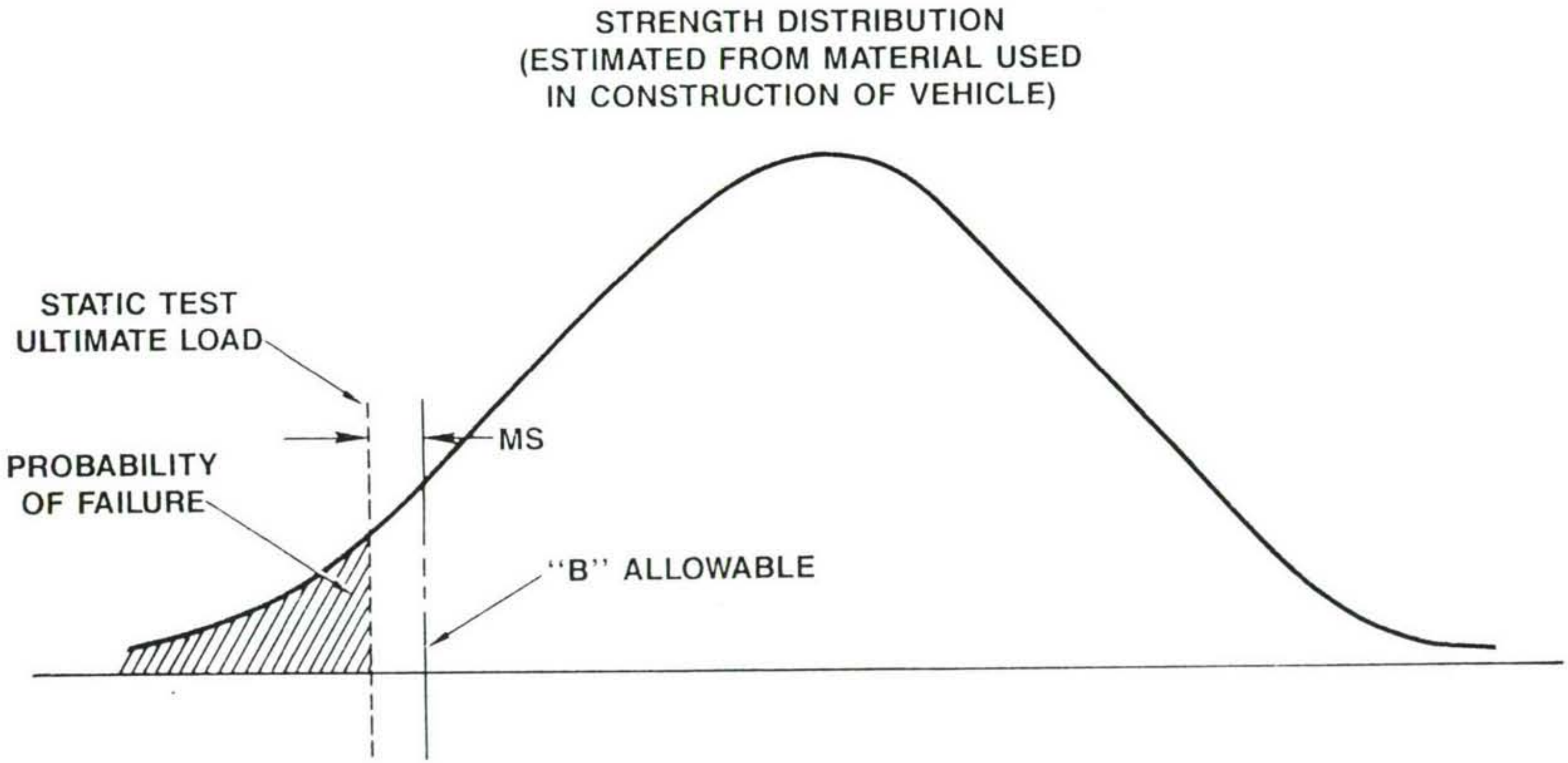
RISK ANALYSIS OF STATIC TEST VEHICLE



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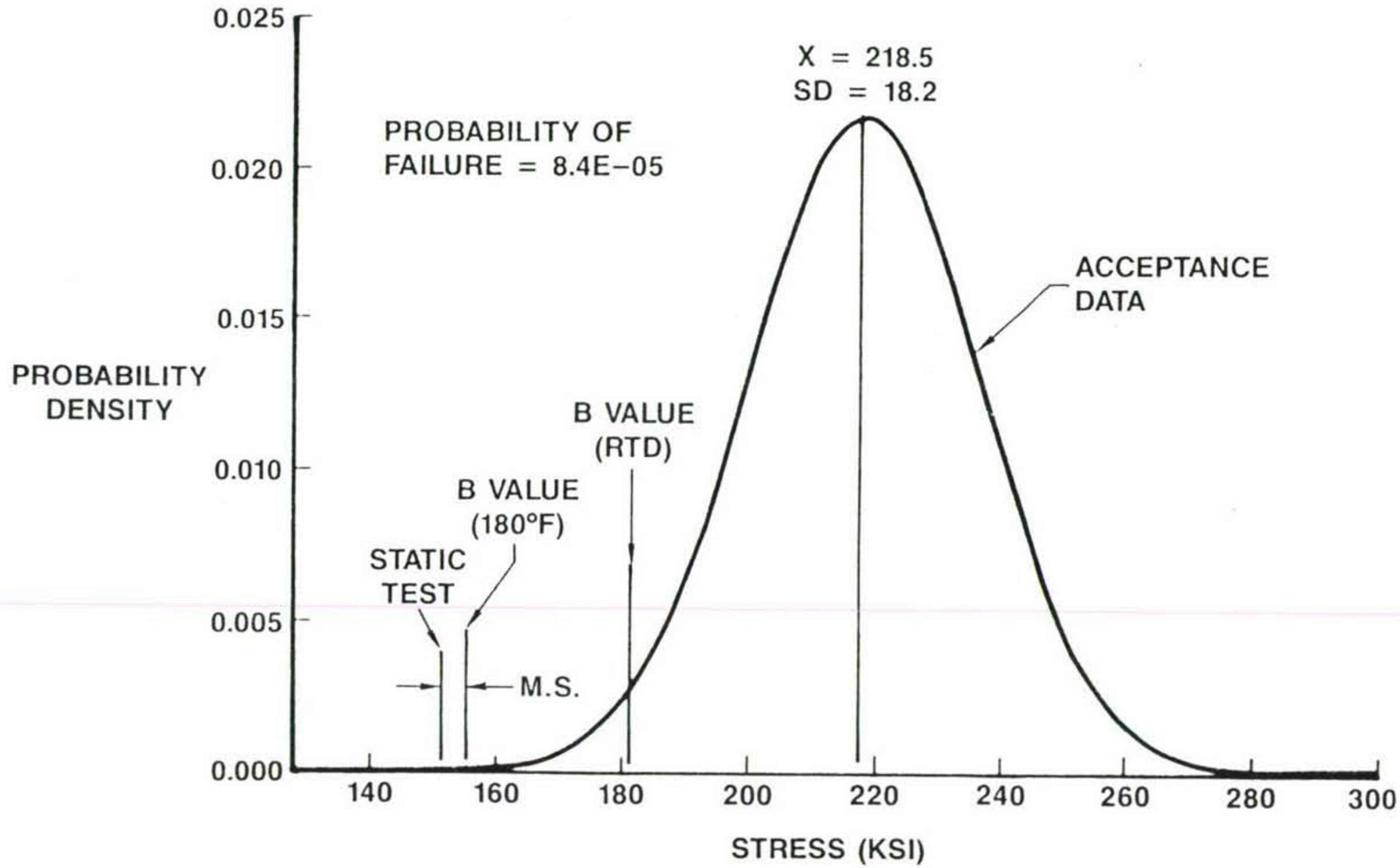
STATIC STRENGTH

AFT BOX UPPER SKIN (LOCATION NO. 4)



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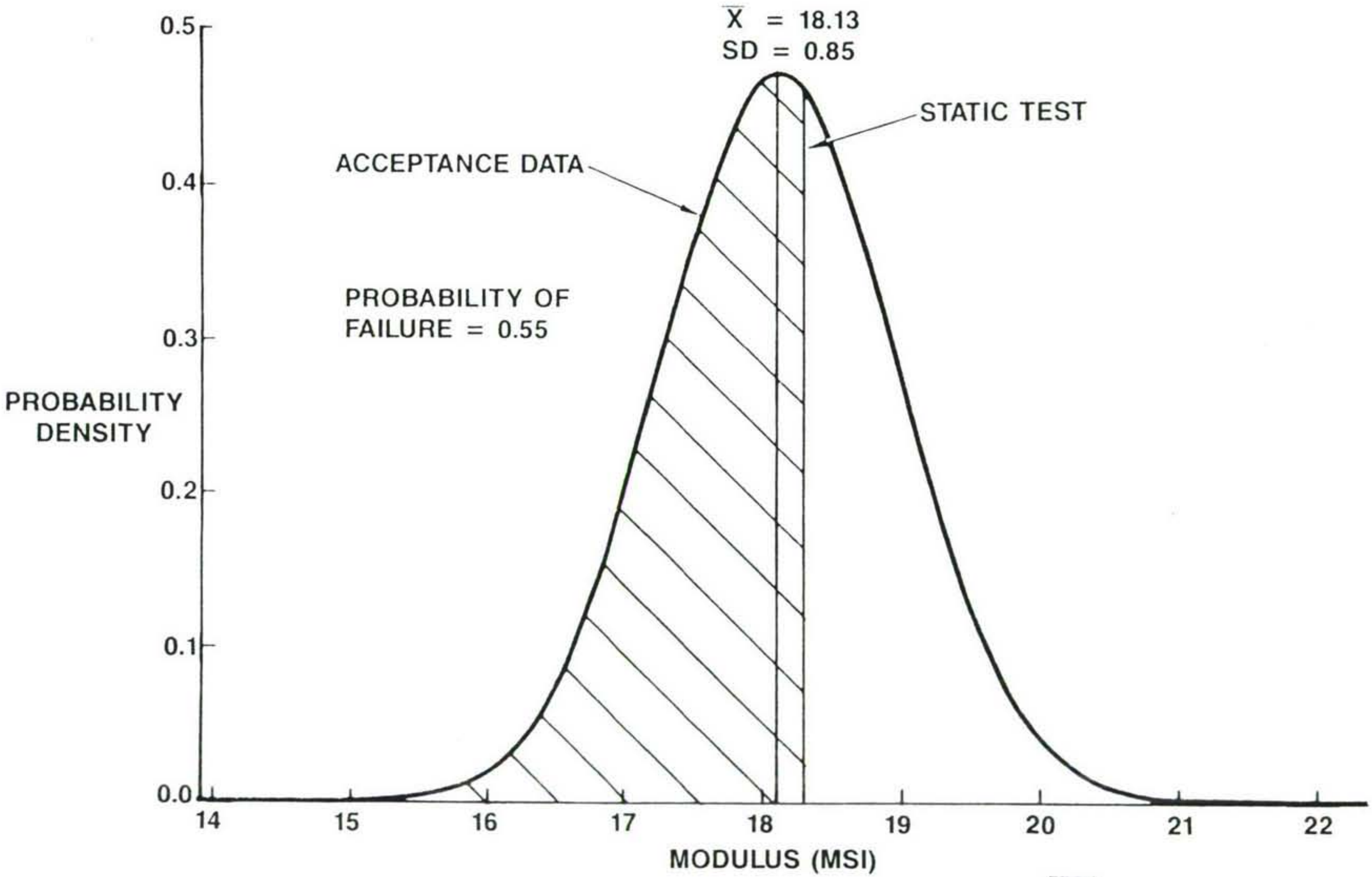
STATIC TEST

REAR SPAR WEB OUTBOARD OF 210 (LOCATION NO. 3)



339-1317-34

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STATIC TEST ANALYSIS

LOCATION	DESCRIPTION	PROBABILITY OF FAILURE
1	NO. 12 BLADE UPPER SKIN	1.8 E-05
2	NO. 4 BLADE UPPER SKIN 210	0.186
3	REAR SPAR WEB OUTBOARD OF 210	0.553
4	AFT BOX LOWER SKIN	8.4 E-05
5	UPPER SKIN	0.374
6	TITANIUM FORGING	0.020

TOTAL PROBABILITY OF FAILURE (ENTIRE STRUCTURE) = 0.95

339-1317-35

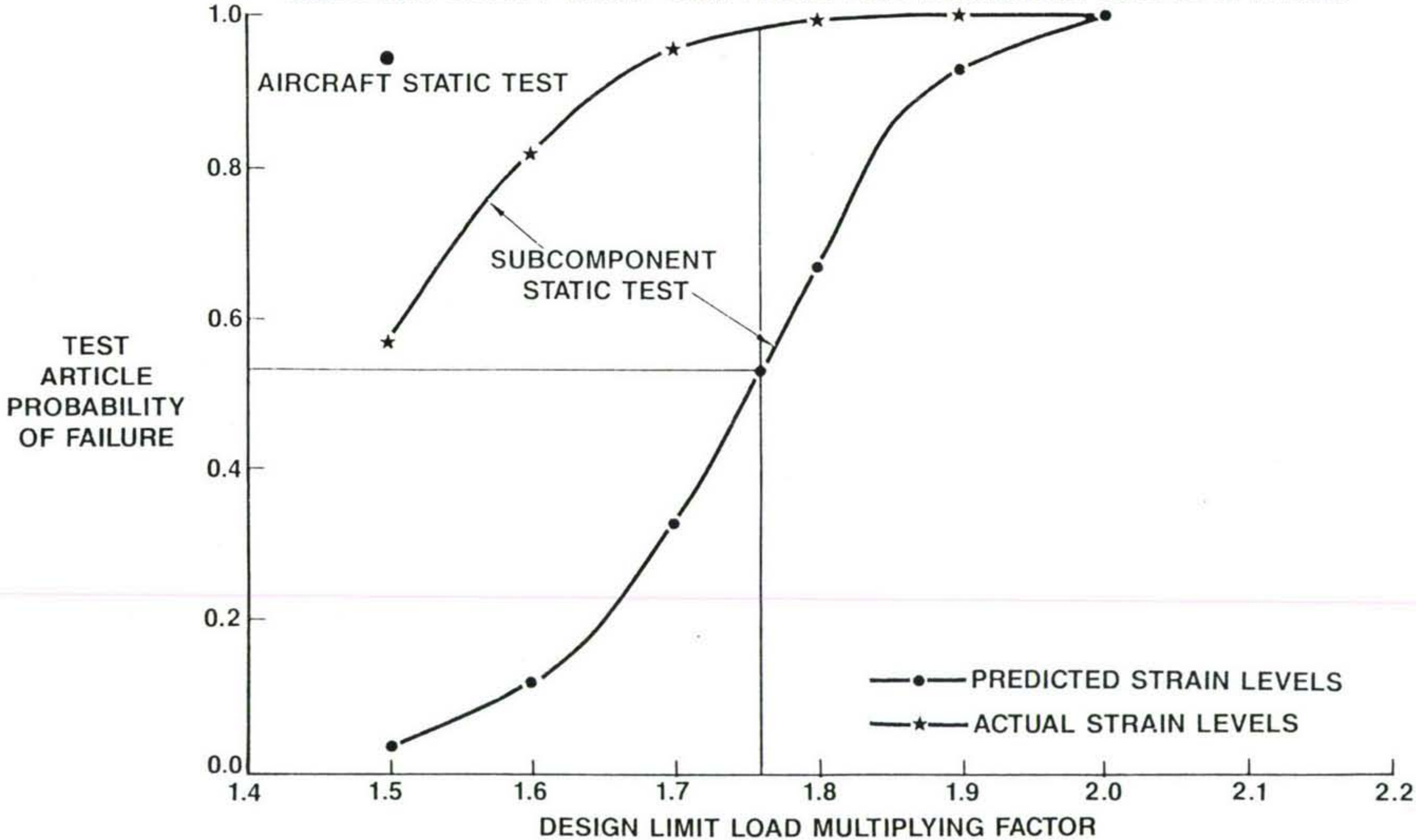
LTV Aircraft Products Group

MAJOR SUBCOMPONENT STATIC TEST

- Identified material batches
- Established probability density functions for material properties
- Calculated risk
- Compared to test results
  - Catastrophic failure at 176% design limit load
  - Failure was not in composite material

TEST LEVEL SENSITIVITY

SUBCOMPONENT TEST: CRIPPLING AND BUCKLING CRITICAL ITEMS



339-1317-39

LTV Aircraft Products Group

CONCLUSIONS
RISK ASSESSMENT

- Analysis procedures contained conservative assumptions
  - Strength variables
  - Stress variables
- Operational aircraft can tolerate a significant variation in material properties. Very high reliability
- Static test article has high probability of failure
  - Not design allowables problem
  - Consequence of failure is not catastrophic
- Subcomponent test shows risk analysis method is conservative

339-1317-40

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DESIGN FOR RELIABILITY

339-1317-41

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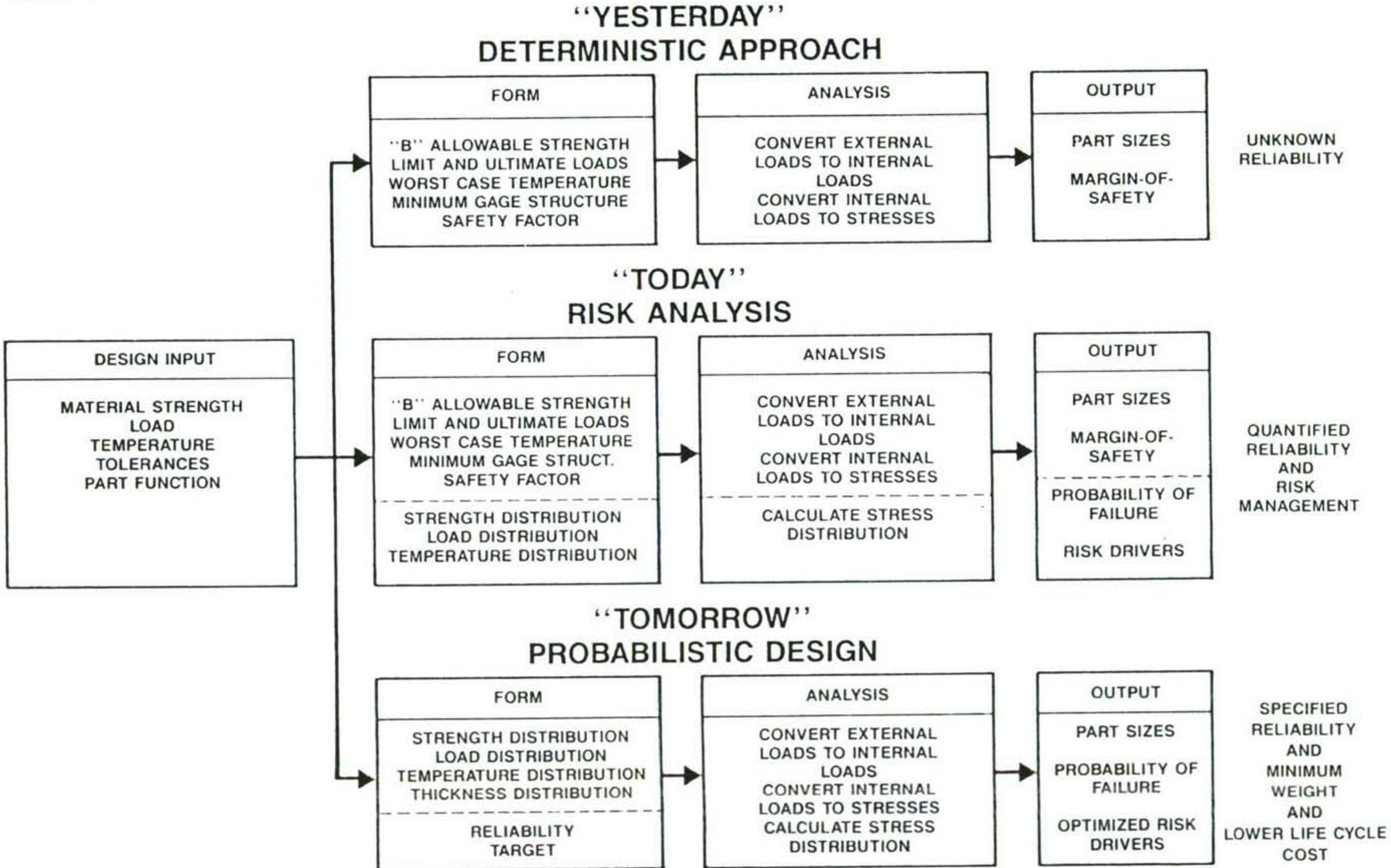
DESIGN EVOLUTION



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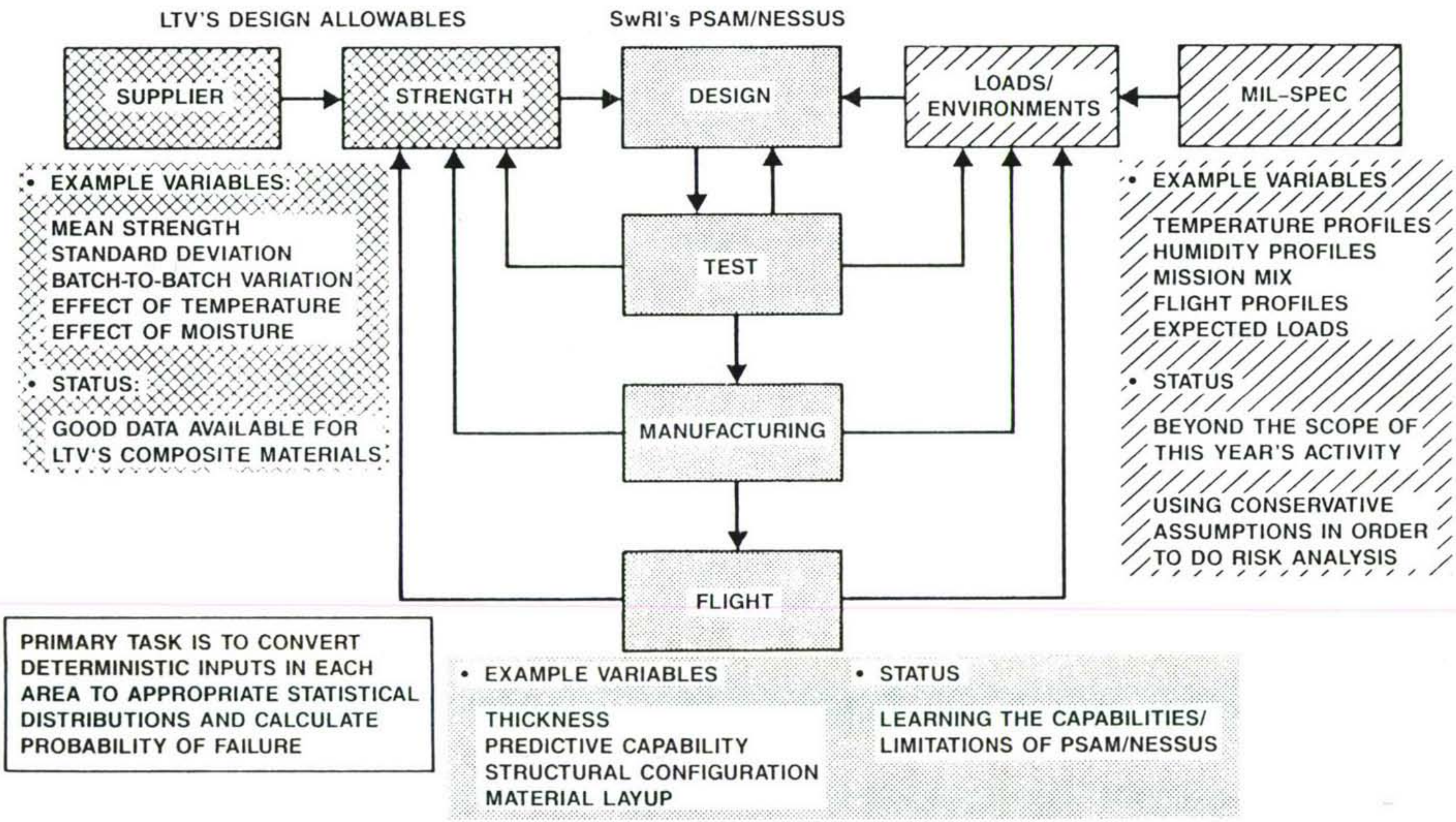
339-1317-42

STRUCTURAL DESIGN DEVELOPMENT CLOSED LOOP APPROACH



PRIMARY TASK IS TO CONVERT DETERMINISTIC INPUTS IN EACH AREA TO APPROPRIATE STATISTICAL DISTRIBUTIONS AND CALCULATE PROBABILITY OF FAILURE

339-1317 43

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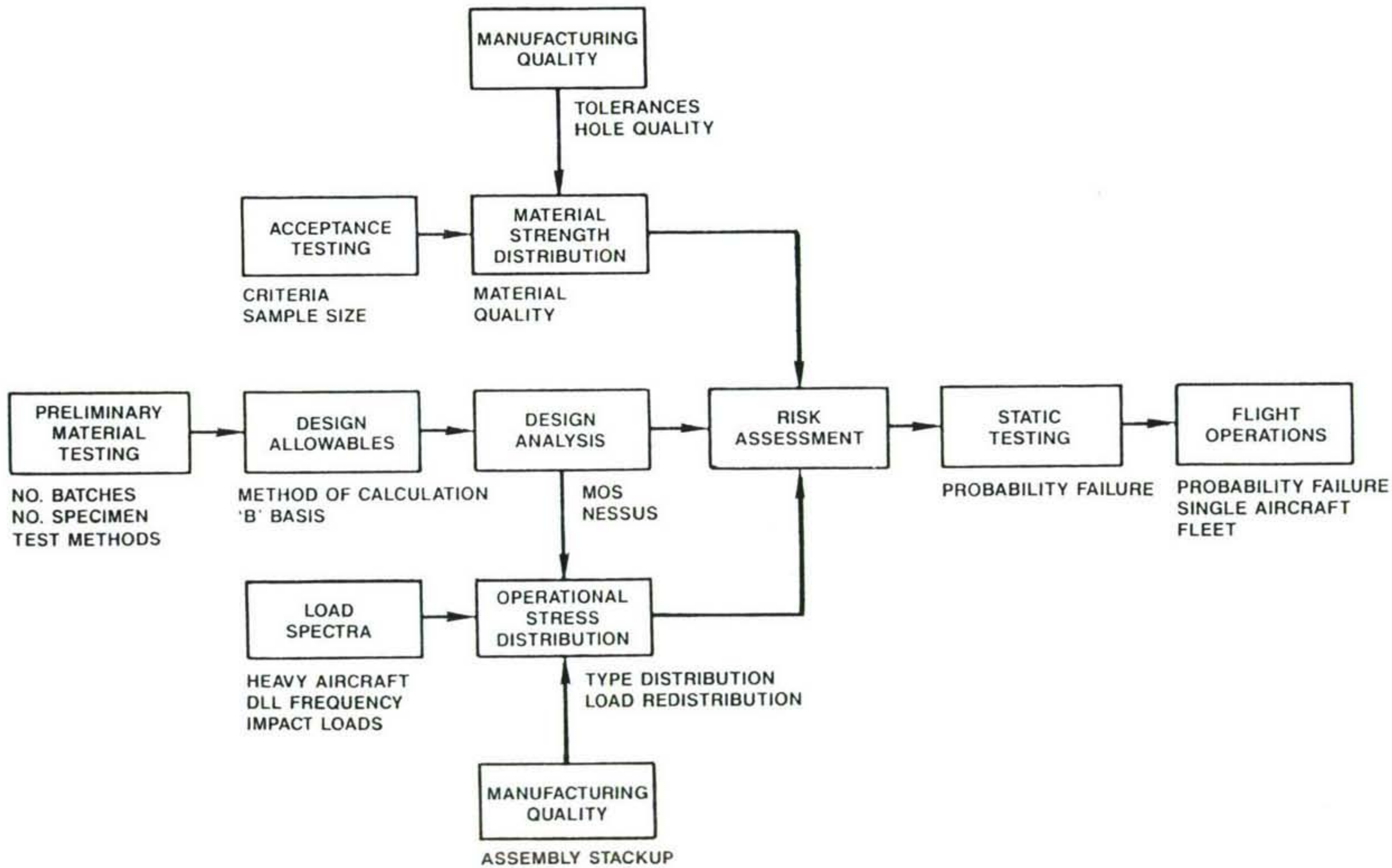
FULL PROBABILISTIC ANALYSIS



(1) BUILD RELATIONSHIP BETWEEN BOXES
(2) WHEN ONE BOX CHANGES, DETERMINE IMPACT ON OTHERS

339-1317-44

LTV Aircraft Products Group

OVERALL SUMMARY

- Material allowables development involves a variety of tasks requiring a number of choices
- Compression properties are very sensitive to test methods
- Material acceptance criteria should be related to material allowables
- Risk analysis provided confidence in current allowables
- Probabilistic design offers a potential for weight savings and lower maintenance cost

339-1317-48

LTV

Lockheed

C-130
Loads Environment Spectra Survey
(LESS)

1989 USAF Structural Integrity
Program Conference
5-7 December 1989

T. C. Bell, Lockheed Aeronautical
Systems Company-Georgia

W. O. Greenhaw, Warner Robins
Air Logistics Center

Transition to the Standard Flight Data Recorder (SFDR) System

GA-8762-1
10-18-89

LESS PURPOSE

THE ESTABLISHMENT AND OPERATION OF A LOADS ENVIRONMENT SPECTRA SURVEY (LESS) FOR U.S. AIR FORCE AIRCRAFT ARE BASED ON THE REQUIREMENTS OF MIL-STD 1530A, SECTION 5.4.4.

THE PURPOSE OF THE LESS IS TO PROVIDE TIME HISTORY RECORDED DATA TO ESTABLISH A BASELINE DESCRIPTION OF THE OPERATIONAL LOADING ENVIRONMENT AND, THEREAFTER, TO PROVIDE PERIODIC COMPARISONS OF CURRENT LOADING EXPERIENCE TO THE ESTABLISHED BASELINE. RESULTS OF THE COMPARISON ARE UTILIZED TO DETERMINE IF CHANGES ARE OCCURRING, THE CAUSES FOR SUCH CHANGES, AND THE IMPACT OF CHANGES UPON THE FORCE MANAGEMENT OF THE AIRCRAFT.

FOR THE C-130 FORCE MANAGEMENT, THE LESS DATA ARE UTILIZED PRIMARILY FOR DERIVATION OF LOADING DESCRIPTION CRITERIA, ANALYTICAL LOADS MODEL CORRELATION, AND MISSION PROFILE DESCRIPTION REFINEMENTS.

Lockheed

C-130 LESS

LESS Purpose

- Basis: MIL-STD 1530A
- Recorded Data To:
  - Define Operational Loading Environment
  - Provide Baseline for Periodic Comparisons
- Determine Loading Changes and Causes
- Provide Input to Aircraft Force Management

GA-8762-2
6-21-89

C-130 LESS BACKGROUND

THE ORIGINAL MIL-STD 1530 BASED C-130 LOADS MONITORING PROGRAM WAS TERMED THE LIFE HISTORY RECORDING PROGRAM (LHRP), AND WAS IMPLEMENTED IN THE EARLY 1970'S. THE C-130 LHRP HAS UTILIZED THE MXU 553/A RECORDING SYSTEM, AS HAVE MANY USAF FLEETS, FOR THE ACQUISITION OF DATA FROM 60 ORIGINALLY INSTRUMENTED C-130 AIRCRAFT. THE SYSTEM IS STILL IN OPERATION ON A REDUCED NUMBER OF C-130 AIRCRAFT, BUT IS SCHEDULED FOR REPLACEMENT WITHIN THE NEXT FEW YEARS BY A REVISED LESS SYSTEM.

THE LESS SYSTEM WILL CONSIST OF COMPLETELY UPDATED AIRCRAFT INSTRUMENTATION AND SENSORS, AND WILL UTILIZE THE USAF STANDARD FLIGHT DATA RECORDER (SFDR). THE LESS SYSTEM WILL ALSO CONSIST OF AN UPDATED AND GREATLY ENHANCED GROUND DATA PROCESSING AND ANALYSIS COMPUTER NETWORK.

Lockheed

C-130 LESS

Background

- Life History Recording Program (LHRP) Implemented in Early 1970s.

- LHRP System Being Replaced by LESS System
  - Updated Instrumentation/Sensors/Recorder
  - Updated/Enhanced Ground Processing/Analysis

GA-8762-3
6-21-89

C-130 LESS PROTOTYPE PROGRAM PURPOSE

IN 1984, WARNER ROBINS AIR LOGISTICS CENTER (WR-ALC) DIRECTED LOCKHEED TO DEVELOP A PROTOTYPE LESS SYSTEM TO DEVELOP AND DEMONSTRATE THE METHODS AND PROCESSES TO BE UTILIZED IN THE REVISED, SFDR BASED C-130 LESS SYSTEM.

THE PURPOSES OF THE PROTOTYPE PROGRAM ARE:

- THE DEFINITION AND IMPLEMENTATION OF DEVELOPMENTAL ONBOARD INSTRUMENTATION HARDWARE AND FIRMWARE ON A SINGLE C-130 AIRCRAFT TO ACQUIRE, PRE-PROCESS, AND STORE THE REQUIRED DATA IN APPROPRIATE FORMS FOR FURTHER GROUND BASED PROCESSING AND ANALYSIS.

- THE DEFINITION AND DEVELOPMENT OF A GROUND BASED COMPUTER PROGRAM NETWORK TO PROCESS THE RESULTING RECORDED DATA.

- THE ACQUISITION OF OPERATIONAL DATA FOR EVALUATION AND ADJUSTMENT OF INSTRUMENTATION, METHODS, AND DATA FORMS.

- THE PROVISION OF SUPPORT TO THE SFDR PRIME CONTRACTOR (SMITHS INDUSTRIES) IN THE TRANSITION OF THE PROTOTYPE METHODS AND PROCESSES INTO THE PRODUCTION SFDR EQUIPMENT.

Lockheed

C-130 LESS

Prototype Program Purposes

- Define and Develop:
  - Developmental Hardware and Firmware
  - Ground Computer Programs
- Acquire Operational Data
- Provide Support in Transition to Production System

GA-8762-4
6-21-89

INSTRUMENTATION

THE C-130 LESS DATA PARAMETERS CONSIST OF 22 DIRECT MEASURED QUANTITIES PLUS MANUALLY GENERATED QUANTITIES FOR PRE-FLIGHT AND POST-FLIGHT WEIGHTS AND INFLIGHT DISCRETE WEIGHT CHANGES (AIRDROP, AERIAL REFUELING, ETC.). THE MANUAL DATA IN THE PROTOTYPE SYSTEM ARE OBTAINED FROM AFTO 151 USAGE FORMS AND IN THE SFDR SYSTEM WILL BE ENTERED THROUGH AN INPUT DATA PANEL BY THE FLIGHT ENGINEER.

THE DIRECT MEASURED PARAMETERS ARE COMPRISED OF 11 BASIC PARAMETERS, INCLUDING VERTICAL ACCELERATION AT THE AIRCRAFT NOMINAL CENTER OF GRAVITY, AS LISTED IN THE ADJOINING FIGURE AND:

- CONTROL SURFACE DEFLECTIONS FOR ELEVATOR, RUDDER, AND LEFT AILERON
- STRAIN GAGE MEASUREMENTS AT THE FOLLOWING LOCATIONS:

    -- LOWER VERTICAL STABILIZER
    -- INBOARD LEFT HORIZONTAL STABILIZER
    -- FOUR LOCATIONS ON THE LEFT WING LOWER AFT BEAM
    -- TWO FUSELAGE CROWN (SINGLE INTEGRATED MEASUREMENT)
    -- NO. 1 ENGINE SUPPORT TRUSS

INSTRUMENTATION WILL BE INSTALLED ON APPROXIMATELY 10% OF THE USAF C-130 FLEET ON PROPORTIONATE NUMBERS OF THE VARIOUS C-130 MODEL TYPES. INSTRUMENTATION WILL BE THE SAME FOR ALL MODELS EXCEPT FOR THE AERIAL REFUELING (AR) CONNECTION SIGNAL WHICH IS ONLY APPLICABLE TO CERTAIN AR EQUIPPED C-130 AIRCRAFT, AND EXCEPT WHERE SPECIAL MODEL EQUIPMENT WOULD CAUSE ROUTING OR UNIT LOCATION CHANGES.

Lockheed

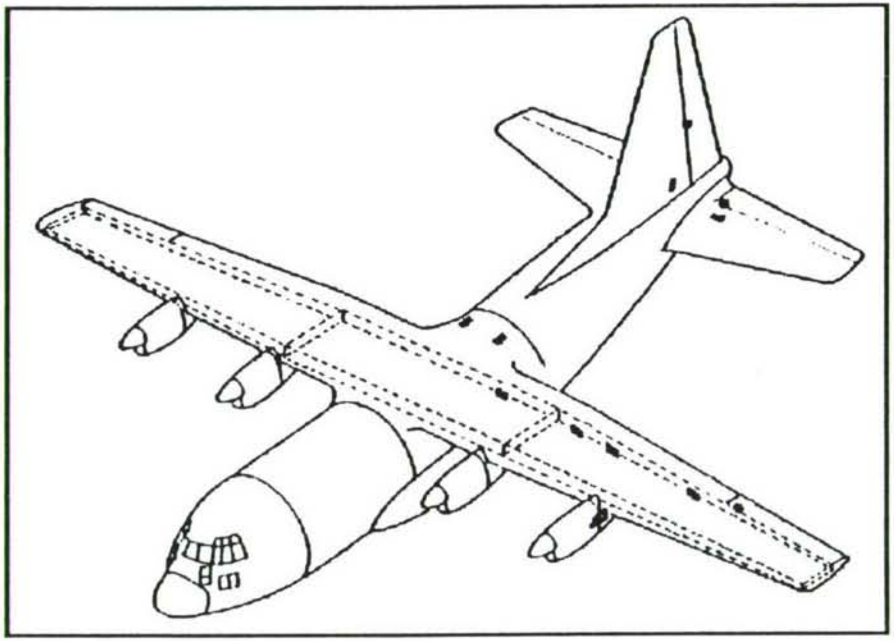
C-130 LESS

Instrumentation

Basic Parameters

- Pressure Altitude
- Radar Altitude
- Radar Altitude Flag
- Airspeed
- Flap Position
- Ground Speed
- Vertical Acceleration at C.G.
- Touchdown Switch
- Ramp Door Position
- No. 1 Engine Torque
- Aerial Refueling Connection *

* Only on Applicable Models



○ - Control Surface Deflections (3)

□ - Strain Gage Channels (8)

GA-8762-5
6-21-89

ONBOARD PROCESSING

THE ONBOARD FUNCTIONS OF THE CURRENT MXU 553/A BASED C-130 LHRP SYSTEM ARE ESSENTIALLY LIMITED TO SIGNAL CONDITIONING AND RECORDING. THE REVISED, SFDR BASED LESS SYSTEM, HOWEVER, UTILIZES THE CAPABILITIES OF THE MICROPROCESSOR TO PERFORM MANY FUNCTIONS ONBOARD WHICH HERETOFORE HAVE BEEN PERFORMED IN GROUND PROCESSING. THE PERFORMANCE OF THESE FUNCTIONS ONBOARD ALLOWS OPTIMIZATION OF MASS MEMORY AND DATA DOWNLOADING INTERVALS AND SHORTER DIAGNOSTIC/MAINTENANCE CYCLES WHILE STILL PROVIDING NECESSARY DATA VISIBILITY THROUGH CONDENSED TIME HISTORY RETENTION.

THE TYPES AND AMOUNT OF ONBOARD PROCESSING ARE DETERMINED BY SEVERAL FACTORS:

- THE NUMBER AND ACTIVITY LEVELS OF MONITORED PARAMETERS
- THE DATA FORMS REQUIRED FOR ANALYSIS
- THE REQUIRED MINIMUM AVERAGE DOWNLOADING INTERVAL
- THE AVAILABLE ONBOARD RANDOM ACCESS AND MASS STORAGE MEMORY

IN NO EVENT, HOWEVER, SHOULD DATA PROCESSING/CONDENSATION GO BEYOND THE POINT WHERE ADEQUATE DATA VISIBILITY CEASES TO BE PROVIDED TO THE DATA ANALYST TO VERIFY TRENDS, QUANTITIES AND THE QUALITY OF THE RECORDED DATA.

EACH OF THESE ONBOARD FUNCTIONS, LISTED IN THE ADJOINING FIGURE, ARE ADDRESSED BRIEFLY IN THE FOLLOWING FIGURES.

Lockheed

C-130 LESS

Onboard Processing

- Data Editing/Health Checks
- Flight Profile Determination
- Data Condensation
- Mean Level Determination
- Peak Counting
- Gust/Maneuver Response Separation
- Abrupt Mean Shift Recognition
- Condensed Time History Retention

GA-8762-6
6-21-89

DATA HEALTH TESTS

IN ADDITION TO THE BUILT-IN-TESTS (BIT) INCORPORATED INTO THE DESIGN OF THE SFDR, FOUR DIRECT TESTS ARE PERFORMED WITHIN THE SFDR TO VERIFY THE QUALITY OF THE ACQUIRED DATA.

THE ACTIVITY TEST VERIFIES THAT THE VARIOUS DATA CHANNELS ARE EXHIBITING MINIMUM REASONABLE ACTIVITY.

THE RANGE TEST ENSURES THAT THE MAGNITUDE OF THE DATA ARE WITHIN REASONABLE UPPER AND LOWER LIMITS.

THE RATE OF CHANGE TEST IS DESIGNED TO IDENTIFY "SPIKES" OR PARAMETER DERIVATIVES BEYOND THE PHYSICAL CAPABILITY OF THE AIRCRAFT.

THE CHATTER TEST MONITORS THE DATA FOR INSTABILITY OR "NOISY" SIGNALS WHICH COULD OTHERWISE SATURATE MASS STORAGE WITH ERRONEOUS DATA.

THE RESULTS OF THESE TESTS AS WELL AS THE BIT DATA WILL BE INTERPRETED BY GROUND SOFTWARE AT THE BASES OF OPERATION SUBSEQUENT TO DATA DOWNLOAD TO PROVIDE IMMEDIATE MAINTENANCE REQUIREMENTS.

Lockheed

C-130 LESS

Data Health Tests

- Activity Test
- Range Test
- Rate of Change Test
- Chatter Test

GA-8762-7
6-21-89

ONBOARD FLIGHT PROFILE DETERMINATION

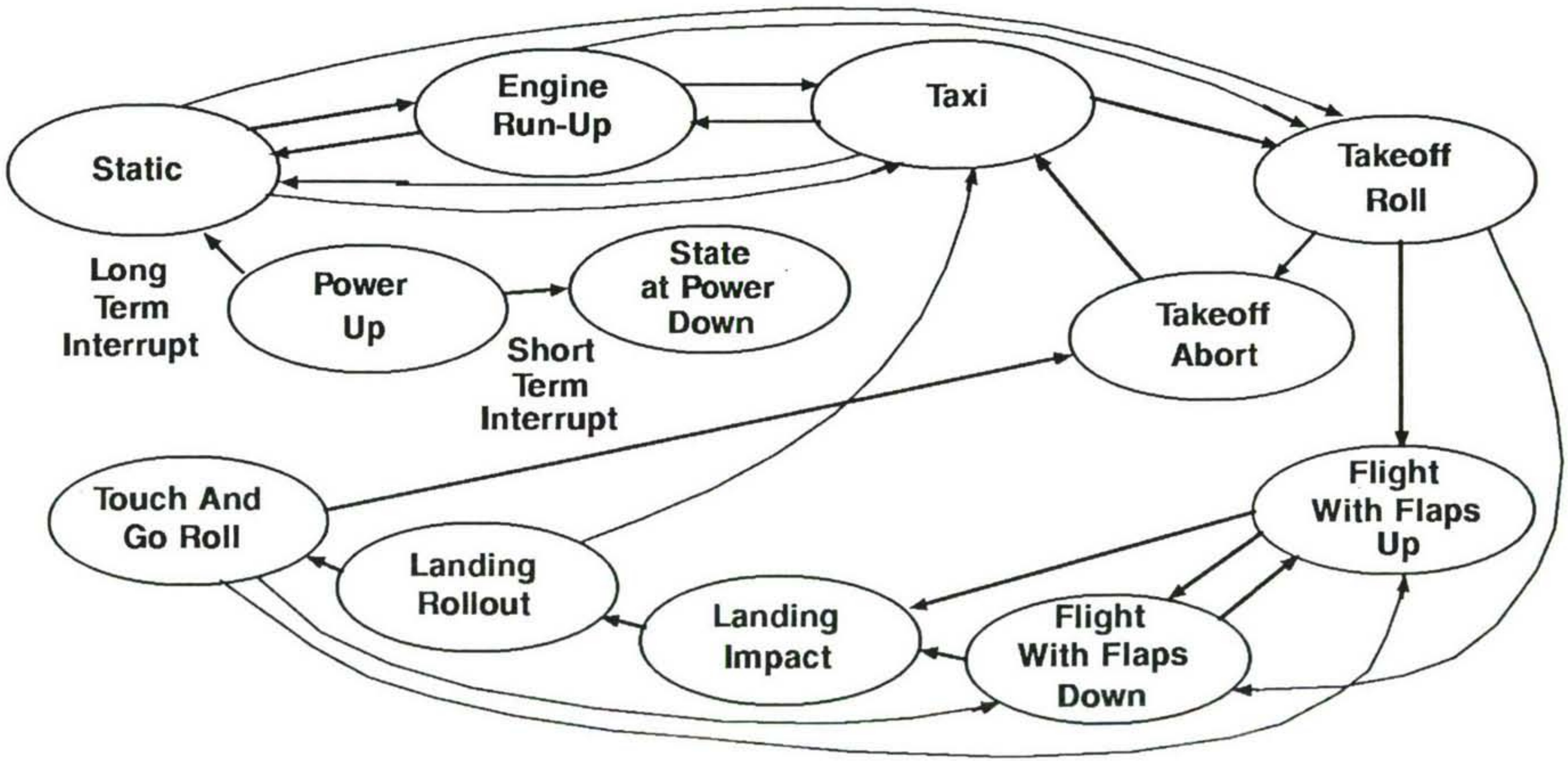
THE ONBOARD MEAN LEVEL CALCULATIONS, WHICH ARE NECESSARY FOR DATA CONDENSATION TO PROVIDE THE REQUIRED DATA DOWNLOAD INTERVAL (APPROX. 30 HOURS) WITH THE AVAILABLE MASS STORAGE (2 MEGABYTES), REQUIRES THE ONBOARD DETERMINATION OF THE BASIC REGIMES OF AIRCRAFT OPERATION OR FLIGHT PROFILE.

THE MEASURED PARAMETERS GROUND SPEED, FLAP POSITION, ENGINE TORQUE, AND TOUCHDOWN SWITCH ARE UTILIZED IN "REAL TIME" TO DETERMINE THE CURRENT GROUND MODE (TAXI, TAKEOFF, LANDING ROLLOUT, ETC.) OR FLIGHT WITH EITHER FLAPS DEPLOYED OR RETRACTED. THE FLIGHT MODES ARE FURTHER SUBDIVIDED IN GROUND PROCESSING INTO THE VARIOUS TYPES OF FLIGHT ACTIVITY (CLIMB, CRUISE, AIRDROP, ETC.).

Lockheed

C-130 LESS

Onboard Flight Profile Determination



GA-8762-8
6-21-89

DATA CONDENSATION FOR RESPONSE PARAMETERS

THE FIRST LEVEL OF DATA CONDENSATION INVOLVES THE ELIMINATION OF DATA POINTS WHICH ARE EITHER UNIMPORTANT IN PEAK/VALLEY DETERMINATION, REFLECT INSIGNIFICANT CHANGES IN MAGNITUDE, OR ARE REDUNDANT.

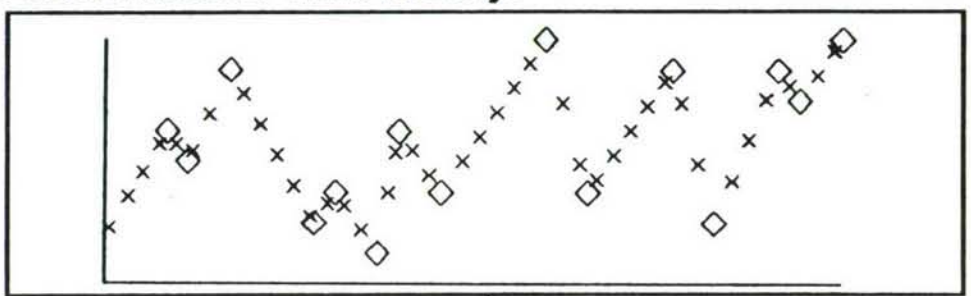
FOR THE STRESS, CONTROL SURFACE DEFLECTION, AND NZCG PARAMETERS WHICH ARE SUBSEQUENTLY "PEAK COUNTED" (TERMED RESPONSE PARAMETERS IN THIS PROCESS), THE FIRST LEVEL OF CONDENSATION CONSISTS OF THE ELIMINATION OF ALL MONITORED POINTS EXCEPT "LOCAL PEAKS" AND FURTHERLY, THE ELIMINATION OF LOCAL PEAKS WHICH REPRESENT A CHANGE IN MAGNITUDE FROM THE LAST RETAINED LOCAL PEAK WHICH IS LESS THAN A PRESCRIBED CONDENSATION THRESHOLD. A LOCAL PEAK IS A POINT OF SLOPE SIGN CHANGE (+ TO - OR VICE VERSA). THE STRESSES AND NZCG ARE MONITORED AT 32 HERTZ; THE CONTROL SURFACES AT 10 HZ, HOWEVER, THE POINTS ON THE SLOPES OF THE TIME HISTORY ARE NOT REQUIRED AND NOT RETAINED. THE CONDENSATION THRESHOLD IS .025g FOR NZCG, 125 PSI FOR STRESSES, AND .25 DEGREES FOR THE CONTROL SURFACE DEFLECTIONS.

THIS CONDENSATION PROCESS FOR THE RESPONSE PARAMETERS IS ILLUSTRATED IN THE ADJOINING FIGURE.

C-130 LESS

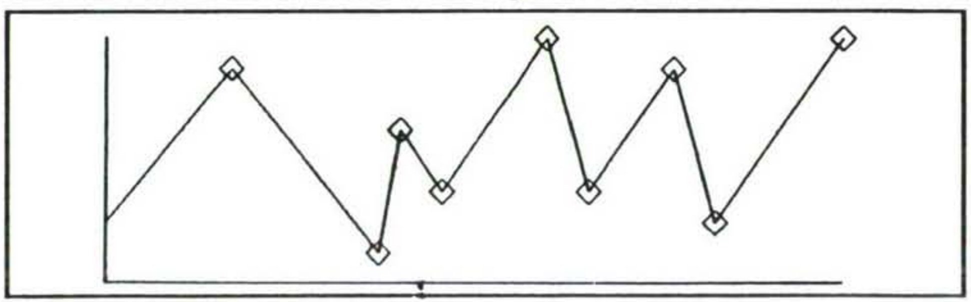
Data Condensation For Response Parameters

Pre-Condensation Time History:



Condensation Threshold

Post-Condensation Time History:



- Stresses
- Control Surface Deflections
- Acceleration (NZCG)

GA-8762-9
6-21-89

DATA CONDENSATION FOR CORRELATION PARAMETERS

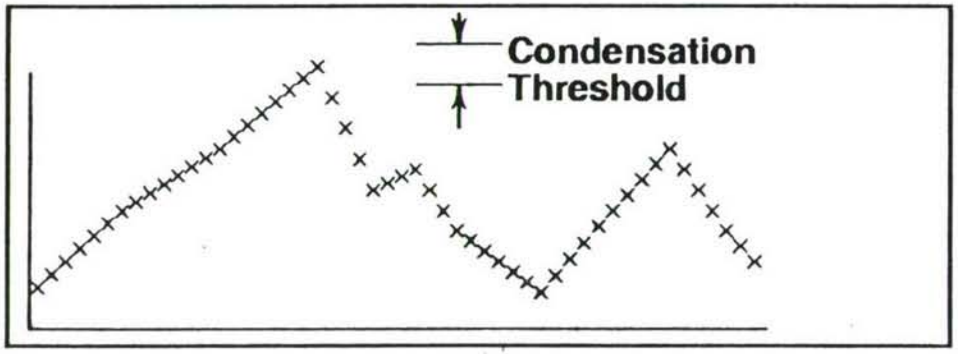
FOR THE REMAINING ANALOG PARAMETERS WHICH ARE NOT SUBSEQUENTLY PEAK COUNTED (LISTED ON THE ADJOINING FIGURE), THE DATA CONDENSATION CONSISTS OF THE ELIMINATION OF POINTS WHICH ARE LESS THAN THE CONDENSATION THRESHOLD PRESCRIBED FOR EACH OF THESE PARAMETERS FROM THE LAST RETAINED POINT FOR THE PARTICULAR PARAMETER. THIS PROCESS IS ILLUSTRATED IN THE ADJOINING FIGURE. THE CONDENSATION THRESHOLDS FOR THESE PARAMETERS ARE: 50 FEET FOR PRESSURE ALTITUDE, 25 FEET FOR RADAR ALTITUDE, 5 KNOTS FOR AIRSPEED, 1 DEGREE FOR FLAP POSITION, .3 KNOTS FOR GROUND SPEED, AND 400 INCH-POUNDS FOR ENGINE TORQUE.

FOR THE DISCRETE (ON/OFF) PARAMETERS OF TOUCHDOWN SWITCH, RAMP DOOR POSITION, AERIAL REFUELING CONNECTION, AND RADAR ALTITUDE FLAG (VALIDITY PARAMETER), REDUNDANT "ON" OR "OFF" VALUES MONITORED BETWEEN CHANGES ARE ELIMINATED. THIS SIMPLE PROCESS IS NOT ILLUSTRATED.

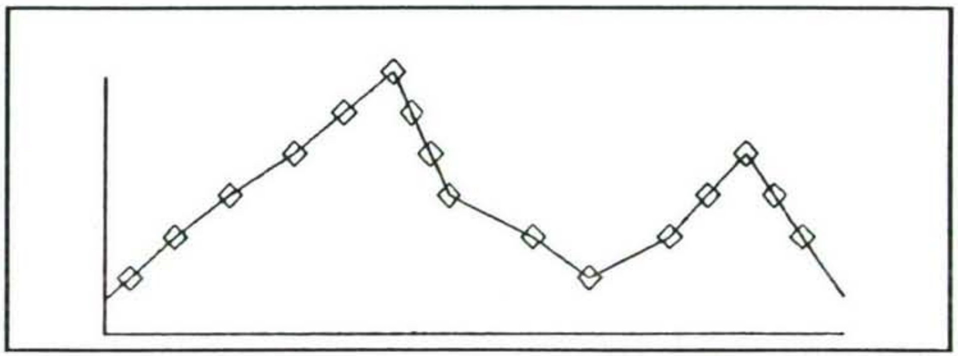
C-130 LESS

Data Condensation For Correlation Parameters

Pre-Condensation Time History:



Post-Condensation Time History:



- Pressure Altitude
- Radar Altitude
- Airspeed
- Flap Position
- Ground Speed
- Engine Torque

GA-8762-10
6-21-89

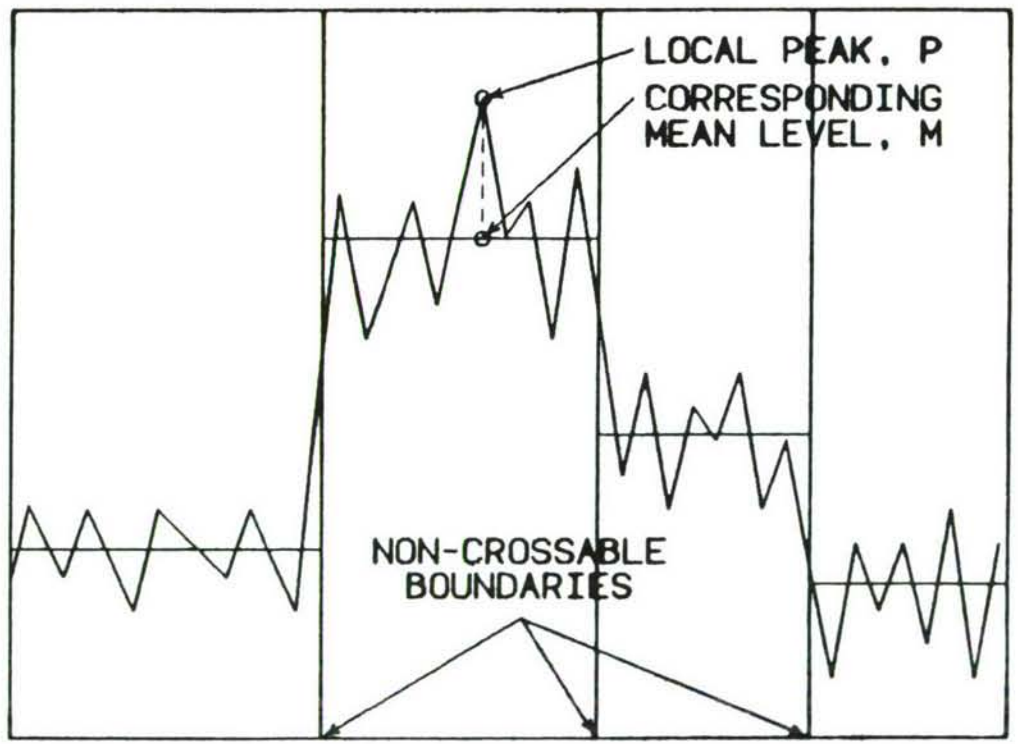
MEAN LEVEL DETERMINATION FOR RESPONSE PARAMETERS

THE PEAK COUNTING METHOD UTILIZED WITHIN THE C-130 LESS REQUIRES THE DETERMINATION OF A MEAN LEVEL FOR THE RESPONSE PARAMETERS ABOUT WHICH THE INCREMENTAL EXCURSIONS ARE OCCURRING. THE MEAN LEVEL IS CALCULATED, AS A VARYING PARAMETER, AT THE TIME OF EACH PEAK/VALLEY AS A LOCAL AVERAGE OF THE TIME HISTORY. THIS MEAN CALCULATION REQUIRES THE HOLDING OF PEAK/VALLEY VALUES IN A BUFFER UNTIL THE MEAN FOR EACH LOCAL PEAK CAN BE CALCULATED.

THE MEAN CALCULATION ALSO MUST RECOGNIZE CHANGES IN THE FLIGHT PROFILE WHICH TYPICALLY REFLECT SHIFTS IN THE STRESS MEAN LEVEL (I.E., FLAP CHANGES, LANDING, ETC.). THESE POINTS OF CHANGE ARE OBTAINED FROM THE PREVIOUSLY DETERMINED FLIGHT PROFILE AND ARE TERMED "NON-CROSSABLE BOUNDARIES." THE MEAN LEVEL COMPUTATION IS RESTARTED AT EACH OF THESE POINTS TO ALLOW FOR THE SHIFT IN MEAN LEVEL.

C-130 LESS

Mean Level Determination For Response Parameters



GA-8762-11
6-21-89

PEAK COUNTING/GUST-MANEUVER SEPARATION FOR RESPONSE PARAMETERS

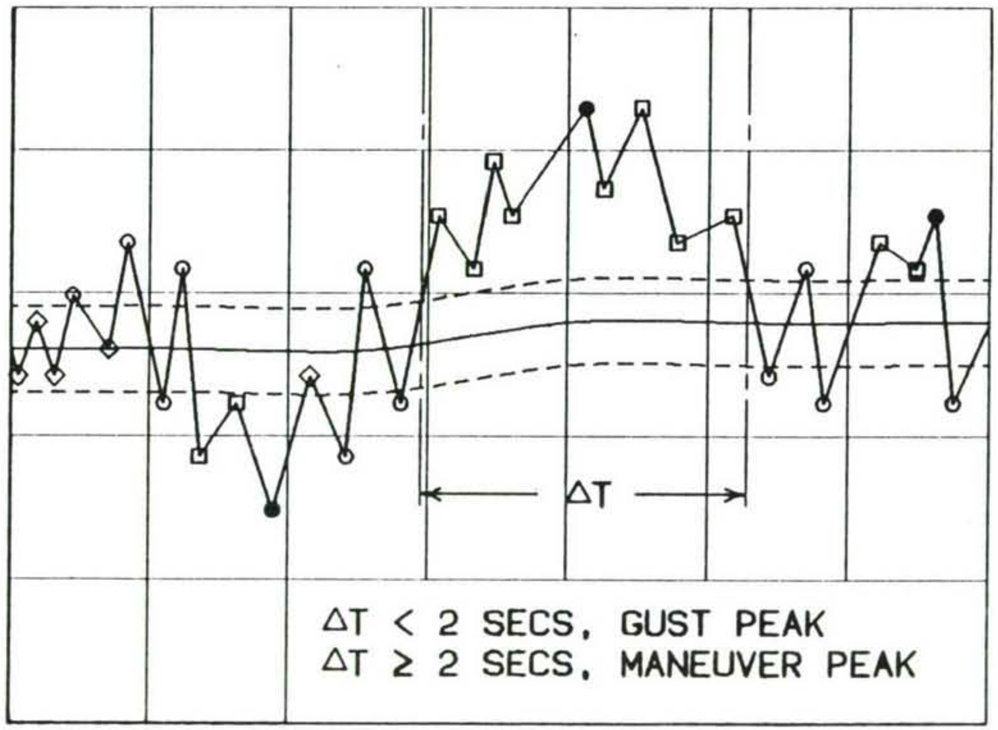
THE PEAK COUNTING PROCESS UTILIZED WITHIN THE C-130 LESS IS A MODIFIED MEAN LEVEL CROSSING METHOD. THEREIN, A PEAK IS DEFINED AS THE MAXIMUM EXCURSION BETWEEN SUCCESSIVE CROSSINGS OF A THRESHOLD LEVEL ESTABLISHED ON EITHER SIDE OF THE LOCAL MEAN LEVEL. THESE THRESHOLD LEVELS ARE SHOWN IN THE ADJOINING FIGURE AS DASHED LINES ON EITHER SIDE OF THE VARIABLE MEAN LEVEL, SHOWN AS A SOLID LINE WITH NO SYMBOLS. THE AREA BETWEEN THE TWO DASHED LINES IS TERMED THE "DEAD BAND". POINTS OUTSIDE THE THRESHOLDS WHICH DO NOT SATISFY THE CRITERIA FOR PEAKS ARE TERMED SECONDARY PEAKS. PEAK COUNTING IS THE SECOND LEVEL OF DATA CONDENSATION ACCOMPLISHED WITHIN THE SFDR IN THAT THE DEAD BAND AND SECONDARY PEAKS ARE ELIMINATED.

THE DETERMINED NZCG AND WING STRESS FINAL PEAKS WHICH OCCUR DURING FLIGHT ARE CLASSIFIED AS GUST OR MANEUVER INDUCED WHILE THE TIME DURATIONS OF THE PEAKS ARE KNOWN. ACCORDING TO CRITERIA BASIC TO C-130 ANALYSIS, NZCG PEAKS WITH A DURATION OUTSIDE THE DEAD BAND LESS THAN 2 SECONDS ARE CLASSIFIED AS GUST; GREATER THAN OR EQUAL TO 2 SECONDS AS MANEUVER. WING STRESS PEAKS ARE CLASSIFIED AS GUST/MANEUVER BASED ON A TIME CORRELATION WITH THE CORRESPONDING NZCG GUST AND MANEUVER PEAKS.

Lockheed

C-130 LESS

Peak Counting/Gust-Maneuver Separation For Response Parameters



GA-8762-12
6-21-89

ABRUPT MEAN SHIFT RECOGNITION FOR RESPONSE PARAMETERS

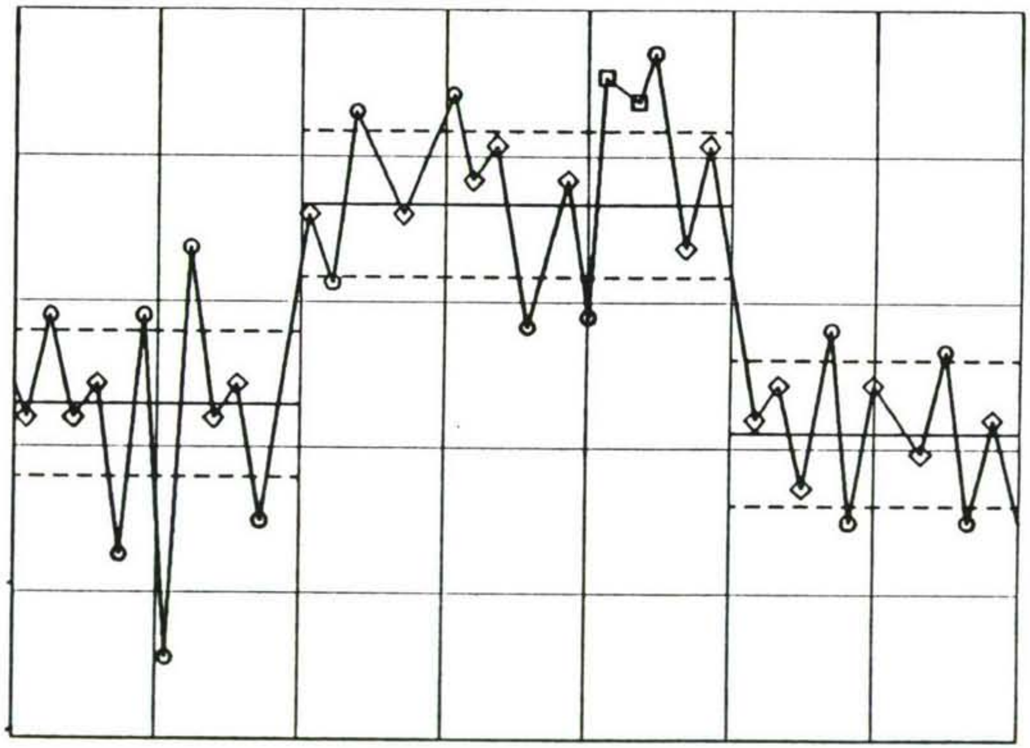
IN ADDITION TO MEAN SHIFTS DUE TO CHANGES IN FLAP CONFIGURATION OR MODE OF OPERATION, THE ONBOARD PROCESSING IS REQUIRED TO RECOGNIZE ABRUPT MEAN SHIFTS DUE TO EXTENDED MANEUVERS. NZCG AVERAGE VALUES ARE MONITORED FOR PRESCRIBED MAGNITUDE CHANGES WHICH EXCEED PRESCRIBED DURATIONS. THE DEPARTURE POINTS AND RETURN POINTS ARE TERMED "BREAKPOINTS". THE CALCULATED MEAN LEVELS ARE RESTARTED AT THESE BREAKPOINTS SIMILARLY AS AT FLIGHT PROFILE CHANGES TO ALLOW FOR THE SHIFT OF THE MEAN IN THE PEAK COUNTING PROCESS.

AS SHOWN IN THE ADJOINING FIGURE, INCREMENTAL EXCURSIONS WHICH OCCUR DURING THE EXTENDED MANEUVER ARE COUNTED ABOUT THE DISPLACED MEAN LEVEL. IN ADDITION, A MANEUVER PEAK EQUAL TO THE MAGNITUDE BETWEEN THE PRIOR AND DISPLACED MEAN LEVEL IS ADDED TO THE SUBSEQUENT MANEUVER PEAK SPECTRA DURING GROUND PROCESSING.

Lockheed

C-130 LESS

Abrupt Mean Shift Recognition For Response Parameters



GA-8762-13
6-21-89

CONDENSED TIME HISTORY RETENTION

THE RESULTS OF THE ONBOARD PROCESSING, PREVIOUSLY DESCRIBED, ARE CONDENSED TIME HISTORIES OF EACH OF THE MONITORED PARAMETERS. THESE TIME HISTORIES ARE STORED IN MASS MEMORY FOR SUBSEQUENT DOWNLOADING, GROUND PROCESSING, AND ANALYSIS.

THE PEAK COUNTED PARAMETERS (NZCG, STRESSES, AND CONTROL SURFACE DEFLECTIONS) RETAINED HISTORIES ARE COMPRISED OF FINAL PEAKS, THE TIME OF THE PEAKS, THE MEAN LEVEL VALUE AT THE TIME OF THE PEAK AND, WHERE APPLICABLE, A GUST/MANEUVER IDENTITY.

THE RETAINED HISTORIES OF THE NON-PEAK COUNTED PARAMETERS CONSIST OF THE CONDENSED DATA POINTS, AS PREVIOUSLY DESCRIBED, AND THE TIME OF THE POINTS.

PEAKS OR DATA POINTS OCCURING WITHIN .125 SECOND TOLERANCES ARE STORED WITH A COMMON TIME VALUE, AND MEAN LEVELS ARE "BLOCKED" IN PRESCRIBED LIMITS TO FURTHER CONSERVE MEMORY.

THE RESULTING FORM OF RETAINED DATA OPTIMIZES MASS STORAGE WHILE COMPLYING WITH SFDR COMMONALITY REQUIREMENTS AND PROVIDING THE DATA ANALYST WITH ADEQUATE DATA VISIBLITY TO VERIFY THE QUALITY OF THE RECORDED DATA.

Lockheed

C-130 LESS

Condensed Time History Retention

- Peak Counted Parameters
  - Final Peaks
  - Time
  - Mean
  - Gust-Maneuver ID
- Non-Peak Counted Parameters
  - Condensed Points
  - Time

GA-8762-14
6-21-89

PROTOTYPE RECORDER/MAINFRAME INTERFACE

THE C-130 LESS PROTOTYPE RECORDER/GROUND SYSTEM INTERFACE NETWORK IS DEPICTED IN THE ADJOINING FIGURE. THIS NETWORK WAS ESTABLISHED FOR THE TRANSFER OF THE PROTOTYPE LESS DATA FROM THE INSTRUMENTED AIRCRAFT, PRODUCTION OF FLOPPY DISKS AND DATA HEALTH INFORMATION, DATA DECOMPRESSION, AND UPLOAD OF THE DATA INTO THE LASC-GA MAINFRAME FOR FURTHER PROCESSING AND ANALYSIS.

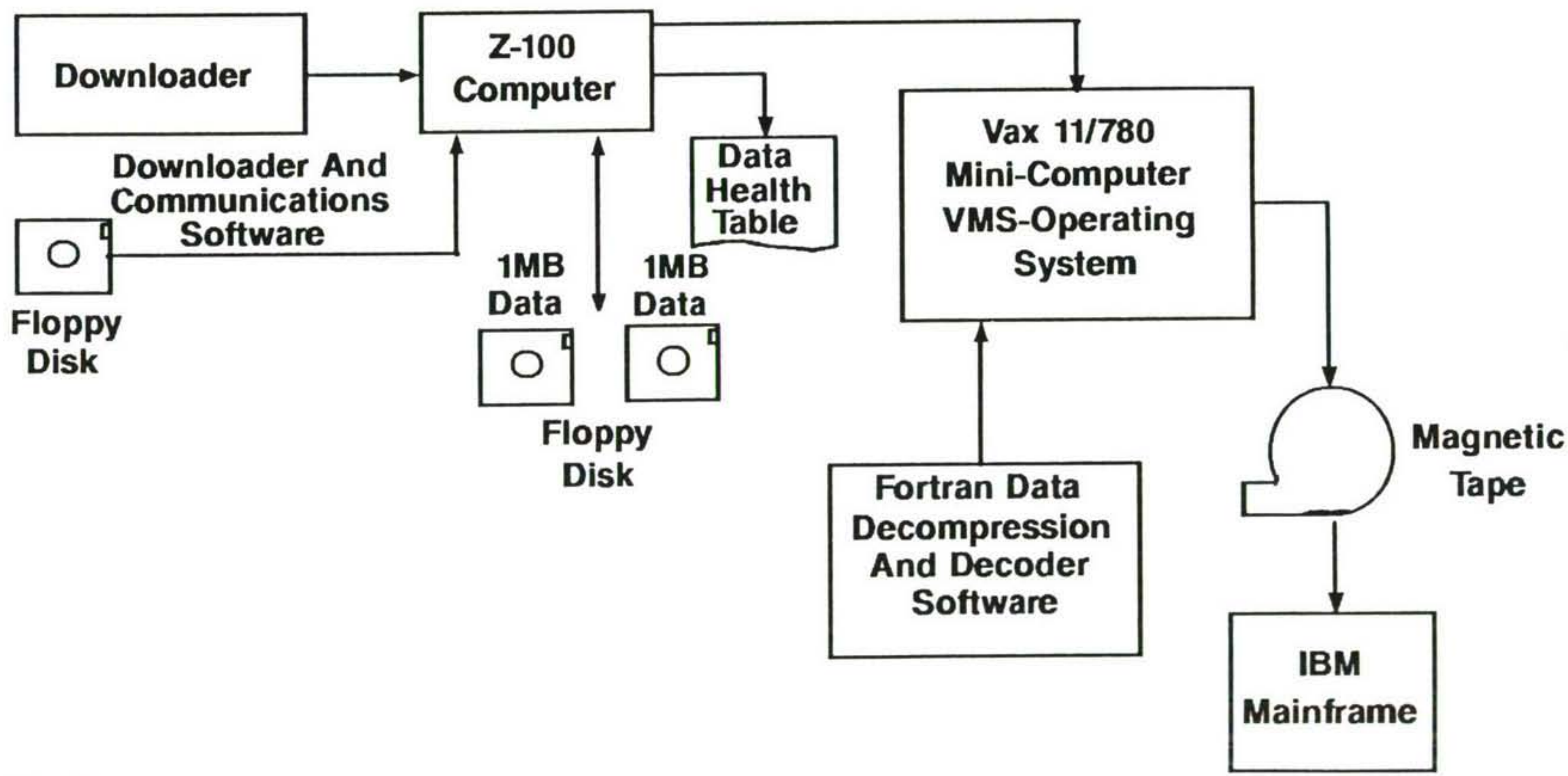
THE Z-100 ELEMENT REPRESENTS THE INDIVIDUAL OPERATING BASE OPERATIONS WHERE MAINTENANCE DATA IN THE FORM OF DATA HEALTH PRINTOUT WILL BE PRODUCED. THE VAX ELEMENT IS THE PROTOTYPE PROGRAM EQUIVALENT OF THE SFDR DECOMPRESSION OPERATION WHERE THE PACKED DATA ARE DECODED AND REFORMATTED IN A COMMON FORMAT. THE C-130 LESS GROUND PROCESSING COMPUTER PROGRAM NETWORK, TO BE OPERATED IN THE PRODUCTION MODE BY OKLAHOMA CITY AIR LOGISTICS CENTER, OC-ALC, HAS BEEN DEVELOPED ON THE LASC-GA IBM SYSTEM IN A FORM COMPATIBLE WITH THE OC-ALC COMPUTING SYSTEM.

THIS PROTOTYPE RECORDER/GROUND SYSTEM INTERFACE WILL BE REPLACED BY THE SFDR DOWNLOADING EQUIPMENT AND DECOMPRESSION SOFTWARE. THE BASE OPERATION'S COMPUTERS WILL READ THE SFDR BIT DATA ON FLOPPY DISKS FOR DATA HEALTH INFORMATION DURING PRODUCTION OPERATIONS.

Lockheed

C-130 Less

Prototype Recorder/Mainframe Interface



GA-8762-15
6-21-89

STANDARD FLIGHT DATA RECORDER (SFDR)

THE C-130 LESS PRODUCTION PROGRAM WILL UTILIZE THE STANDARD FLIGHT DATA RECORDER (SFDR), BEING DEVELOPED BY SMITHS INDUSTRIES OF GRAND RAPIDS, MICHIGAN, FOR ONBOARD DATA ACQUISITION, PROCESSING, AND RETENTION. LASC-GA, UNDER CONTRACT TO WARNER ROBINS AIR LOGISTICS CENTER (WR-ALC), HAS BEEN WORKING WITH SMITHS INDUSTRIES SINCE JULY 1988 TO INCORPORATE THE FUNCTIONS AND PROCESSES DEVELOPED UNDER THE PROTOTYPE PROGRAM INTO THE SFDR SYSTEM. A PRE-PRODUCTION SFDR SET OF EQUIPMENT WILL BE INSTALLED IN THE C-130 PROTOTYPE LESS AIRCRAFT IN PARALLEL WITH THE PROTOTYPE EQUIPMENT IN FEBRUARY 1990. A DUAL OPERATION OF THE SYSTEMS WILL FOLLOW TO VERIFY AND ADJUST THE SFDR OPERATIONS.

THE SFDR MODEL 2865E1 SIGNAL DATA RECORDER - REPRODUCER (SDRR) AND MODEL 3446A FLIGHT DATA PANEL (FDP) UNITS WILL BE UTILIZED FOR THE C-130 LESS APPLICATION. SOME PERTINENT FEATURES OF THE SFDR FOR THE C-130 APPLICATION ARE:

- 2 MEGABYTES OF MASS MEMORY FOR CONDENSED DATA STORAGE
- BUILT-IN-TESTS (BIT) FOR SYSTEM VALIDITY/FAILURE VISIBILITY
- ESTIMATED APPROXIMATELY 30 FLIGHT HOUR AVERAGE DATA DOWNLOAD INTERVAL
- FDP UTILIZED FOR PRE-FLIGHT, POST-FLIGHT, AND LIMITED INFLIGHT MANUAL INPUTS
- "80% AND 100% FULL" LIGHTS ON FDP TO ALERT CREW OF NEED FOR DATA DOWNLOADING

Lockheed

C-130 LESS

Standard Flight Data Recorder (SFDR)

- 2 Megabyte Memory
- Built-In-Tests (BIT)
- Average Download Interval of 30 Flight Hours
- Flight Data Panel (FDP)
- "80% and 100% Full" Lights

GA-8762-16
6-21-89

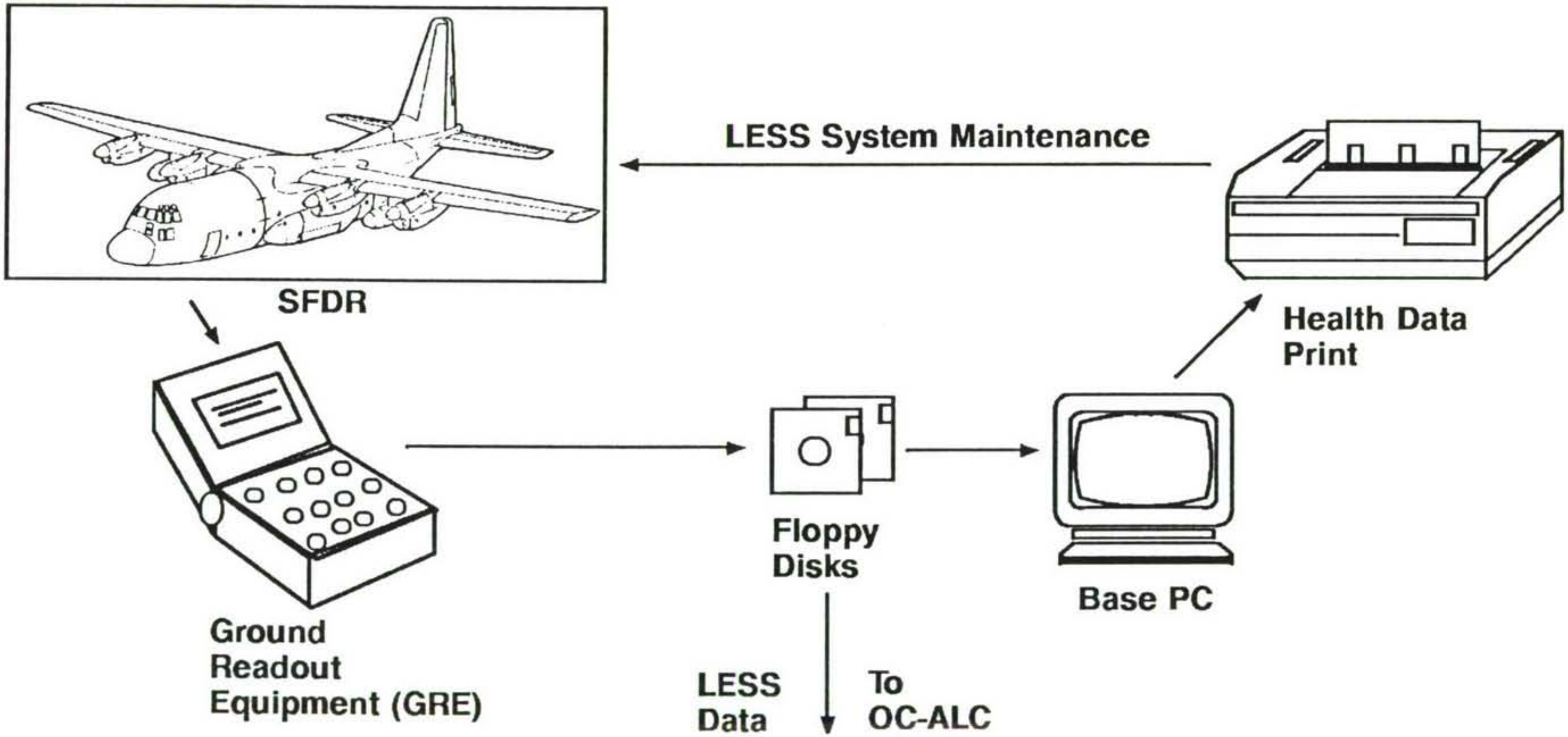
PRODUCTION LESS DATA FLOW - AT OPERATIONAL BASES

THE DESIGNED C-130 LESS DATA FLOW FROM AIRCRAFT THROUGH ENGINEERING ANALYSIS IS SHOWN IN GENERAL FORM IN THE NEXT THREE FIGURES. THE ADJOINING FIGURE ILLUSTRATES THE DATA FLOW AT EACH OF THE C-130 OPERATIONAL BASES AT WHICH A LESS EQUIPPED C-130 WILL BE STATIONED (75 C-130 AIRCRAFT TO BE INSTRUMENTED; STATIONED AT APPROXIMATELY 50 BASES).

THE RECORDED DATA WITHIN THE MASS MEMORY OF THE ONBOARD SFDR WILL BE PERIODICALLY DOWNLOADED (AVERAGE TWICE MONTHLY) BY USE OF THE SFDR GROUND READOUT EQUIPMENT (GRE). FLOPPY DISKS CONTAINING THE BIT AND CONDENSED TIME HISTORY DATA WILL BE PRODUCED BY THE GRE. THE DATA WILL BE ENTERED INTO A BASE PERSONAL COMPUTER FOR GENERATION OF A HEALTH DATA PRINTOUT DERIVED FROM THE BIT DATA. THE FLOPPY DISKS WILL BE COPIED FOR TEMPORARY DATA BACK-UP AND TRANSMITTED BY MAIL TO OC-ALC FOR PROCESSING.

C-130 LESS

Production
LESS Data Flow
Operational Base



GA-8762-17
6-21-89

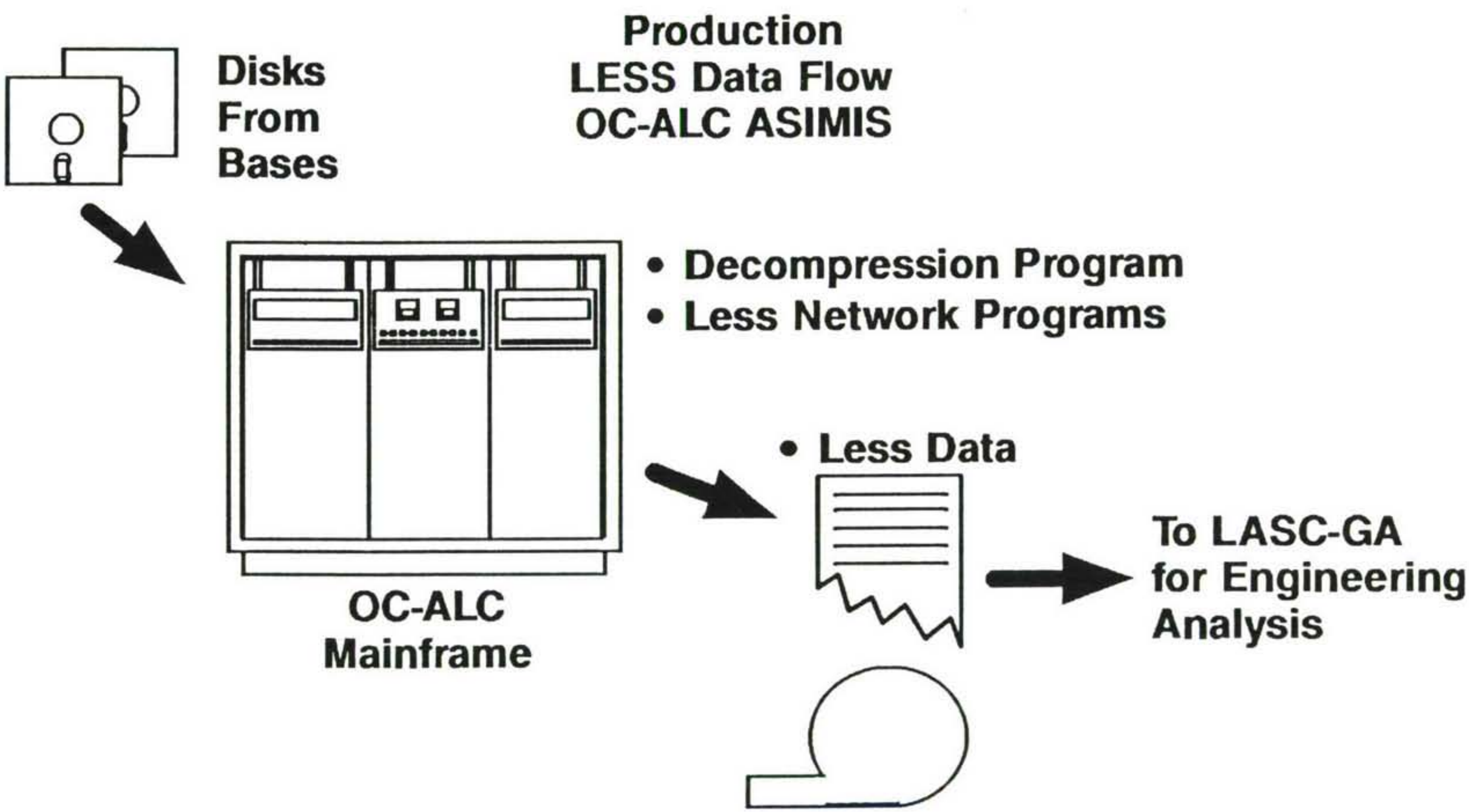
PRODUCTION LESS DATA FLOW - AT OC-ALC ASIMIS

THE ADJOINING FIGURE ILLUSTRATES THE GENERAL DATA FLOW OF THE C-130 LESS DATA AT OKLAHOMA CITY - AIR LOGISTICS CENTER (OC-ALC) WITHIN THE AIRCRAFT STRUCTURAL INTEGRITY MANAGEMENT INFORMATION SYSTEM (ASIMIS).

THE DISKS RECEIVED FROM THE VARIOUS C-130 OPERATIONAL BASES ARE ENTERED INTO THE OC-ALC COMPUTER SYSTEM. THE FIRST EXECUTION OF PROCESSING IS THE DECOMPRESSION/REFORMATTING OF THE DATA INTO A STANDARD FORMAT THROUGH THE USE OF A GENERIC SFDR DECOMPRESSION PROGRAM DEVELOPED BY SMITHS INDUSTRIES. THIS GENERIC DECOMPRESSION PROGRAM WILL BE UTILIZED FOR ALL USAF SFDR AIRCRAFT APPLICATIONS. FOLLOWING DECOMPRESSION, THE C-130 LESS DATA WILL BE PROCESSED BY OC-ALC UTILIZING A SERIES OF COMPUTER PROGRAMS DEVELOPED BY LASC-GA SPECIFICALLY FOR THE C-130 LESS SYSTEM (THE C-141 LESS PROGRAM NETWORK WILL BE VERY SIMILAR). THIS LESS NETWORK CONSISTS OF EIGHT COMPUTER PROGRAMS WHICH TRANSLATE THE STANDARD DECOMPRESSED DATA FORM TO THE FORM REQUIRED BY THE LESS NETWORK, PERFORM FURTHER DATA QUALITY CHECKS, GENERATE PEAK SPECTRA BY INDIVIDUAL FLIGHT SEGMENT, LOAD SOURCE, ETC., ACCUMULATE MEASURED DATA BY AIRCRAFT, MODEL TYPE, CATEGORY, ETC., GENERATE DATA STATISTICS AND OPERATING TECHNIQUE INFORMATION, AND CALCULATE ANALYTICAL STRESS EXPERIENCE BASED ON AIRCRAFT USAGE.

VARIOUS FORMS OF THESE DATA PRODUCTS ARE PERIODICALLY TRANSMITTED TO LASC-GA FOR FINAL ENGINEERING ANALYSIS.

C-130 LESS



GA-8762-18
6-21-89

PRODUCTION LESS DATA FLOW - AT LASC-GA

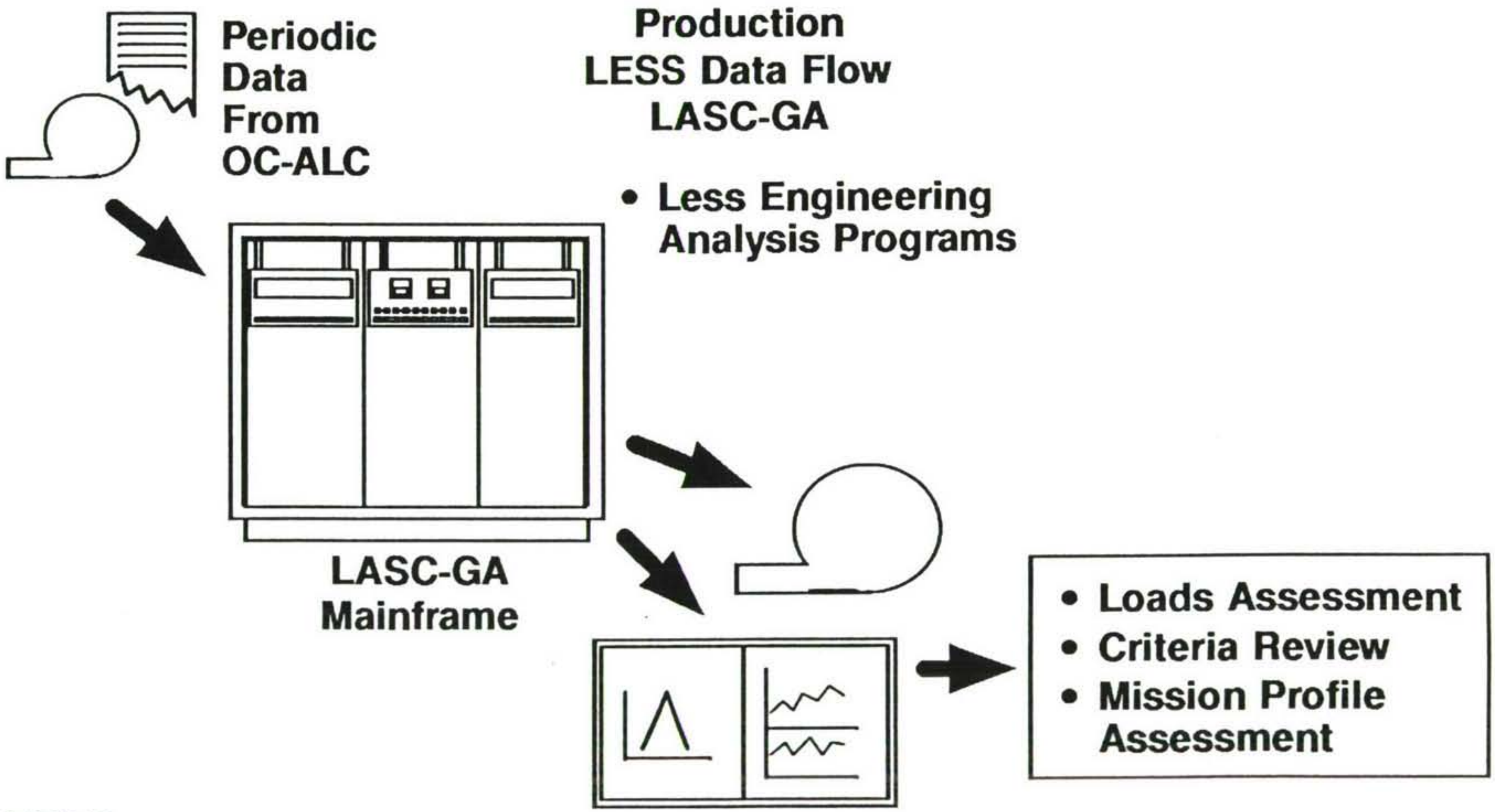
THE ADJOINING FIGURE ILLUSTRATES THE LESS DATA FLOW IN GENERAL FORM THROUGH THE ENGINEERING ANALYSIS OPERATIONS AT LASC-GA.

THE PROCESSED LESS DATA PERIODICALLY RECEIVED FROM OC-ALC ARE ENTERED INTO THE LASC-GA COMPUTER SYSTEM FOR ANALYSIS. THE MEASURED AND CORRESPONDING ANALYTICAL BASED DATA ARE COMPARED TO EACH OTHER AND TO BASELINE DATA UPON WHICH THE C-130 INDIVIDUAL AIRCRAFT TRACKING PROGRAM (IATP) ANALYSES ARE BASED. DATA ARE EVALUATED TO DETERMINE IF CHANGES ARE OCCURRING IN THE AIRCRAFT LOADING ENVIRONMENT, THE CAUSES FOR SUCH CHANGES AND THE IMPACT UPON STRUCTURAL TRACKING AND AIRCRAFT FORCE MANAGEMENT.

THE LESS RESULTS ARE PROVIDED TO IATP FUNCTIONS AT LASC-GA FOR LOADS ASSESSMENT, LOADING CRITERIA REVIEW/ADJUSTMENT, AND MISSION PROFILE REPRESENTATION ASSESSMENT.

RESULTS OF THE LESS DATA ANALYSES AND ASSESSMENTS ARE DOCUMENTED AND PUBLISHED IN PERIODIC ENGINEERING REPORTS.

C-130 LESS



GA-8762-19
6-21-89

CONDENSED TIME HISTORY - STRESS AT WING STATION 95

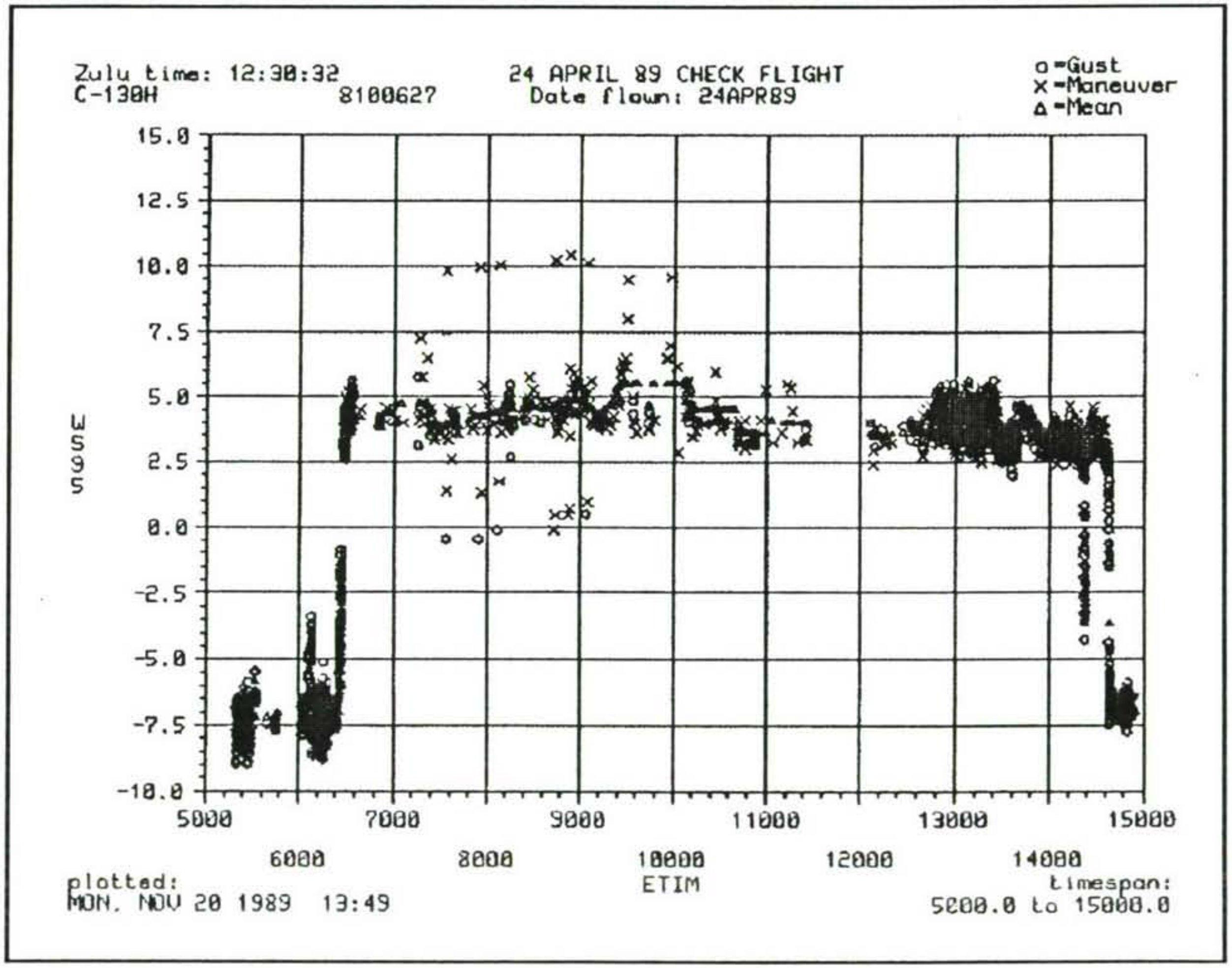
THE NEXT THREE FIGURES ILLUSTRATE THREE DIFFERENT FORMS OF THE C-130 LESS DATA. THESE PARTICULAR DATA SAMPLES WERE PRODUCED THROUGH THE PROTOTYPE LESS SYSTEM, HOWEVER, THE SAME DATA FORMS WILL BE PRODUCED IN PRODUCTION OPERATIONS WITH THE SFDR.

THE ADJOINING FIGURE SHOWS A CONDENSED TIME HISTORY OF THE STRESS PARAMETER MEASURED ON THE LOWER AFT BEAM OF THE LEFT WING AT WING STATION 95. THIS DATA SAMPLE WAS RECORDED ON THE PROTOTYPE LESS EQUIPPED AIRCRAFT, SERIAL NUMBER 81-0627 STATIONED AT DOBBINS AFB, DURING A LESS SYSTEM CHECK FLIGHT ON 24 APRIL 1989. THE TIME HISTORY OF FINAL PEAKS, MEAN VALUES, AND GUST/MANEUVER IDENTIFICATIONS IS PLOTTED IN COMPRESSED FORM TO SHOW THE HISTORY DURING THE ENTIRE FLIGHT. THIS IS THE FORM OF THE RESPONSE PARAMETER DATA TO BE PRODUCED ONBOARD BY THE SFDR. THE LARGER EXCURSIONS BETWEEN ELASPED TIME (ETIM) OF 7000 AND 10,000 SECONDS ARE THE RESULT OF ROLLER COASTER AND BANKED TURN MANEUVERS ACCOMPLISHED TO EXERCISE THE LESS PROTOTYPE SYSTEM LOGIC. THIS DATA SAMPLE WAS SELECTED FOR ILLUSTRATION TO DEMONSTRATE THE IMPORTANCE OF CONDENSED TIME HISTORY RETENTION. EXAMINATION OF THIS TIME HISTORY REVEALED TWO PROBLEMS WITHIN THE SYSTEM WHICH OTHERWISE, WITHOUT THIS VISIBILITY, WOULD BE DIFFICULT TO ISOLATE. THE INDICATED GUST PEAKS, SHOWN AS O, BETWEEN 7500 AND 8100 SECONDS AND LESS THAN 0.0 PSI, WERE IN FACT MANEUVER INDUCED. A FLAW IN THE GUST-MANEUVER LOGIC WHICH OCCURRED FOR NZCG VALUES LESS THAN 0 G'S WAS THE CAUSE. SECONDLY, THE ABSENCE OF DATA BETWEEN 11,500 AND 12,000 SECONDS IS OBVIOUS. THIS DATA VOID WAS CAUSED BY AN OVERFLOW PROBLEM WITHIN THE PROTOTYPE SYSTEM PROCESSING.

Lockheed

C-130 LESS

Condensed Time History
Stress-Wing Station 95



GA-8762-20
6-21-89

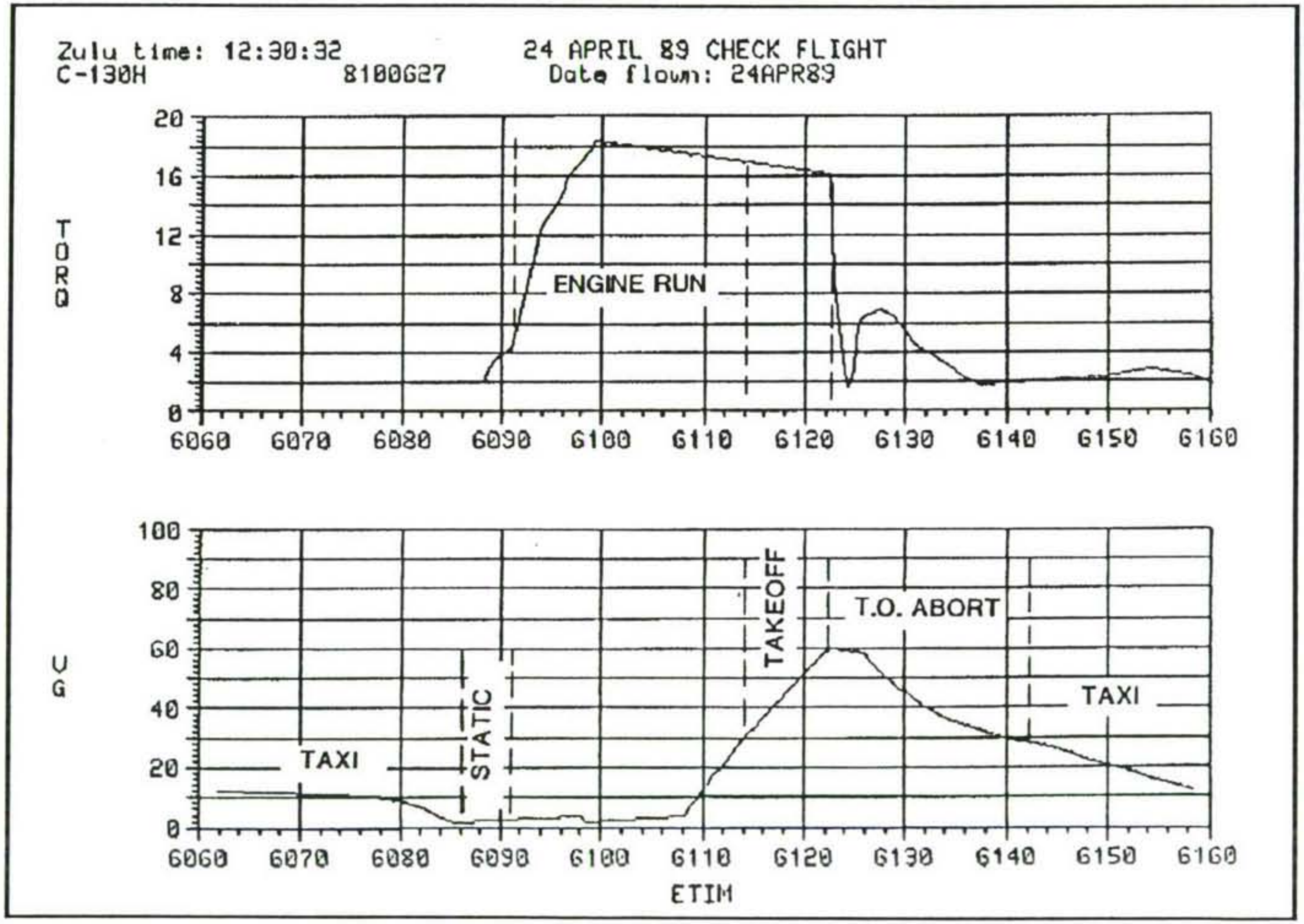
CONDENSED TIME HISTORIES - ENGINE TORQUE AND GROUND SPEED

THE ADJOINING FIGURE ILLUSTRATES THE ONBOARD PRODUCED CONDENSED FORM OF TWO OF THE CORRELATION PARAMETERS, NO. 1 ENGINE TORQUE AND GROUND SPEED. THESE TWO PARAMETERS ARE UTILIZED WITHIN THE ONBOARD PROCESSING IN THE DETERMINATION OF THE FLIGHT PROFILE. THE DISPLAYED DATA SAMPLE IS FROM A PORTION OF THE 24 APRIL 1989 CHECK FLIGHT OF THE PRECEEDING FIGURE. FOR THE CORRELATION PARAMETERS, THE CONDENSED DATA POINTS ARE CONNECTED BY STRAIGHT LINES IN THE GROUND GRAPHICS ROUTINE WHICH PRODUCED THIS FIGURE. THE VALUES OF THESE PARAMETERS ARE TESTED AGAINST CRITERIA WITHIN THE ONBOARD SYSTEM TO DIVIDE THE FLIGHT INTO ITS APPROPRIATE SEGMENTS. AS SHOWN, THE DATA SAMPLE DISPLAYS A SEQUENCE OF TAXI, STATIC, ENGINE RUN, TAKEOFF ROLL, TAKEOFF ABORT, AND TAXI.

Lockheed

C-130 LESS

Condensed Time Histories
Engine Torque And Ground Speed



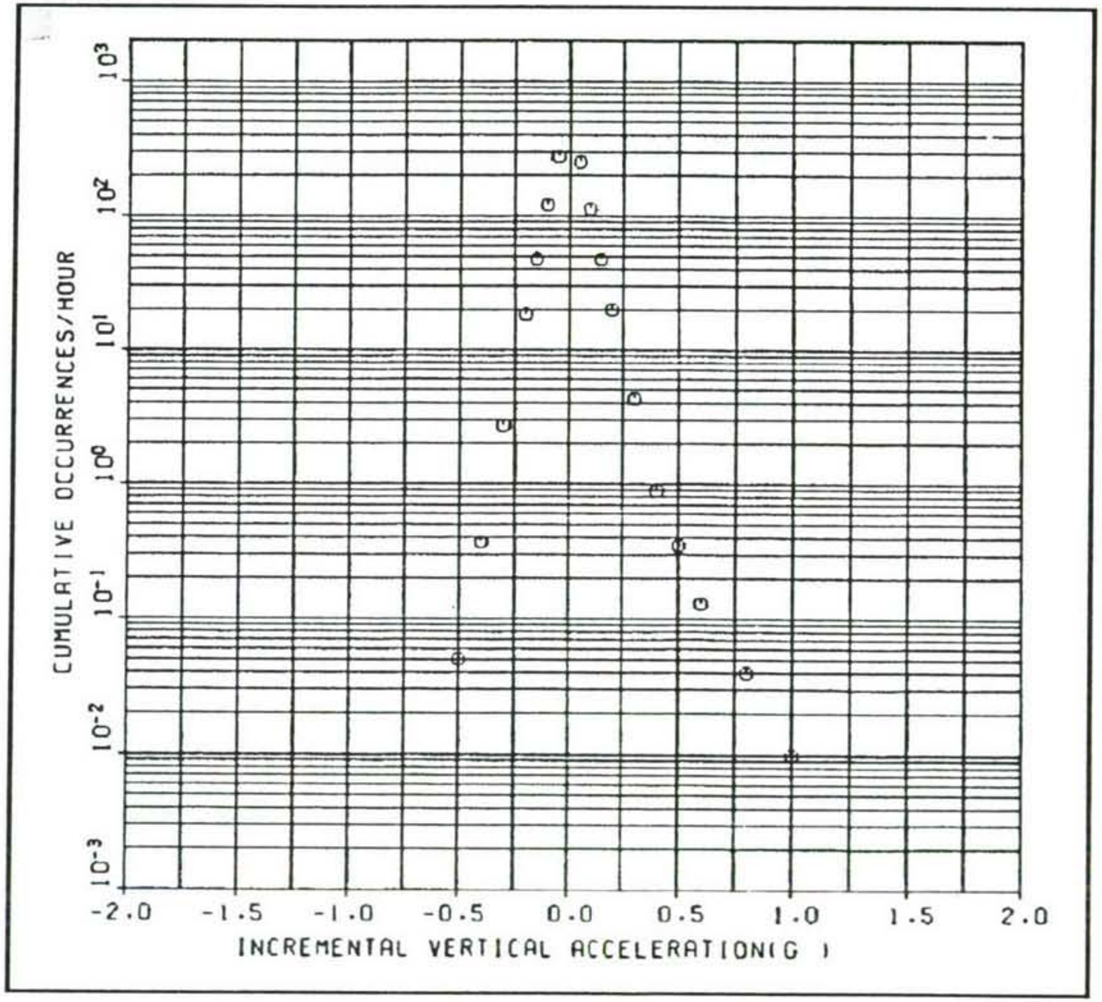
GA-8762-21
6-21-89

CUMULATIVE OCCURRENCE SPECTRUM
NZCG - TOTAL AIR - ALL MISSIONS - ALL ALTITUDES

THE ADJOINING FIGURE ILLUSTRATES THE CUMULATIVE INCREMENTAL OCCURRENCE SPECTRA FORM OF THE RESPONSE PARAMETERS AS PRODUCED WITHIN THE GROUND BASED LESS COMPUTER PROGRAM NETWORK. THIS CLASSICAL FORM OF REPRESENTATION OF STATISTICAL CYCLIC DATA IS UTILIZED FOR THE COMPARISONS OF LOADING EXPERIENCE WITHIN THE C-130 LESS DATA ANALYSIS. THE DATA SAMPLE PRESENTED WAS OBTAINED FROM PROCESSED PROTOTYPE C-130 LESS DATA AND REPRESENTS NZCG PEAK OCCURRENCES FROM ALL PORTIONS OF INFLIGHT OPERATIONS (TOTAL AIR) DURING ALL MISSION TYPES AND AT ALL ALTITUDES FOR APPROXIMATELY 100 HOURS OF OPERATION. THE LESS GROUND PROCESSING SYSTEM SEGREGATES AND PRODUCES GRAPHIC PLOTS OF SUCH SPECTRA FOR ANY OF THE APPROPRIATE PARAMETERS FOR ANY COMBINATION OF FLIGHT SEGMENT, MISSION TYPE, ALTITUDE, AIRCRAFT SERIAL NUMBER/MODEL TYPE/FLEET, OR TIME FRAME. THIS PLOT IS PRESENTED FOR ILLUSTRATION ONLY AND IS NOT INTENDED TO REFLECT A STABLE, STATISTICAL DESCRIPTION OF C-130 LOADING EXPERIENCE.

C-130 LESS

Cumulative Occurrence Spectrum
NZCG-Total Air-All Missions-All Altitudes



Based On 100 Hours Of Accumulated Data

GA-8762-22
6-21-89

CURRENT STATUS

THE PROTOTYPE ONBOARD LESS SYSTEM IS OPERATING ON A SINGLE C-130H AIRCRAFT, 81-0627, AT DOBBINS AIR FORCE BASE, GEORGIA, ASSIGNED TO THE 700TH TACTICAL AIRLIFT SQUADRON, AFRES. INITIAL INSTALLATIONS WERE ACCOMPLISHED IN JUNE 1986 AND HAVE SINCE BEEN UPDATED AS REQUIRED. TO DATE 310 HOURS OF FINAL FORM PROTOTYPE DATA FROM 185 INDIVIDUAL FLIGHTS HAVE BEEN ACQUIRED THROUGH 28 SEPARATE DOWNLOADS. NUMEROUS OTHER DOWNLOADS HAVE BEEN ACCOMPLISHED ACQUIRING DATA IN A PRELIMINARY UNCONDENSED "RAW" DATA FORM FOR TROUBLESHOOTING AND DEVELOPMENT PURPOSES. THE RECORDING SYSTEM HAS NOT BEEN CONTINUOUSLY INSTALLED DUE TO REMOVAL FOR MODIFICATION AND FIRMWARE CORRECTION AND REVISION.

THE LESS GROUND PROCESSING NETWORK HAS BEEN DEVELOPED AND OPERATED AT LASC-GA WITH THE PROTOTYPE LESS DATA. PRELIMINARY LESS PROGRAMS HAVE BEEN DELIVERED TO OC-ALC WITH PROTOTYPE DATA FOR ESTABLISHMENT OF THE OC-ALC PROCESSING FRAMEWORK AND FAMILIARIZATION WITH C-130 LESS PROCESSING OPERATIONS.

THE TRANSITION FROM THE PROTOTYPE TO PRODUCTION SFDR SYSTEM HAS BEEN UNDERWAY SINCE JULY 1988. THE C-130 SFDR CRITICAL DESIGN REVIEW WAS CONDUCTED IN SEPTEMBER 1989. THE LESS WIRING ON THE PROTOTYPE EQUIPPED AIRCRAFT WAS MODIFIED IN NOVEMBER 1989 TO ACCEPT THE SFDR PRE-PRODUCTION EQUIPMENT FOR DUAL OPERATION WITH THE PROTOTYPE SYSTEM (COMMON SIGNAL SOURCES/INDEPENDENT PROCESSING) WHICH WILL BEGIN IN FEBRUARY 1990.

Lockheed

C-130 LESS

Current Status

- Prototype System on C-130H A/C 81-0627
- Acquiring Operational Data
- Evaluations/Adjustments Ongoing
- Ground Processing Network in Evaluation/Adjustment Mode
- Transition to Production System Underway

GA-8762-23
6-21-89

PROTOTYPE TO PRODUCTION TRANSITION SCHEDULE

THE TRANSITION BEGAN IN JULY 1988 AND HAS CONSISTED OF AN INTENSE EFFORT BY WR-ALC, ASD, SMITHS INDUSTRIES, AND LASC-GA TO INCORPORATE THE REQUIREMENTS, PROCESSES, AND METHODS DEVELOPED UNDER THE LASC-GA PROTOTYPE PROGRAM INTO THE SMITHS INDUSTRIES DEVELOPED SFDR SYSTEM. NUMEROUS JOINT MEETINGS, ACTION ITEMS, AND INNOVATIVE APPROACHES HAVE BEEN REQUIRED TO RESOLVE THE SPECIFIC REQUIREMENTS OF THE C-130 LESS WHILE MAINTAINING THE STANDARDIZATION MANDATES OF THE SFDR. TWO ENGINEERING CHANGE PROPOSALS (ECP) TO THE CORE SFDR HAVE BEEN REQUIRED TO MEET THE 2 MEGABYTE MASS MEMORY AND THE SPECIAL LESS PROCESSING REQUIREMENTS FOR THE C-130. THE C-141 AND PROBABLY OTHER SFDR APPLICATIONS WILL DIRECTLY BENEFIT FROM THESE ECPs. OTHER DIFFERENCES BETWEEN PROTOTYPE AND SFDR FUNCTIONALITY HAVE BEEN RESOLVED THROUGH COMPROMISE AND INNOVATION WITHOUT DEGRADATION TO THE DATA FORMS OR VISIBILITY.

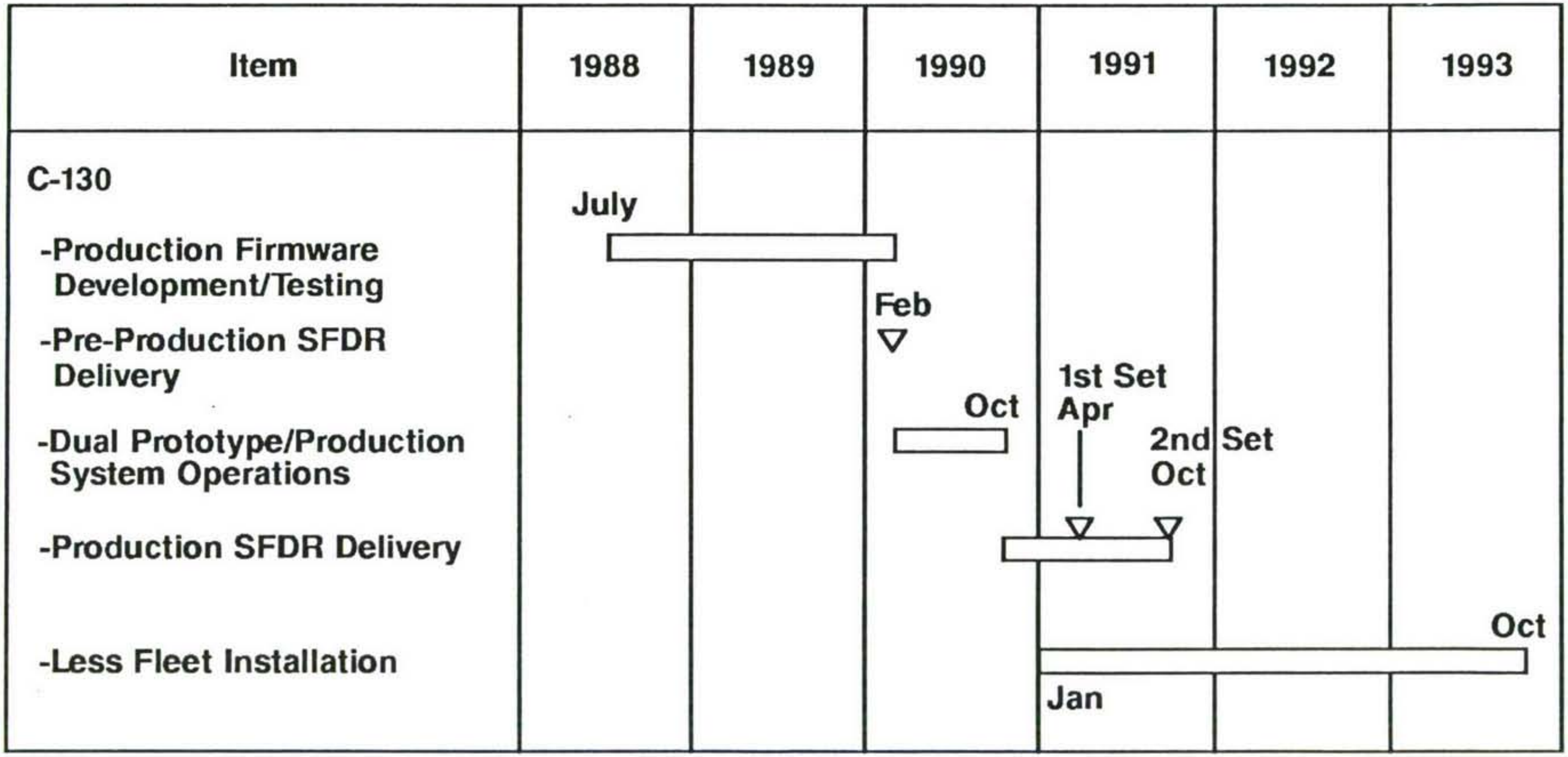
AN EIGHT MONTH DUAL OPERATION OF THE PROTOTYPE AND PRE-PRODUCTION SFDR SYSTEMS ON C-130H 81-0627 WILL BEGIN IN FEBRUARY 1990 FOR THE PURPOSE OF VERIFYING SFDR FUNCTIONALITY UNDER OPERATIONAL CONDITIONS. ADJUSTMENTS TO THE SFDR SHOWN NECESSARY BY RESULTS OF THIS DUAL OPERATION WILL BE RECOMMENDED TO THE AIR FORCE.

PRODUCTION SFDR UNITS DELIVERY IS SCHEDULED FOR COMPLETION IN 1991 AND C-130 USAF FLEET LESS INSTRUMENTATION (75 AIRCRAFT) IS TENTATIVELY SCHEDULED FOR ACCOMPLISHMENT DURING 1991 THROUGH 1993.

Lockheed

C-130 LESS

Prototype-to-Production Transition Schedule



GA-8762-24
6-21-89

SUMMARY

THE REVISED, SFDR BASED C-130 LESS SYSTEM WILL GREATLY ENHANCE THE C-130 FORCE MANAGEMENT BY PROVIDING MORE DATA IN TERMS OF TYPES AND PROJECTED YIELD, MORE CAPABILITY FOR DATA ANALYSIS WITH WORKLOAD RELIEF THROUGH GRAPHICS DISPLAY/PRESENTATION AND OTHER AUTOMATIONS, AND THROUGH FASTER DATA TURN-AROUND TIME IN TERMS OF DATA PROCESSING AND SYSTEM MAINTENANCE REQUIREMENTS INFORMATION.

THE C-130 LESS PROTOTYPE PROGRAM HAS GREATLY BENEFITTED THE FINAL SFDR BASED C-130 LESS SYSTEM BY DEVELOPING AND VERIFYING THE ONBOARD METHODS, PROCESSES, AND OTHER AIRCRAFT SYSTEM INTERFACES AND THE REVISED LESS GROUND PROCESSING NETWORK THROUGH ACQUISITION AND PROCESSING OF OPERATIONAL C-130 DATA. CHANGES AND MODIFICATIONS TO A PROTOTYPE SYSTEM ARE MUCH MORE EASILY ACCOMPLISHED WITH LOWER COST THAN CHANGES TO A DOCUMENTED PRODUCTION SYSTEM.

THE C-130 LESS PROTOTYPE-TO-PRODUCTION EFFORT CAN DIRECTLY BENEFIT THE OTHER AIRCRAFT SFDR APPLICATIONS THROUGH THE EXPANDED SFDR CAPABILITIES ADDED BY THE SFDR ECPs. THESE INCLUDE EXPANDED MEMORY (2 MEGABYTES) AND SEVERAL PROCESSING ADDITIONS SUCH AS MEAN CALCULATION, GUST-MANEUVER SEPARATION, AND EXPANDED DATA REASONABLENESS TESTS (RATE OF CHANGE AND CHATTER TESTS).

FINALLY, THE C-130 LESS/SFDR TRANSITION AND INTERFACE EFFORTS TO DATE BETWEEN SFDR USER AND SFDR PROVIDER ORGANIZATIONS CAN SERVE AS A MODEL FOR OTHER AIRCRAFT APPLICATIONS. THERE HAVE BEEN INCOMPATIBILITIES BETWEEN THE CORE SFDR REQUIREMENTS AND SPECIFIC C-130 LESS REQUIREMENTS AND PROTOTYPE FUNCTIONS. THE RESOLUTION OF THESE DIFFERENCES HAS NOT ALWAYS BEEN EASY, BUT THE COOPERATION AMONG ALL PARTIES TOWARD THE FINAL GOAL, A STANDARDIZED SYSTEM WHICH PROVIDES FOR THE SPECIFIC NEEDS OF THE C-130, HAS BEEN EXEMPLARY.

Lockheed

C-130 LESS

Summary

- Revised, SFDR Based C-130 LESS System Will Greatly Enhance C-130 Force Management
- C-130 LESS Prototype Program Has Greatly Benefitted the Production SFDR Based C-130 LESS
- C-130 LESS Prototype-To-Production Effort Can Directly Benefit Other Aircraft SFDR Applications
- C-130 LESS Transition/Interface Can Serve as a Model for Other Aircraft Applications

GA-8762-25
10-27-89

A-10A SERVICE LIFE MONITORING PROGRAM

Presented at

1989 USAF STRUCTURAL INTEGRITY
PROGRAM CONFERENCE

5-7 December 1989
San Antonio, Texas

Presented By

Herman Axelrod
Grumman Corporation

Kent McPhaul
SM-ALC/MMRSC

(pw-1/52)

The A-10A is a close air support single-place, low wing, low tail configuration with two engines installed in nacelles that are mounted on pylons extending from the fuselage just aft of and above the wing. Twin vertical tails are located at the outboard tips of the horizontal tail. The A-10A is capable of delivery up to 16,000 pounds of expendable ordnance from eleven wing pylon stations. A 30mm Gatling gun with a capacity of 1100 rounds is installed in the forward fuselage Figure 1 presents the major topics of discussion of this paper. The overall dimensions of the A-10A aircraft are presented in Figure 2.

Fairchild and Northrop were selected for a competitive prototype development in December 1970. In October of 1972, a competitive flyoff was held between the Northrop A-9 and the Fairchild A-10. The A-10 was judged to have superior protection against gunfire and missiles and required the least maintenance manhours per flight hour. On January 18, 1973 the Air Force selected the A-10 for production.

In April 1974, a flyoff was conducted between the A-7 and the A-10. The A-7's 20mm gun was ineffective against hard targets such as tanks, whereas the A-10's 30mm gun could destroy tanks and other hard targets. The A-10 attack speed maneuverability delivered weapons with higher overall effectiveness. The A-10 has superior protection against missile and gunfire presenting a vulnerable area from the bottom of approximately 1/16 that of the A-7. Finally, on an equivalent cost basis the A-10 was judged twice as effective as an A-7D.

It is interesting to note that the A-10 was the first USAF Airplane to be designed to a damage tolerance criteria. It was designed to MIL-A-83xxx. This specification later became MIL-A-83444.

The A-10A is monitored through the implementation of the Individual Aircraft Tracking Program (IATP). The IATP consists of counting accelerometers which are installed on every airplane and record the normal load factor (N_z) exceedances for six N_z levels. Also, approximately 84 aircraft contain multi-channel life history recorders which greatly expand the loads/environment parameters being recorded.

The damage tolerance and fatigue spectra were derived from MIL-A-008866 and the air vehicle specification. During the design phase many pertubations were made on the original spectrum based on changes in mission profiles, mission mix and incorporation of operational usage data from counting accelerometers (4750 hours) and multi-channel data (400 hours). Meetings were held with the USAF and a final test spectrum was agreed upon. Figure 3 displays the difference between the design and test spectra. The A-10A successfully completed the durability test.

Early on it was determined that all aircraft do not fly to the same spectrum. Cumulative probabiltiy curves of the number of aircraft

attaining a specific N_z exceedance rate (N_z exceedances per 1000 hours) at a given N_z level were generated from counting accelerometer usage data. An example of the cumulative probability for a given N_z is presented in Figure 4. Again meetings with the USAF concluded that all aircraft flying in the lower 15% of the probability curve would be considered flying a mild spectrum and those in the upper 15% were flying in a severe environment. Between 15% and 85% the aircraft were considered flying in an average environment. The mid-point of each zone, that is, the 7.5%, 50.0% and the 92.5% probability points, were selected at each N_z level and used to develop the tracking spectra shown in Figure 5.

The airplane tracking is based on fracture analysis. 63 points on the A-10 were analyzed. Damage coefficients defined as the average crack growth per N_z occurrence were calculated. The coefficient for each point investigated was a function of the stress spectrum, geometry and material properties which exist at each location. Evaluations of the analyses indicated that the airplane could be tracked by 10 control points - 4 wing, 3 fuselage, 1 nacelle and 2 empennage points. Damage coefficients were computed for mild, average and severe spectra for each control point.

The selection of which spectrum, and therefore which set of damage coefficients to use for each component was made by comparison of data in the counting accelerometer windows. Control points which were N_z critical were related to the 5.5g window. The number of occurrences are normalized to 1000 flight hours and compared to established boundary values of either mild, average or severe usage. Below a certain value of N_z occurrences, 580, the mild spectrum was used; above 1570 N_z occurrences the severe spectrum used, and between the two the average spectrum is used. Other control points were similarly tracked by appropriate N_z occurrences and window.

The airplanes are individually tracked by components. Thus a wing, tail or nacelle could be removed from one airplane and installed on another and still be constantly monitored. In addition, major structural changes , i.e. block changes, are accommodated by calculating a new set of coefficients, installing them and making minor software changes to the program. Figures 6,7 and 8 display the methodology, components and control points.

Software has been written and installed in computers at the Tinker Air Force Base at Oklahoma City. It accepts the previously mentioned damage tolerance coefficients for determining crack growth, Flight logs, counting accelerometer data, N_z occurrences, usage and tail numbers, and any TCTO compliance repairs or design changes. This data is processed as depicted in Figure 9 and results in printed reports of spectrum hours used, spectrum hours remaining, TCTO compliance etc.

Software has been written to evaluate the fleet quarterly. Figure 10 graphically indicates the method of generating statistical spectra from fleet IAT usage. Fleet usage statistics are developed and are used to generate statistical mild, average and severe

spectra, as presented in Figure 11. A base or group of bases are selected for comparison to the statistical spectra. When a match is found, the base or bases therefore provide the mission mix as well as the basis for parameter expansion, such as, gross weight, velocity, N_y etc. (obtained from the MXU mutli-channel recorder).

Multi-channel Loads Environment Spectra Survey (MXU/LESS) data is required to reduce counting accelerometer data to a spectrum for detail stress analysis. This is accomplished by assembling the MXU/LESS data from the same base or group of bases used to determine the mild, average and severe spectra that are used for individual aircraft tracking. A minimum of approximately 150 hours of MXU data is required. If insufficient hours are in the bases selected, the entire process is repeated, i.e. select another group of bases that match the statistical survey and then determine the MXU flight hours.

Raw data exceedance curves are constructed to ensure that the data has not been skewed. The IAT mission mix is consolidated and enforced upon the MXU dataset and new mild, average and severe curves are constructed. These curves are compared to the corresponding IAT curves and the difference at each load factor noted. These differences are later used to enforce the spectrum generated from the MXU dataset to the IAT spectrum. Figure 12 is a pictorial display of the spectrum development.

Two options exist at this point; (1) review the mission mix and mission mix consolidation and proceed to the development of the stress spectrum and damage analysis or (2) add a new (notional) mission to the current mission mix. The notional mission is inserted in the mission mix prior to accomplishing the detail analysis, reference Figure 13.

A notional mission for the A-10A can be developed from the current software and dataset. As seen on Figure 14 a new mission profile is proposed. The mission is defined as seen is Figure 15. All possible stores that can be carried at each pylon station, their weight and drag indices are stored. An example of an aircraft configuration selected from this menu is shown in Figure 16. Similarly, each leg of the proposed mission has a selection menu that describes the type of performance that is desired for each particular leg. An example of the selection is shown in Figure 17. Having configured the airplane and selected each mission leg, the performance program is run to determine whether the mission can be accomplished. Figure 18 is an example of the output.

After verifying the performance adequacy, a structural notional correspondence is equated to each leg of the mission. A menu exists for each selected leg. Allowing the analyst the option of selecting specific usage desired. Figure 19 is an example of these selections. Figure 20 presents the library of structural mission segments that correspond to the 7 mission segment categories. Each

structural mission segment contains a specific load occurrence table for that segment that was developed either from MIL-A-008866B, statistical and/or actual A-10A data. Each load occurrence table contains a statistical compilation of velocity, Qdot, N_z, and N_y to be applied to the gross weight of that particular mission segment.

Once all the performance segments have a corresponding structural segment, the mission load occurrence table can be developed. As seen in the example in Figure 21, this table consists of the gross weight developed from the performance analysis (formatted into gross weight ranges) and the velocity, N_z, N_y, Qdot and occurrences derived from the structural segment library. Exceedance of aircraft structural limits are reported on this table. This data can be used to evaluate the damage of this mission independently or in combination with other missions.

Figure 22 is a repeat of Figure 13 and indicates how the notional mission can be inserted into the aircraft usage and a new mission mix determined. New Load occurrence tables are generated which are used in the development of stresses for 10 aircraft control points. The development of the stress spectrum is presented in Figure 23. Once the stresses have been calculated for each load condition, the load conditions are consolidated by stress. A flight-by-flight randomization is then performed, and the stress spectra is cycle-counted. The cycle-counted spectrum is then used in the development of crack growth curves, crack growth coefficients and inspection reports.

In conclusion, the A-10A service life monitoring program has the capability to develop aircraft maneuver spectra more accurately, faster and with the flexibility to include variation of aircraft loads parameters. The computer software includes the capability of inserting a notional mission and to evaluate that the aircraft performance can be attained and ensure that structural limits are not violated. The remaining life can be calculated either of the isolated notional mission or in conjunction with other existing missions. This program offers a tool which helps improve the USAF's ability to track individual aircraft life and the ability to optimize the A-10A's remaining life by analyzing the effect of notional missions on expected airframe life. A summary of this paper is presented in Figure 24.

Acknowledgement:

The authors would like to express their gratitude to James Kokoris of Grumman's Loads and Criteria group and Ronald E. Smith of Grumman's Computer-Aided Structural Analysis group for their work in assisting in the preparation of this paper.

- History
- Spectrum Development
- Tracking Program
- Spectrum Updating
  - IAT
  - L/ESS
  - Notional Mission
- Fracture Analysis/Reports
- Summary

FIGURE 1 AGENDA

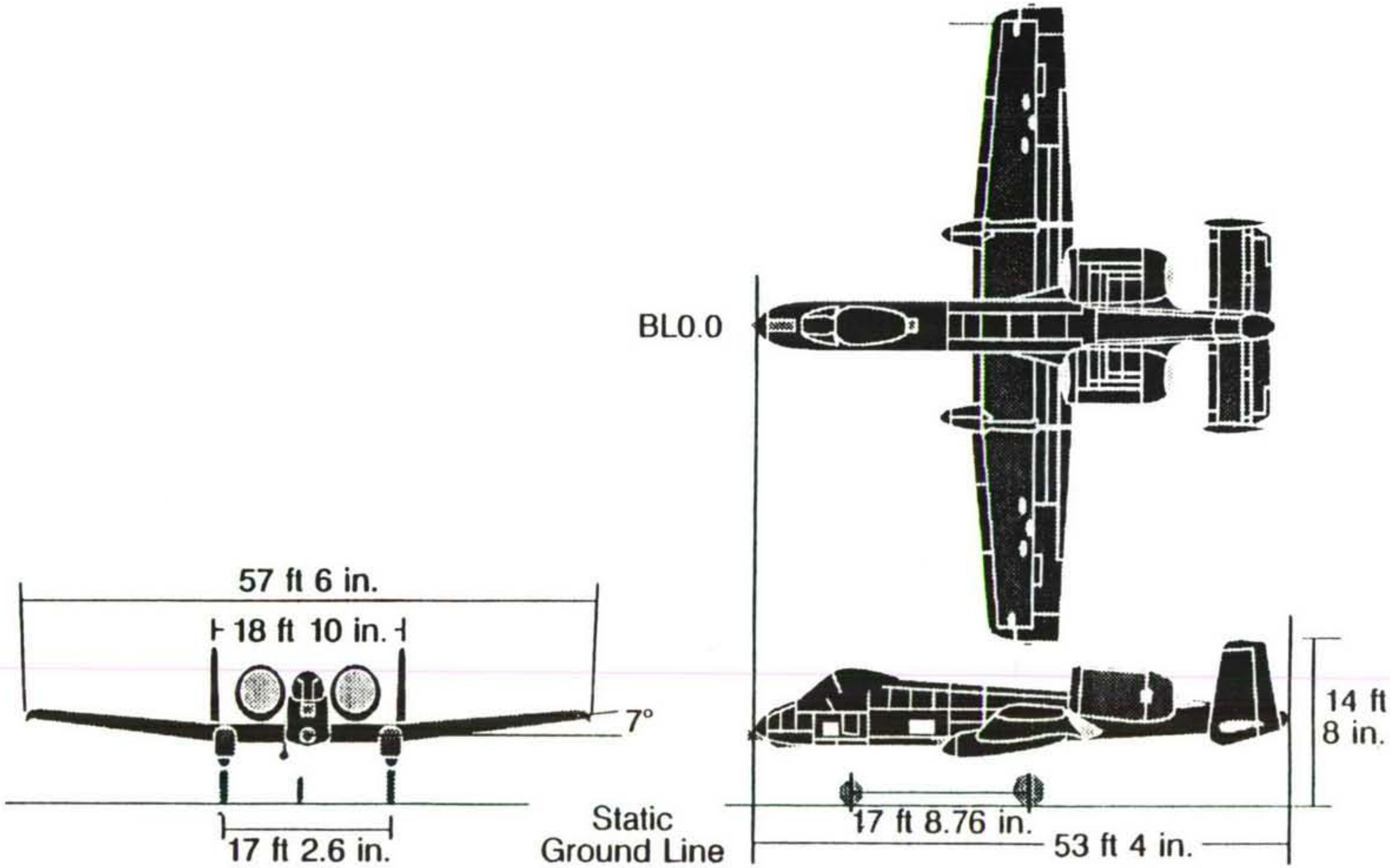


FIGURE 2 A-10A DIMENSIONS

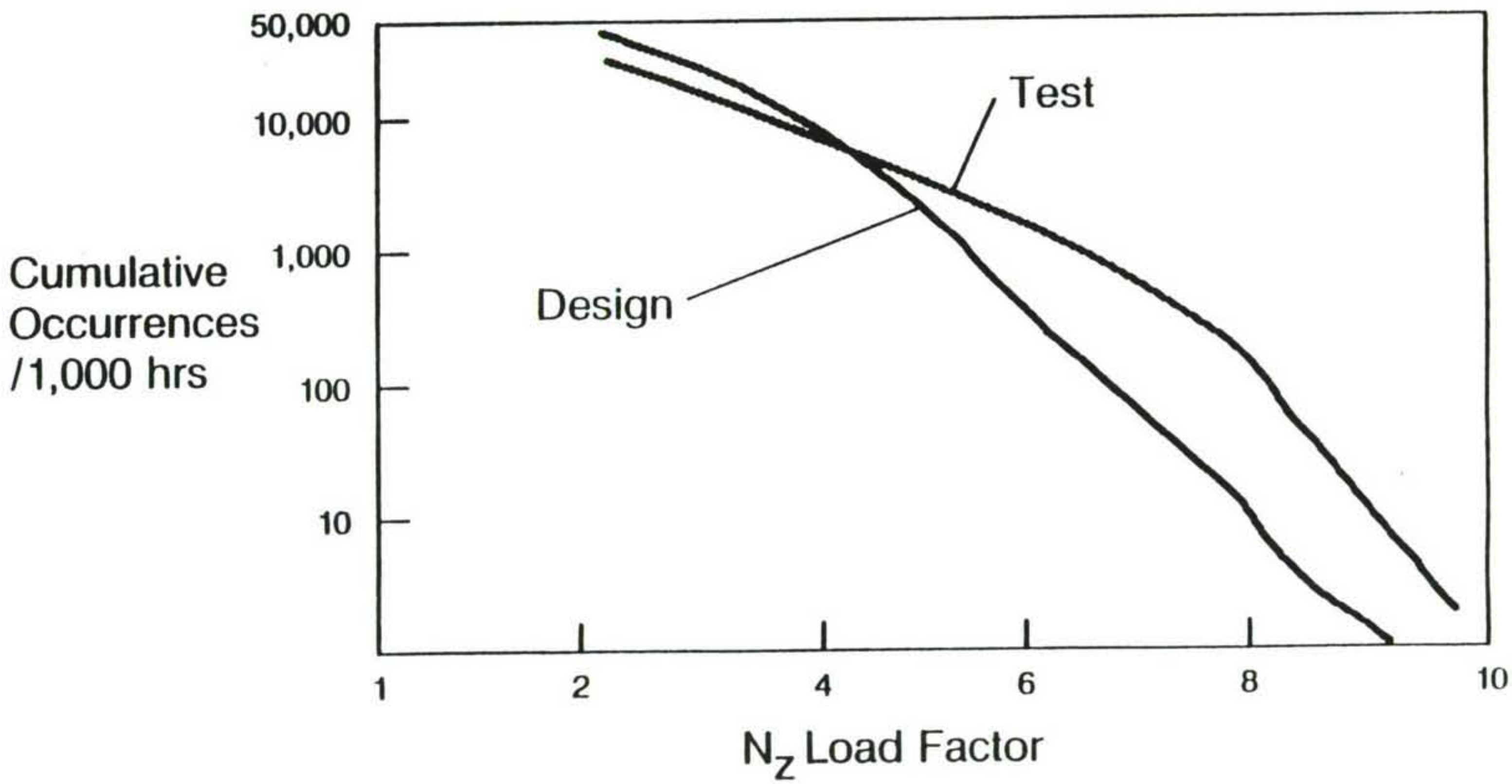


FIGURE 3 A-10A DESIGN AND TEST SPECTRA

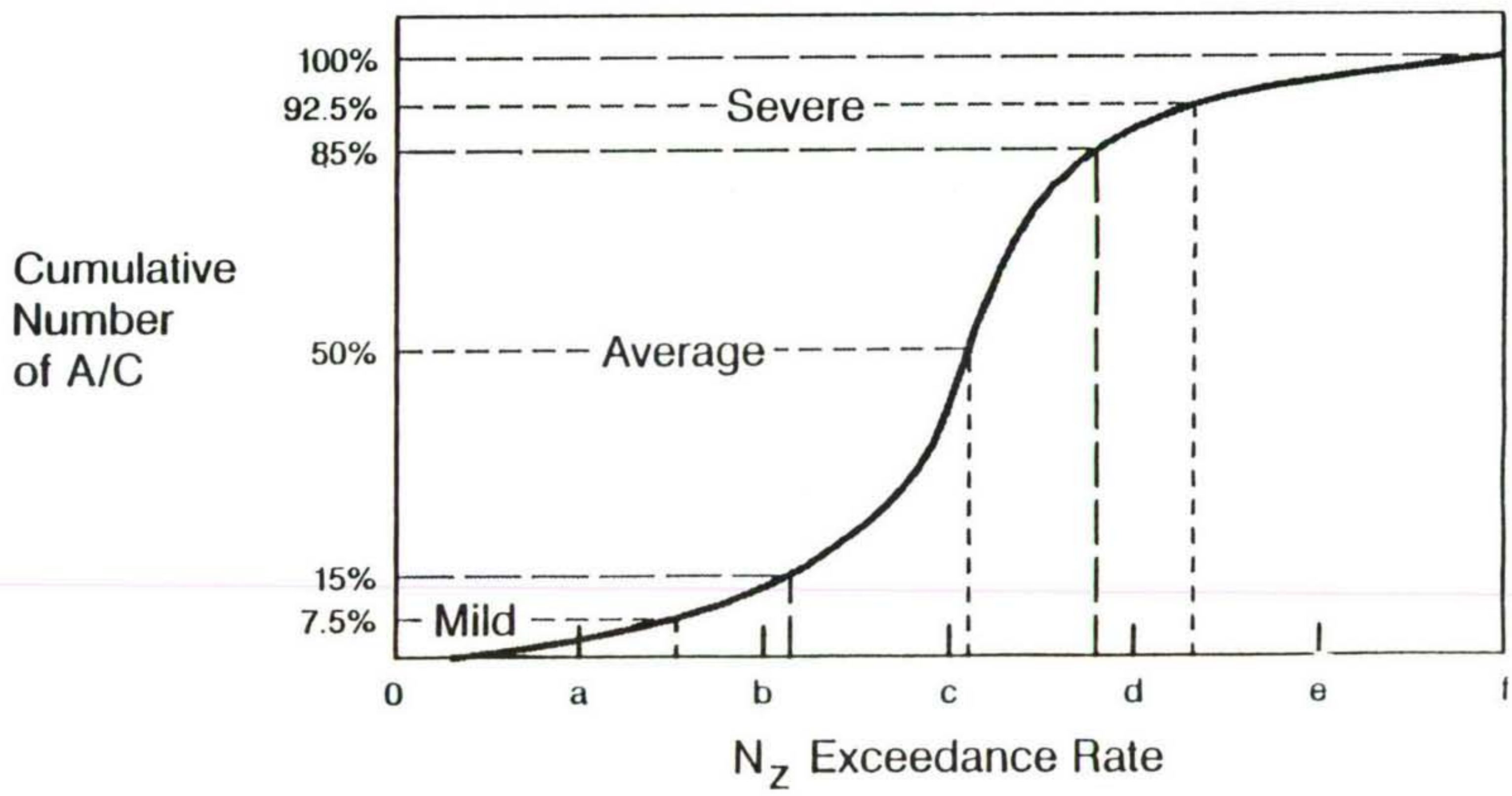


FIGURE 4 CUMULATIVE PROBABILITY CURVE FOR A GIVEN Nz

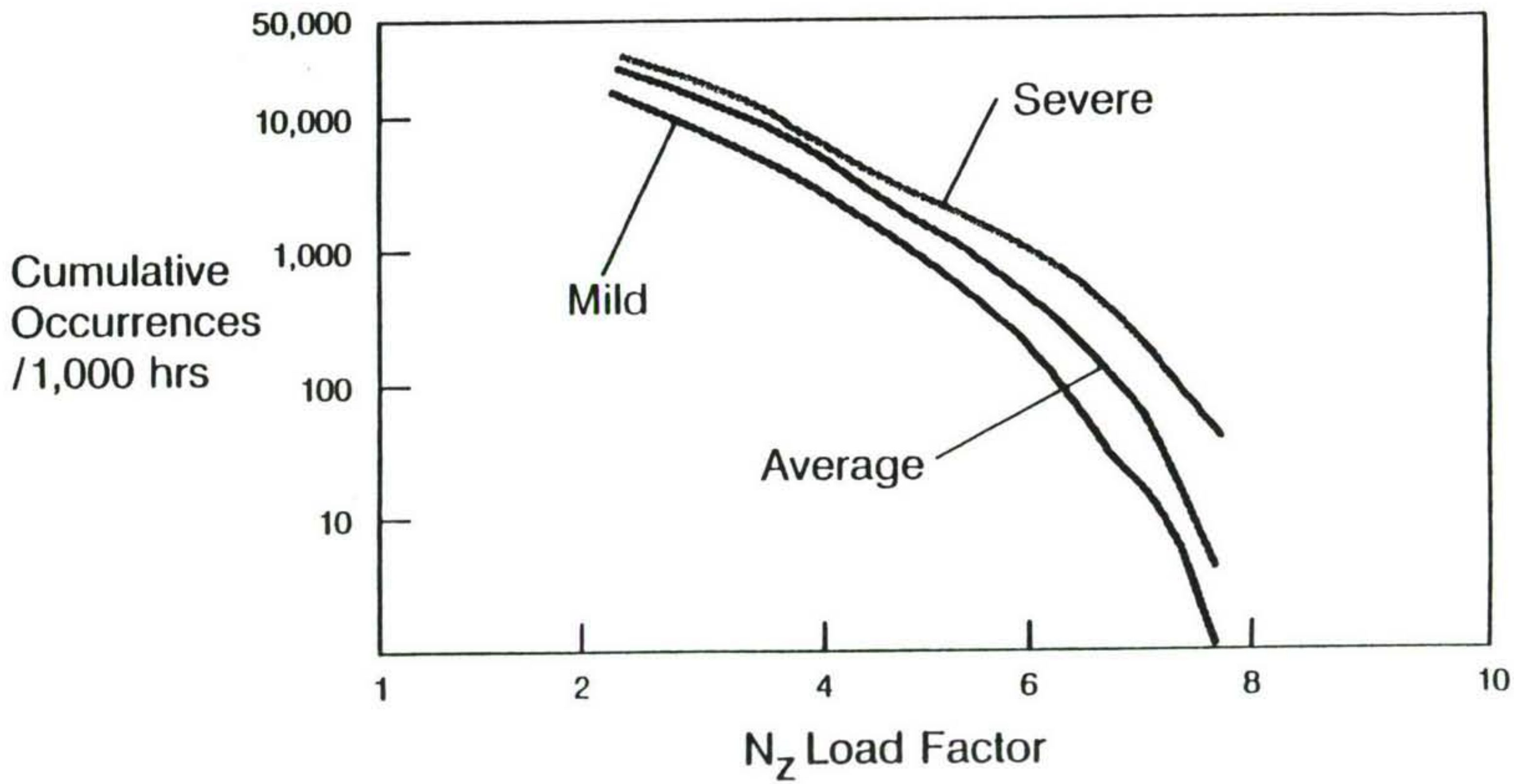


FIGURE 5 MILD, AVERAGE AND SEVERE TRACKING SPECTRA

Methodology

- Based on fracture analysis
- Damage coefficients in terms of crack extension per vertical load factor occurrence for ten control points
- Three usage spectra
  - Mild
  - Average
  - Severe
- Track by Components
- Automatic update by TCTO incorporation

FIGURE 6 INDIVIDUAL AIRCRAFT COMPONENT TRACKING PROGRAM - METHODOLOGY

The A-10 aircraft is tracked as if it were a group of components flying in close formation

FIGURE 7 INDIVIDUAL AIRCRAFT COMPONENT TRACKING PROGRAM - COMPONENTS

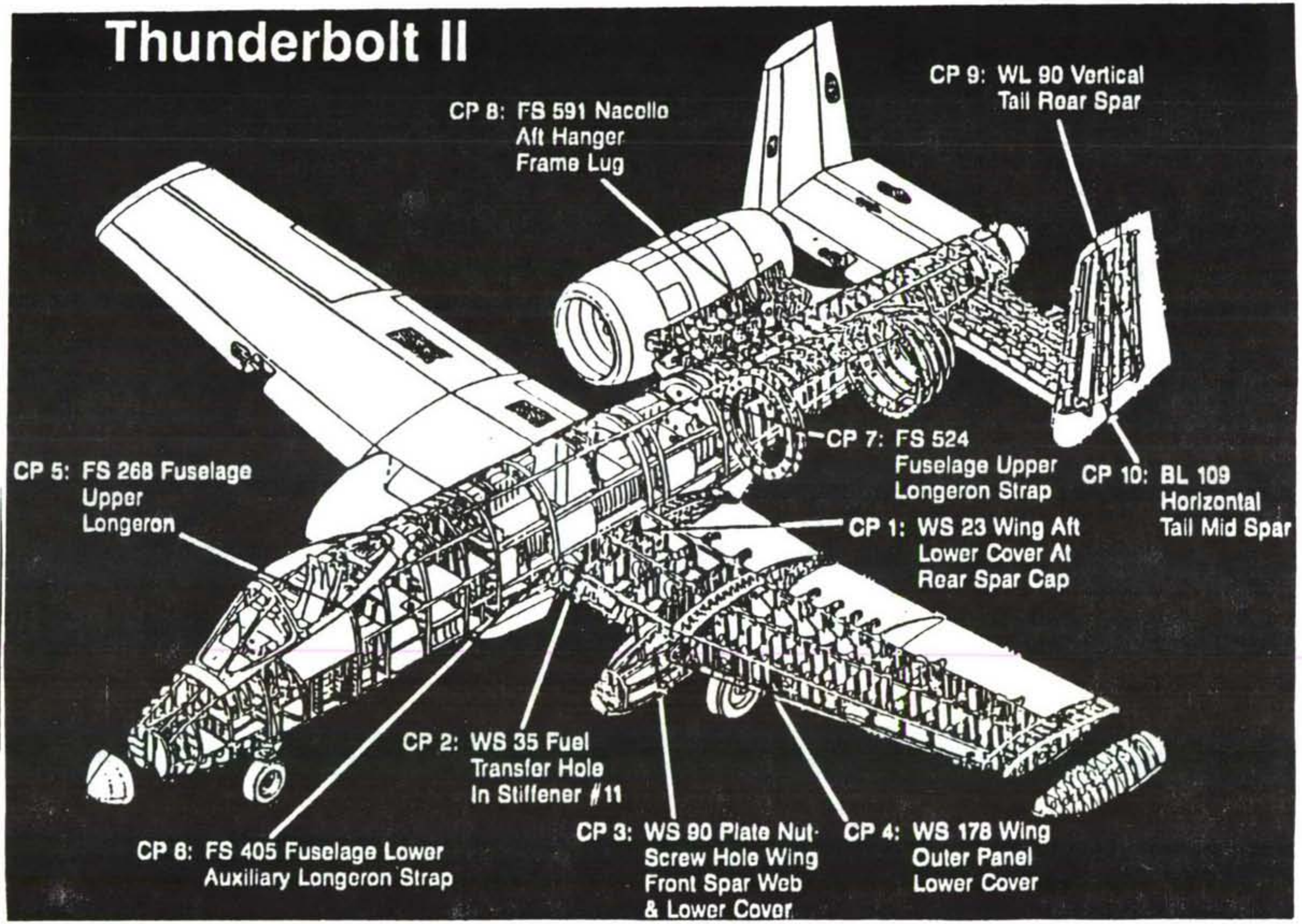


FIGURE 8   A-10A IATP CONTROL POINTS

Utilizing Fracture Analysis

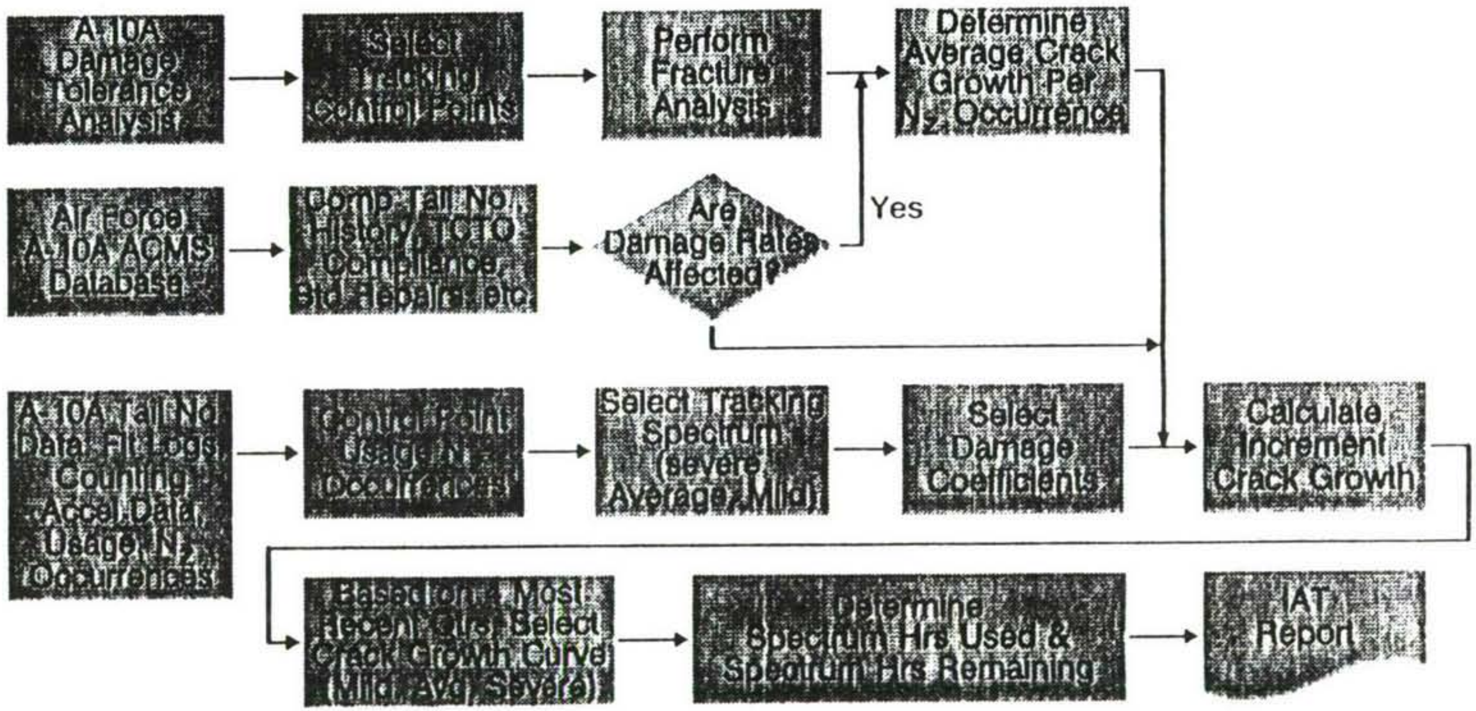


FIGURE 9 A-10A IATP FLOW CHART

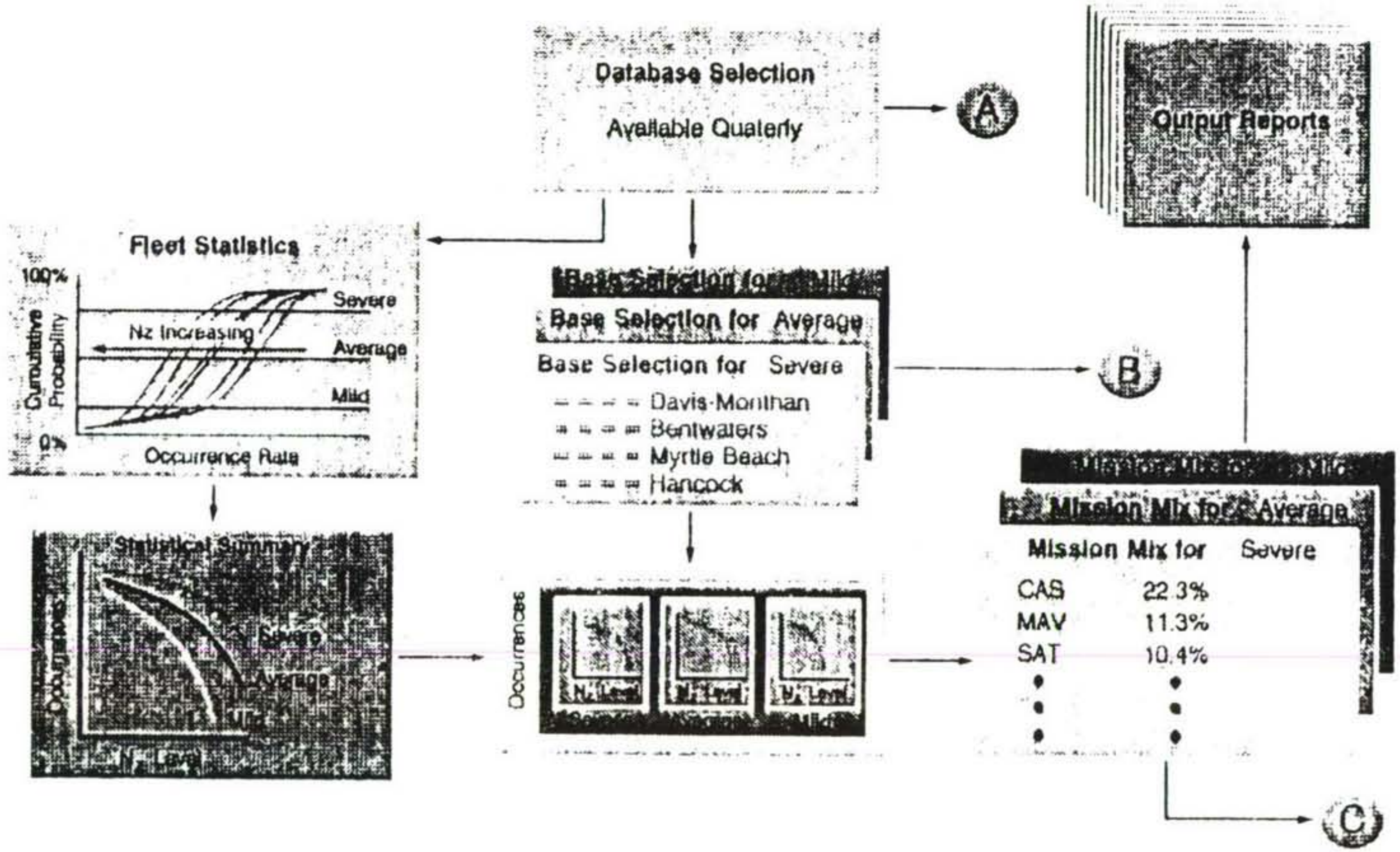


FIGURE 10 SPECTRA DEVELOPMENT - IAT USAGE FLOW CHART

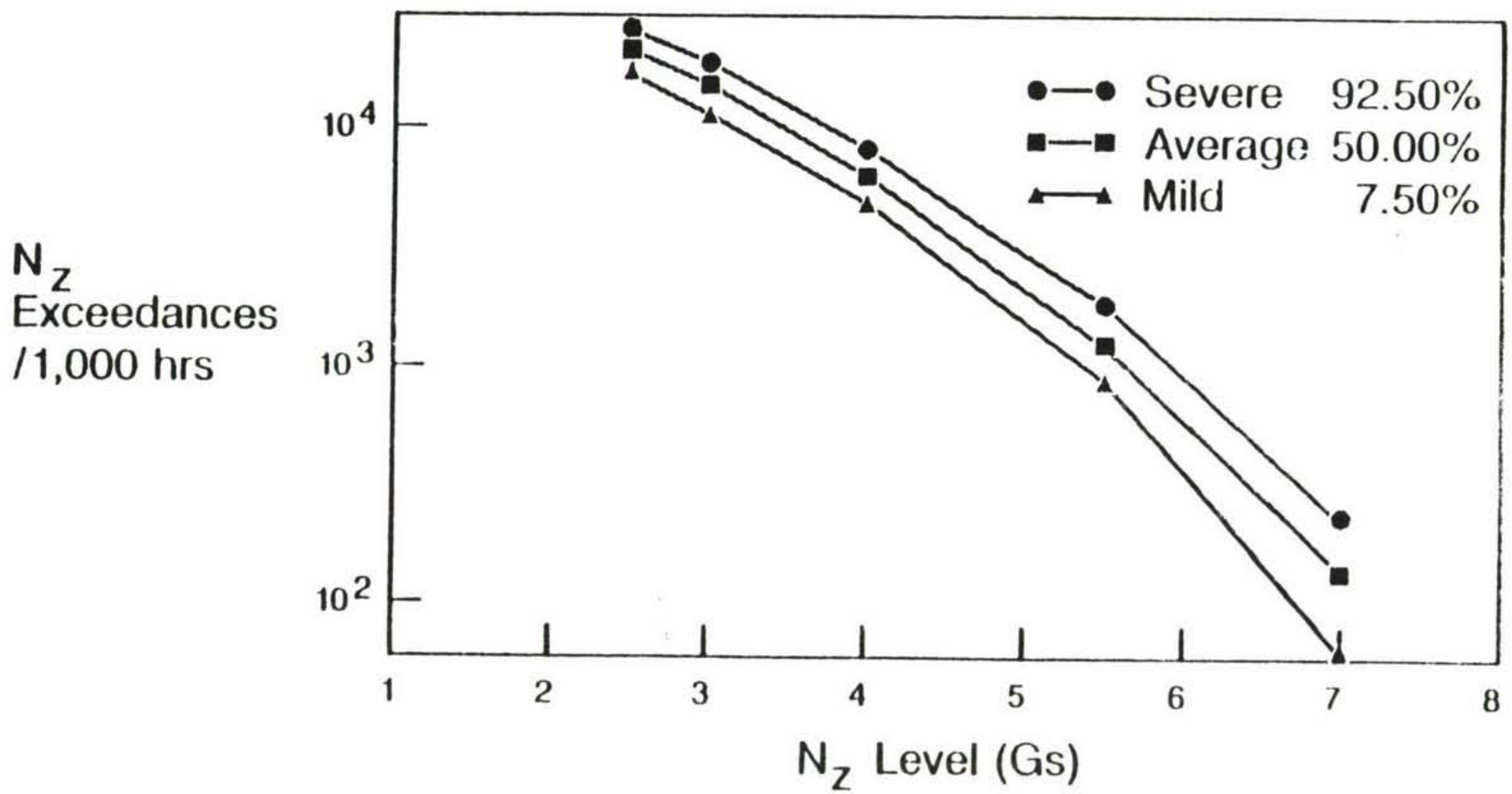


FIGURE 11 TYPICAL MILD, AVERAGE AND SEVERE STATISTICAL SPECTRA

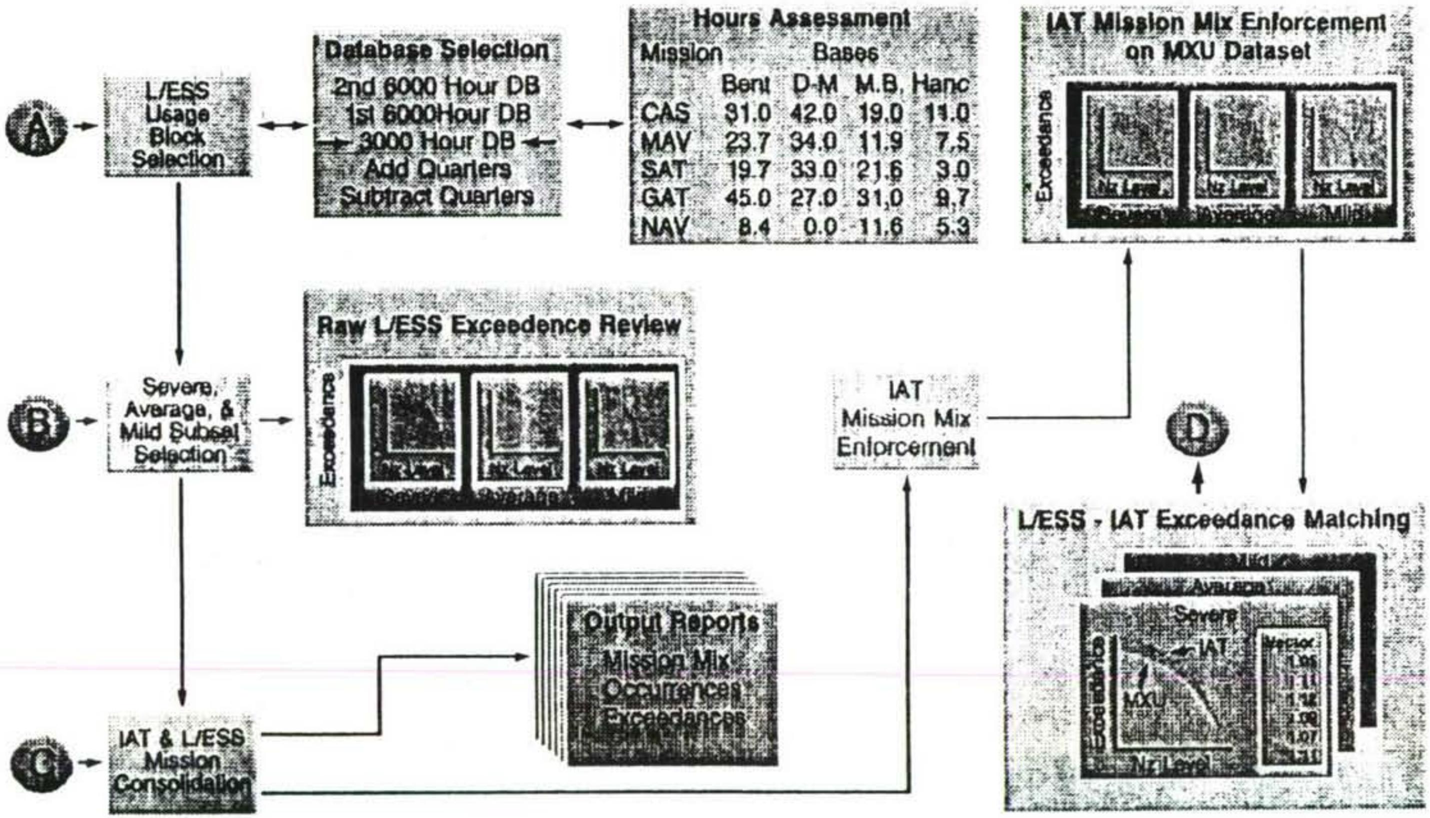


FIGURE 12 SPECTRA DEVELOPMENT - MXU L/ESS USAGE FLOW CHART

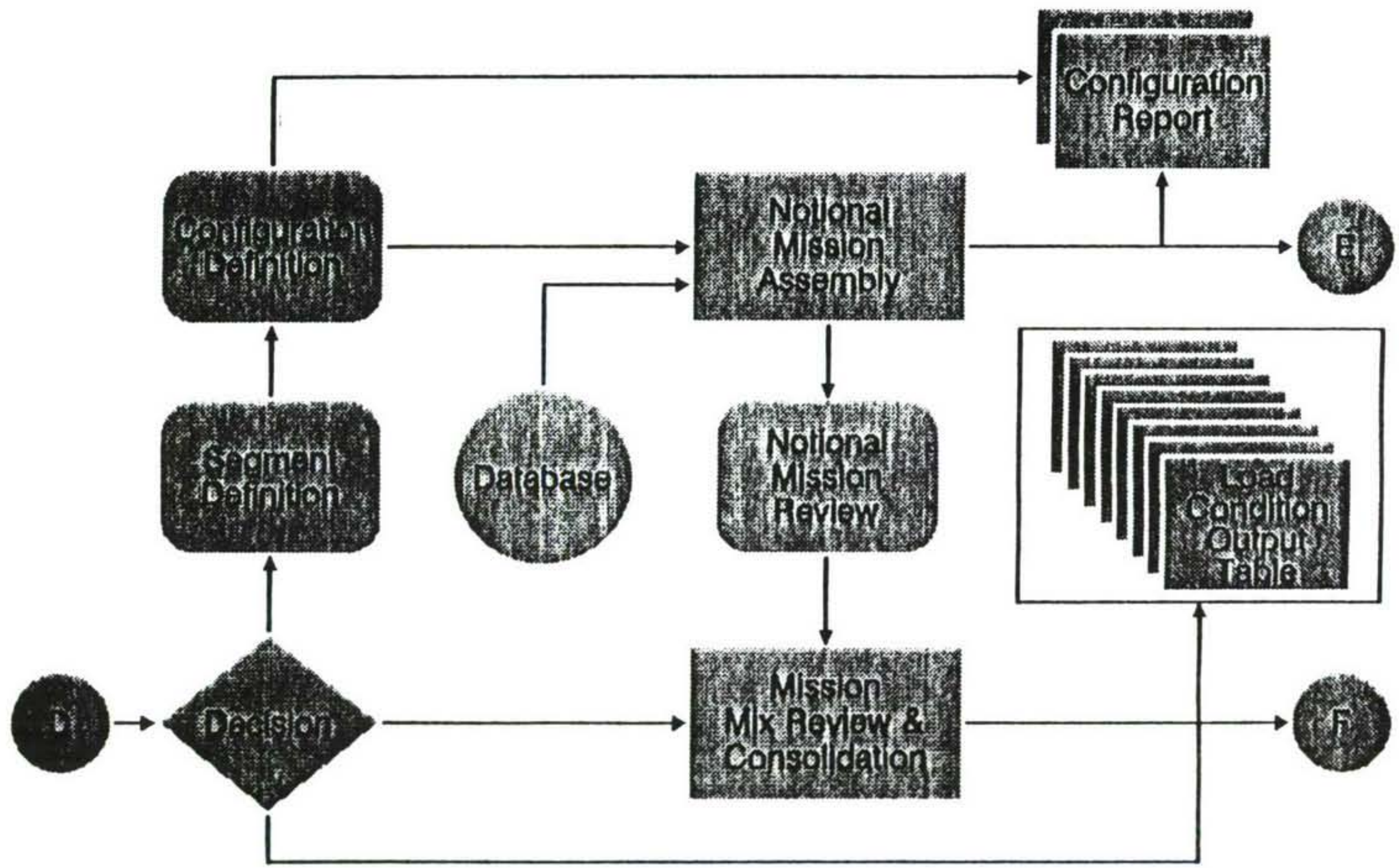


FIGURE 13 MISSION MIX DEVELOPMENT FLOW CHART

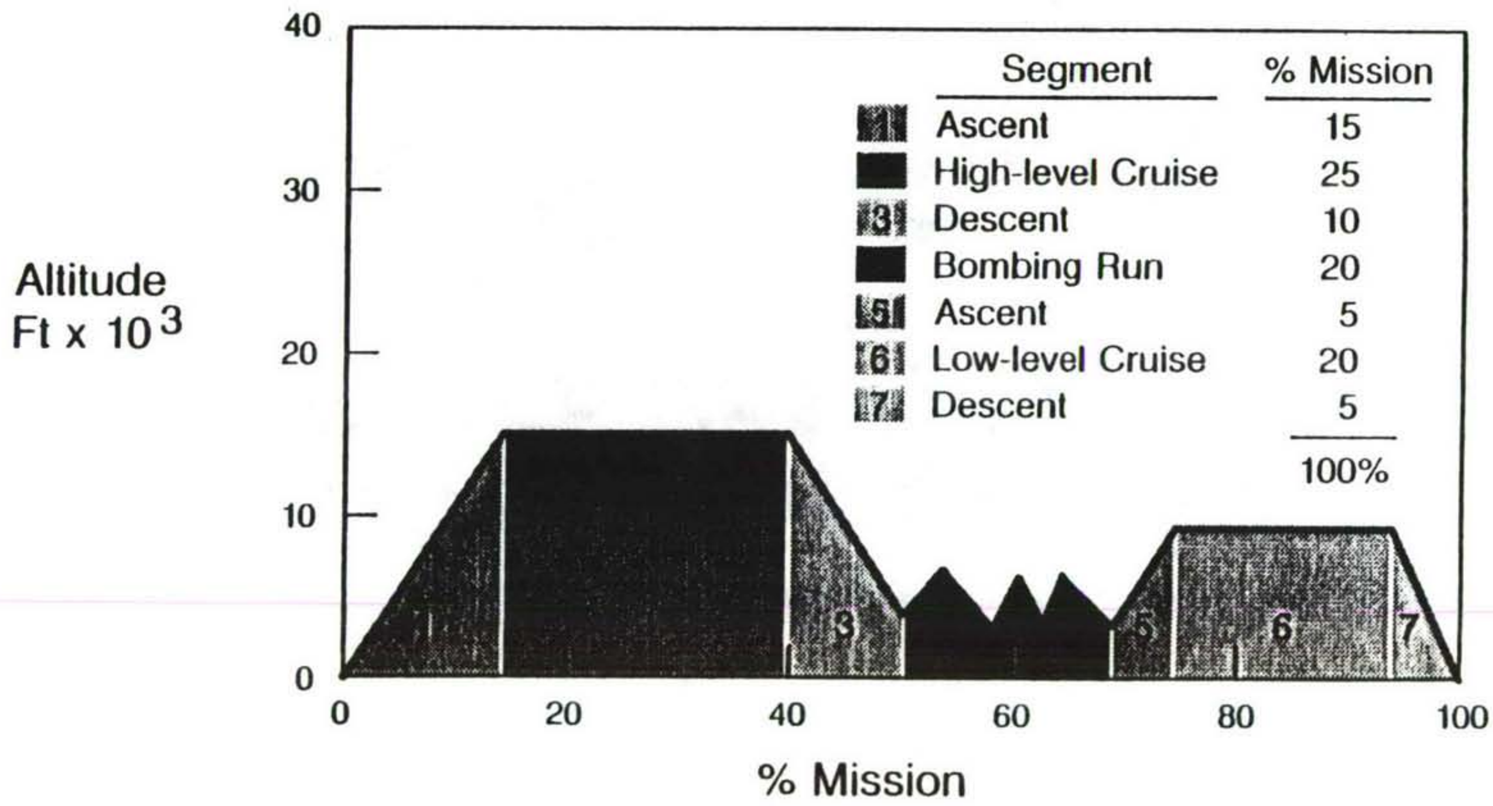


FIGURE 14   NOTIONAL MISSION PROFILE

Notional Mission - NOT: Example Mission Number 1

Segment	Configuration	Description	Duration
1	1 48000	1 Ascent	10
2	2 44000	5 High-Level Cruise/Navigation	30
3	2 44000	9 Rocket Delivery	12
4	3 40000	5 High-Level Cruise/Navigation	25
5	4 36000	2 Descent	10
			87

1 48000	Takeoff Weight
2 44000	Outward Leg
3 40000	Return Leg

Description Reference
1 Ascent
2 Descent
3 Low-Level Cruise/Navigation

FIGURE 15 NOTIONAL MISSION DEFINITION

Stores Manager Initial Configuration

Stores Weight
10698.00

Center Line
Pylon Installed

Incremental Drag
6.06

Station 5
(6) MK-82 LDGP MER

Station 7
(6) MK-82 LDGP MER

Station 4
MK-82 LDGP

Station 8
MK-82 LDGP

Station 3
MK-82 LDGP

Total 30MM
Rounds of
Ammunition
750

Station 9
MK-82 LDGP

Station 2
MK-82 LDGP

Station 10
MK-82 LDGP

Station 1
Pylon Installed

Station 11
Pylon Installed

FIGURE 16 NOTIONAL MISSION STORES CONFIGURATION DATA

Mission Segment Selection Menu

1.	Climb/Cruise	Climb
2.	Cruise	find both Mach & Altitude to yield max Naut. mi/lb
3.	Loiter	Highest Mach Number for Minimum Drag or Minimum Fuel Flow
4.	Dash	DASH LEG
5.	Accel/Decel	Acceleration
6.	Combat	Combat type 1
7.	Allowances or Stores Events	Take off Allowance

FIGURE 17 NOTIONAL MISSION SEGMENT PERFORMANCE MENU

Aircraft Mission Program

A-10A/TF34-GE-100 SAC Chart Hi-Lo-Lo-Hi Mission Std Day

(18) MK-82 on (2) MERS, (11) Pylons, 750 Rds Ammo, Full Fuel

Gross Weight = 47415

Maneuver	Altitude (feet) Init.	Altitude (feet) Final	Mach No. Init.	Mach No. Final	Final Weight (lbs)	Fuel Used (lbs)	Time Used (min)	Final Time (min)	Dist. Traveld (N. mi)	Final Dist. (N. mi)
Takeoff Allow.					46954	460	0.0	0.0	0.00	0.00
Climb	0	22500	0.300	0.413	45622	1331	17.9	17.9	67.86	67.86
Cruise	22500	22500	0.450	0.450	43167	2455	47.9	65.8	218.36	286.22
Dash	0	0	0.490	0.490	42046	1120	9.2	75.1	50.00	336.22
Drop Bombs					32506	0	0.0	75.1	0.00	336.22
Combat	0	0	0.511	0.511	31287	1219	10.0	85.1	0.00	336.22
Fire Ammo					30440	0	0.0	85.1	0.00	336.22
Dash	0	0	0.511	0.511	29358	1082	8.9	93.9	-50.00	286.22
Climb	0	35000	0.300	0.450	28431	927	16.1	110.1	-61.11	225.11
Cruise	35000	35000	0.486	0.486	26971	1460	48.2	158.3	-225.10	0.00
Loiter	0	0	0.213	0.213	26325	646	20.0	178.3	0.00	0.00

FIGURE 18 NOTIONAL MISSION PERFORMANCE OUTPUT

Performance & Notional Correspondence Table

Sequence No.	Performance Segment	Corresponding Notional Mission Segment
1	Miscellaneous	none
2	Climb/Descent	Ascent
3	Cruise	High-level Cruise
4	Dash	Low-level Penetration
5	Miscellaneous	none
6	Combat	Bombing
7	Miscellaneous	none
8	Dash	Low-level Penetration
9	Climb/Descent	Ascent
10	Cruise	High-level Cruise
11	Miscellaneous	none
12	not used	not used

FIGURE 19 NOTIONAL MISSION PERFORMANCE/STRUCTURAL CORRESPONDENCE MENU

1. Climb/Descent
    - Ascent
    - Descent
2. Cruise
    - Low-level Cruise
    - High-level Cruise
    - Air Refueling
    - Reconnaissance
    - Observation
    - Terrain Following
3. Loiter
    - Loiter
    - Observation
4. Dash
    - Low-level Penetration
    - Reconnaissance
5. Accel/Decel
    - none
6. Combat
    - Transition
    - Air to Air
    - Rocket Delivery
    - Search & Rescue
    - Electronic Warfare
    - Bombing
    - Strafing & Gunnery
    - Escort
    - Functional Check Flight
7. Allowances or Stores Events
    - none

FIGURE 20 NOTIONAL MISSION STRUCTURAL CORRESPONDENCE SEGMENTS

Title: Notional Mission Summary

Page 1 of 25

Number of flights = 159.96 Total flights hours = 240.00

Nz:	-3.00	-2.50	-2.00	-1.50	-1.00	-0.50	0.00	1.00	2.00
Occurs:	0.00	0.00	0.20	0.90	6.30	49.30	0.00	0.00	0.00
Nz:	2.50	3.00	3.50	4.00	4.50	5.00	5.50	6.00	6.50
Occurs:	2306.00	1753.40	1280.80	899.10	565.40	379.90	218.20	121.30	63.00
Nz:	7.00	7.50	8.00	8.50	9.00	9.50	10.00	10.50	11.00
Occurs:	34.40	14.60	5.20	1.90	0.30	0.50	0.00	0.00	0.00

Gross Wt.	Vel	Nz	Ny	Qdot	Occurs
28000.000	150.000	2.500	0.000	0.000	69.03
28000.000	200.000	2.500	0.000	0.000	111.27
28000.000	250.000	2.500	0.000	0.000	5.16
28000.000	200.000	3.000	0.000	0.000	74.25
28000.000	250.000	3.000	0.000	0.000	49.50
28000.000	200.000	3.500	0.000	0.000	34.77
28000.000	250.000	3.500	0.000	0.000	23.19
28000.000	200.000	4.000	0.000	0.000	13.29
28000.000	250.000	4.000	0.000	0.000	8.88
*	*	*	*	*	*
*	*	*	*	*	*
*	*	*	*	*	*
*	*	*	*	*	*
44000.000	*	*	*	*	*

FIGURE 21 NOTIONAL MISSION LOAD OCCURRENCE TABLE

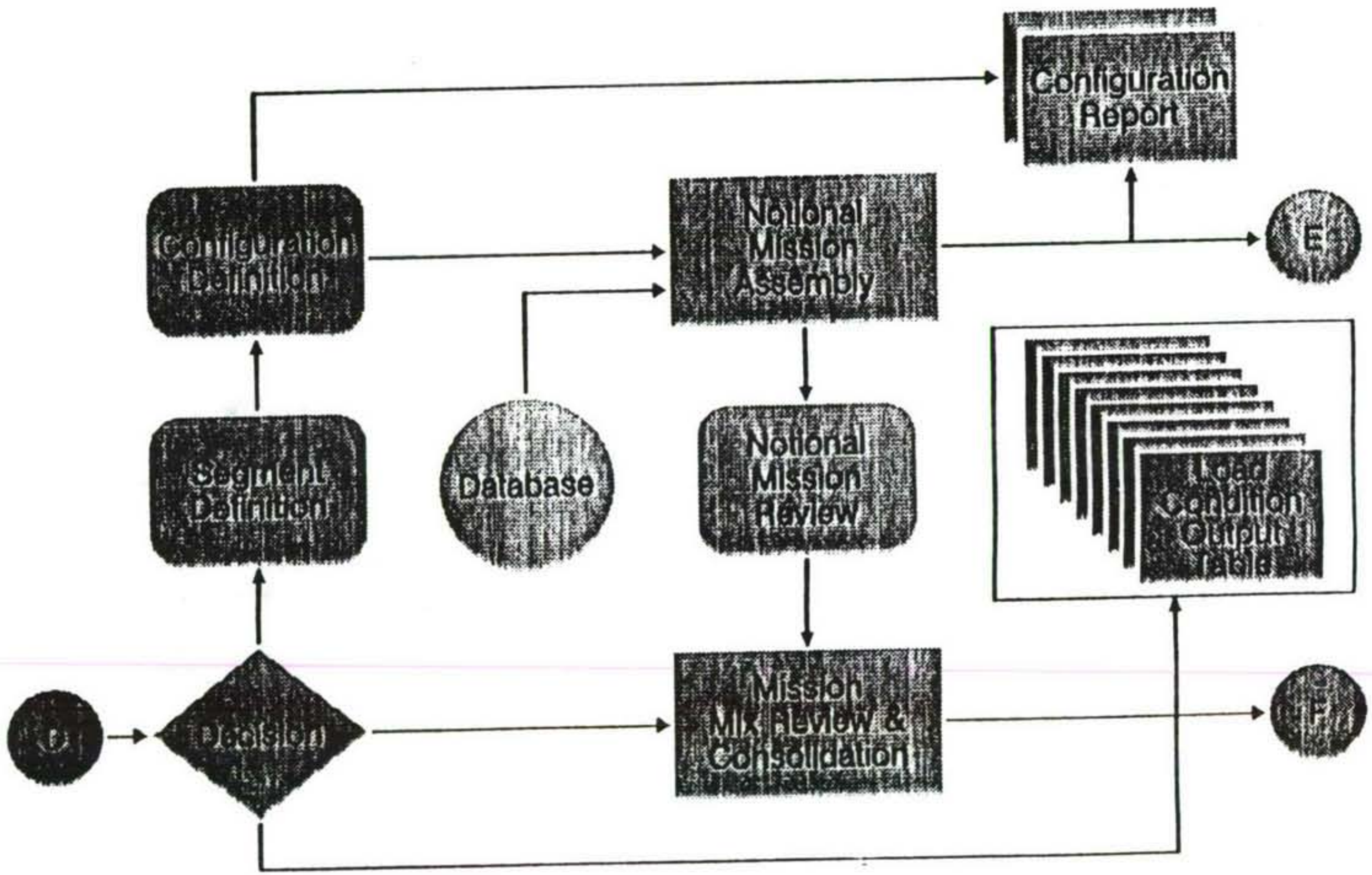


FIGURE 22 MISSION MIX DEVELOPMENT FLOW CHART

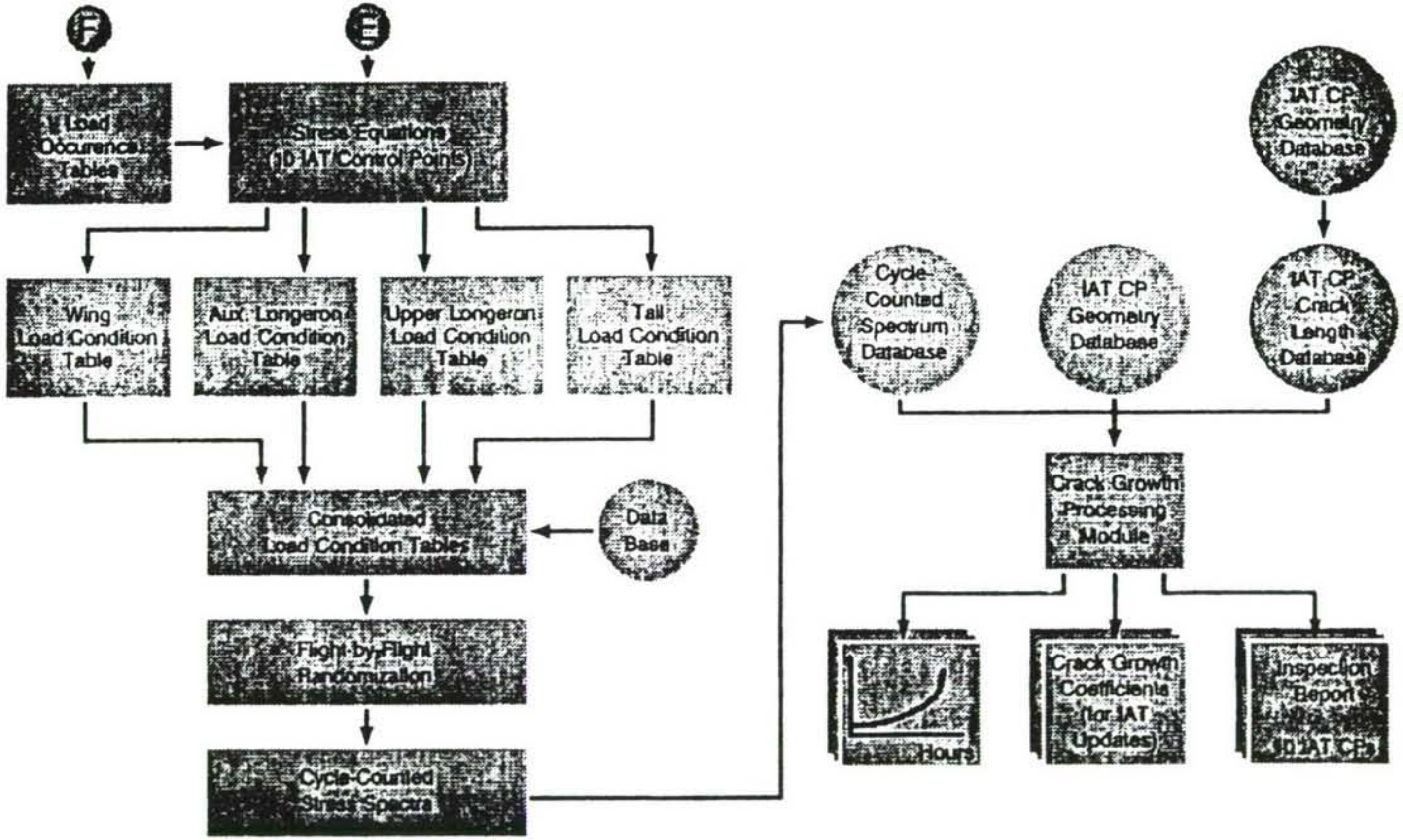


FIGURE 23 STRESS SPECTRUM FLOW CHART

- Development of Spectra
  - More accurate
  - Faster
  - Variation of parameters
- Development of a notional mission
  - Calculate performance
  - Ensure that structural limits are not violated
- Calculate remaining life
  - Variation of parameters
  - Variation of mission mix
  - Incorporation of notional mission

FIGURE 24 SUMMARY

Improvement of Structural Integrity Data Validity Utilizing Airborne Parameter Reasonableness Techniques

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Abstract

There are a number of factors which tend to invalidate data collected for Aircraft Structural Integrity Program (ASIP) analysis. Depending upon the aircraft, historical evidence indicates that in excess of 50% of the structures data collected with existing tape-based signal data recorders may be erroneous and unusable. Much of this is attributable to undetected failure of the recorder (or its associated multiplexer) and noise, bias, drift, or outright failure of the transducers sensing the structural parameters. While detection of signal data recorder failure is largely a solved problem with new equipments utilizing advanced Built-in Test (BIT) techniques, timely determination of transducer failure continues to be a significant issue. With most currently fielded systems, these failures are typically not identified until post-processing is accomplished at the Aircraft Structural Integrity Management Information System (ASIMIS) facility at Tinker AFB. By the time that it is determined that a probable transducer failure has occurred and corrective action has been taken, a period of several weeks to a month or more may have elapsed. In the meantime, additional quantities of contaminated data will have been collected from the defective transducer.

With the advent of microprocessor-based signal data recorders, the means now exists for the recorder to perform BIT not only on itself, but also on its input devices. A variety of algorithms are available which detect suspect structural data inputs in real time and identify potential transducer/wiring problems for immediate maintenance. A brief description of each of these algorithms is presented along with an assessment of their applicability and relative feasibility of implementation. Potential pitfalls such as invalid transducer failure indications are addressed and avoidance techniques are discussed. The paper concludes with several suggestions for further improvements in structural data quality.

Improvement of Structural Integrity Data Validity Utilizing Airborne Parameter Reasonableness Techniques

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Introduction

The military standard for the Aircraft Structural Integrity Program (ASIP), MIL-STD-1530A, states:

"The Air Force will be responsible for the overall planning and management of the loads/environment spectra survey, and will... ensure that the data are of acceptable quality and are obtained in a timely manner so that the contractor can analyze the results, develop the baseline spectrum, and update the analyses and force structural maintenance plan."

Historically, tape-based structural data recorders on USAF aircraft[1] have included sufficient circuitry to perform only a rudimentary self-calibration and health status check. No capability to assess inflight data validity has been implemented, and support for postflight assessment has been limited to simple framing and parity checks. As a result, malfunctions of dedicated structures transducers[2] with these systems are typically not detected at the organizational level. Generally, the first indication of a dedicated transducer failure occurs at the Aircraft Structural Integrity Management Information System (ASIMIS) facility at Tinker AFB. At ASIMIS, both automated and manual means are utilized to determine if the recorded data are valid for subsequent structural analysis. When invalid data are identified, attempts are made to correct the data and, if possible, the root cause of the failure is determined. As appropriate, the cognizant Air Logistics Center (ALC) for the source aircraft of the invalid data is advised that maintenance action is required. The delays inherent in this approach tend to be lengthy and, as might be expected, a significant quantity of the total recorded structural data is either completely unusable or requires extensive manual correction. It is estimated that, depending upon

[1] e.g., AN/ASH-28, MXU-553, A-24U-10, A-24U-6 etc.

[2] i.e., those not utilized by other flight-critical avionics systems

(WP1403)

the aircraft, *from 15 to 56% of all data recorded by existing tape based systems is invalid, uncorrectable, and must be discarded.*

With the advent of third generation structural data recorders containing embedded microprocessors, the opportunity now exists for airborne implementation of at least a portion of the data validity analysis previously performed on the ground. Elements of this analysis, hereafter referred to as "parameter reasonableness testing", have already been successfully demonstrated on at least one operational system[3]. As new recorder systems are introduced and existing systems are upgraded, it is important that both government and industry have a clear understanding of the capabilities, potential, and limitations of this new technology. Accordingly, the purpose of this paper is to present a broad overview of parameter reasonableness testing, with particular emphasis on the following areas:

- Importance of parameter reasonableness testing
- Considerations in the selection of the parameters to be tested
- Algorithm descriptions
- Algorithm applicability
- Fault processing and annunciation

Motivation for Parameter Reasonableness Testing

Transducer Maintenance Support As newer and more reliable solid-state recorders become operational, the incidence of invalid data attributable to all causes is expected to decrease to about one-half of the current level[4]. Much of the remaining invalid data is then related to transducer malfunction. If the structural data recorder itself performs sufficient parameter reasonableness testing onboard the aircraft to support organizational level identification of dedicated transducers which are defective, dramatic improvements in the

[3] e.g., Smiths Industries Standard Flight Data Recorder

[4] This projection is based on ASIMIS' experience that the cause of invalid data is evenly distributed between the failure of recorders and associated equipment, and transducer malfunctions. Based on actuals from the Maintenance & Operational Data Access System (MODAS), MTBFs for various tape-based recorders are as follows: A-24U-10 — 70 hours; A-24U-6 — 66 hours; MXU-553A — 55 hours; AN/ASH-28 — 150 hours. MTBFs for solid-state recorders are generally an order of magnitude greater: Smiths Industries CSFDR — 9491 hours (based on limited data base); Northrop TEMS — 1448 hours; Electro-dynamics FDR — 846 hours. Accordingly, it is apparent that the incidence of corrupted data due to recorder failure should be virtually negligible with solid-state recorders vis-a-vis tape-based systems.

timeliness of maintenance are achievable. This, in turn, leads to more accurate structural analysis and reduced manual intervention during postflight processing.

Efficient Utilization of Solid-State Memory A second motivation for implementing airborne parameter reasonableness testing is related to the processing performed on data by recorders which utilize solid-state memory for in-flight data storage. In order to minimize memory weight, volume, and power requirements, these recorders typically utilize various types of data compression schemes which provide a three to hundredfold reduction in the volume of data to be stored, depending upon data dynamics. Certain types of transducer failures, however, can result in erroneous, but highly dynamic, inputs being provided to the recorder. When this occurs, compression efficiencies suffer and the solid-state memory is rapidly filled with invalid or redundant data. To provide a degree of robustness in the system design, software must be implemented which identifies and responds appropriately to inputs which appear to be limit-cycling or displaying excessive rates of change. The resulting functionality can also be used to identify and provide built-in test (BIT) failure indications for those transducers that provide the suspect data.

Calculated Data Integrity A third reason for implementing onboard parameter reasonableness testing results from the greater capability of microprocessor-based structural data recorders. Rather than simply recording parameters directly for subsequent ground processing and analysis, many third generation structural recorders now have sufficient processing to perform complex calculations during flight. This capability supports more efficient memory utilization and the implementation of functionalities that do not depend upon postflight processing and analysis. Reportedly, at least one USAF aircraft type[5] has been fitted with an experimental system which monitors various parameters and calculates structural crack growth during flight. Since the recorded data comprises only the final calculation results rather than the raw inputs, such systems must be robust enough to maintain a minimum degree of accuracy in the presence of noisy or sporadically erroneous inputs[6]. This, in turn, implies the implementation of sufficient intelligence in the recorder system to assess the input data validity and respond appropriately when that data is suspect.

---

5 Fairchild A-10

6 In the event of a parameter error, the absence of input data records tends to make it difficult to "back out" the effects of the errors and correct the predictions for crack growth.

(WP1403)

Error Analysis

Prior to the selection of an algorithm suite for parameter reasonableness testing, it is helpful to develop an understanding of the behavior, criticality, and relative incidence of significant transducer errors. In this context, a transducer error is defined to include not only failures and degradations in accuracy of the transducer itself, but also of its associated wiring, excitation, and actuation mechanisms, as well as any externally induced errors due to such sources as electromagnetic interference (EMI) or mechanical vibration. In general, data reasonableness algorithms tend to be "tuned" to very specific error behaviors or modes. Each algorithm implemented in a structural data recorder requires both program memory and processor throughput, and unlike a ground processing environment, the resources available in an airborne system for data reasonableness testing tend to be in short supply. Accordingly, the strategy for algorithm selection must consider the failure modes which can occur, their likelihood, and the criticality of the various parameters with respect to subsequent structural analysis.

Transducer Error Modes It is important to make a distinction between the specific failure mechanism of a transducer and the manifestation of that failure at the input port to the structural data recorder. A data reasonableness test can not directly determine the functional status of a specific transducer. Rather, it infers a probable failure or error based on atypical characteristics of the input data from that transducer. The root cause of the failure, of course, determines the particular corrective maintenance action. For the purposes of this paper then, failed transducer behavior is categorized in the following manner:

- **Erratic** Recorder input data differs from the actual parameter value in a seemingly non-deterministic manner. Subsets of this category include:
  - **Spikes** Input data may exhibit periodic or aperiodic, short-duration, high-amplitude transient behavior that is inconsistent with the physical process being monitored. Spikes may be caused by pulse noise emanating from onboard radars or by switching transients which occur as various electrical subsystems on the aircraft are actuated and secured.
  - **Chatter** Input data may exhibit repeated changes of value or state at frequencies which are significantly greater than that expected of the physical process being monitored. This error mode is observed in discrete inputs from a misadjusted switch.
- **Calibration Error** Input data may differ from the actual parameter value in a deterministic and repeatable fashion due to degradations in transducer accuracy resulting from misadjustment, aging, or environ-

(WP1403)

mental stress. Typically, an out-of-calibration transducer exhibits errors in scale factor, bias, or both. Special cases of calibration error include:

- **Frozen** The transducer output assumes a static value although the physical process being monitored continues to exhibit dynamic behavior. This situation corresponds to a scale factor of "zero" and represents a total failure of the transducer.

- **Reversed** The leads of a transducer inadvertently become reversed so that the scale factor is erroneously complemented.

Frequency of Occurrence Relatively little hard data is available to support a rigorous characterization of the frequency of occurrence of various transducer error modes. In the absence of more accurate data, we have relied on anecdotal evidence from the staff at ASIMIS as well as our own experience in compiling the list depicted in Table 1. Blanks in the table represent an absence of data for the indicated combination of aircraft and parameter.

In no particular order, the parameters with the greatest incidence of errors were:

- Vertical Acceleration
- Roll Acceleration
- Control Surface Positions:
  - Aileron
  - Flaperon
  - Elevator
  - Horizontal Tail
  - Rudder
  - Flap
- Strain Gages

Reportedly, of all the recorder inputs, those provided by strain gages have consistently experienced the highest error rates. On some aircraft types, it is not uncommon for as much as 80% of the strain gage data to be invalid.

Transducer/Parameter Criticality In the processing of Individual Aircraft Tracking (IAT) or Loads/Environment Spectra Survey (L/ESS) data, recorded parameters may be categorized as either critical or non-critical. Clearly, all of the

Table 1
Error Mode Incidence for Various Parameters

PARAMETER	AIRCRAFT						
	TRANSPORT			FIGHTER		TRAINER	
	Type "A"	Type "B"	Type "C"	Type "A"	Type "B"	Type "A"	Type "B"
Altitude	U-f, b	U-f	U-f, b	U-b, f	U-b, f		U-b
Airspeed	U-f, b	U-f, b	U-f, b	U-b, f	U-b		U-e, f
Taxi Speed	U-f		U-f, b				
Long. Accel.			U-f	U-b, e	U-e, f		
Lateral Accel.				U-e	U-f	U	U-f
Vertical Accel.	C-f	C-f, e	C-f	O-b, f	U-f	O	O-e
Roll Rate					U-f	U	U-b
Pitch Rate						U	U-b
Yaw Rate						U	U-b
Roll Accel.				C	U-f		
AOA				U			
Power Lever Angle					U-b, f, e		
Aileron Position	O-b		O-b	C-b, f, r		C-b, f, e	C
Flaperon Position					C-b, e		
Elevator Position	O-b		O-b			C-b, f, e	
Hor. Tail Position					O-b, e		C
Rudder Position	O-b		O-b	C-b, f, r	C-e, b	C-b, f, e	C
Flap Position	O-b	O-f	O-b				
Fuel Quantity		U		U-f, e	U		
Strain	C-f, b	C-f, e	C-b, f				
Weapon Count				U-e			
Speedbrake Position				U			
Landing Gear Position				U			
Autopilot Switch		O-f					
Weight on Wheels		U					

Legend: Capital letters denote relative frequency of occurrence:
U - Uncommon, O - Occasional, C - Common

Lower case letters denote failure modes:
b - Biased, f - Frozen, e - Erratic, r - Reversed

parameters utilized for a structural calculation or table entry are important; however, some of them are considered critical on the basis that, if corrupted, the entire data set is unusable. Non-critical parameters are those whose values can be approximately inferred from other parameters in the data set, or alternatively, do not *greatly* influence the results of the analysis. The specific parameters considered critical tend to differ from aircraft to aircraft since there are differences in their respective structural analysis methodologies. For ASIP purposes, however, the following parameters represent a common critical core for most aircraft:

- Nz
- Yaw
- Velocity
- Altitude

Additional parameters may be considered critical on specific aircraft.

Reasonableness Testing Algorithms

Algorithm Description With assistance from ASIMIS personnel and engineers from several airframe manufacturers, Smiths Industries has identified nine different types of algorithms which are suitable for parameter reasonableness testing. These include:

- **Activity Test** This test is intended to detect parameters which have become frozen at some level. At take-off, a copy of the first parameter sample is recorded, as well as the difference between the first and second sample values. Each subsequent sample is compared to the first sample copy and their difference is calculated. If that difference exceeds the previously recorded difference, the old value is replaced. Following landing, and prior to power-down, the resulting largest difference value is compared with a pre-established threshold to determine if the parameter has exhibited a degree of activity consistent with a functional transducer. In the event that the flight duration does not exceed an established minimum length, the logic for the postflight portion of the test is not initiated. Some care must be taken to ensure that the activity test is not applied to a parameter whose value may not change during a flight[7].

- **Chatter Test** This test is intended to identify erratic parameters with an abnormal degree of activity. It is particularly applicable to "trigger"

[7]e.g., the capacity transducer of an unused fuel tank

parameters, i.e., those parameters whose exceedance or change initiates the recording of a complete data set[8]. This is referred to as a "snapshot". Each incidence of a sample which triggers a snapshot causes a chatter counter to be incremented. Each incidence of a sample which does *not* trigger a snapshot causes the chatter counter to be decremented. Decrementing may continue to zero; however, the chatter counter is not allowed to assume a negative value. After each increment or decrement, the chatter counter value is compared to a predetermined "chatter limit". If the chatter counter value exceeds the limit, then a chatter fault is indicated.

- **Known Condition Test** This test is applicable to any parameter which assumes a relatively narrow range of values when the aircraft is in a known, well-defined condition. One implementation of this test involves determining when take-off has occurred[9], pausing for a few seconds, and comparing the current sample values of selected parameters with established maximum and minimum limits.

- **Limit Test** The intent of the Limit Test is to detect occurrences of a parameter value exceeding the physically realizable range that can be achieved by that parameter. Each sample of a parameter is compared to a pre-established minimum and maximum to determine if an exceedance has occurred. An exceedance is indicative of a fault condition. Care must be exercised in the selection of the minimum and maximum values since the physically realizable range of the parameter may exceed the design range. As a result, the parameter reasonableness thresholds for maximum and minimum limits must be set sufficiently high (or low) to prevent a fault indication on a valid parameter excursion beyond its design range.

- **Rate of Change Test** This test is intended to detect erratic parameter behavior with particular emphasis on spikes resulting from EMI. Many parameters have physical limits on their rates of change, or alternatively, there are rate of change limits that are exceeded only under the most extraordinary of circumstances. The test involves calculating the difference between subsequent samples and comparing that difference with a pre-established maximum rate of change. In the event that the difference is greater, a fault is indicated.

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[8] For example, when Weight on Wheels (WOW) transitions from "true" to "false", many Standard Flight Data Recorder (SFDR) -equipped aircraft initiate the recording of a complete set of L/ESS data.

[9] Take-off can be inferred from air speed, throttle settings, and WOW discrete transition.

- **Correlation Test** Many types of correlation tests can be postulated which compare related parameters for consistency. For example, rudder pedal position and rudder deflection parameters can be compared for consistency in both sign and magnitude. A limitation of the test is that when a fault condition is indicated, it is frequently difficult to determine which parameter is in error without resorting to further comparisons. This test has been implemented on several SFDR applications by comparing the state of the weight on wheels (WOW) discrete with airspeed and altitude to determine the validity of the discrete. In the event that the WOW discrete is true *and* the airspeed exceeds a minimum threshold *and* the altitude exceeds a minimum threshold for a minimum duration, a fault condition is indicated.

- **Distribution Test** This test is a very useful method of detecting several types of errors, including frozen, erratic, or biased data. Presently, ASIMIS utilizes a distribution test in the postflight processing of L/ESS data with a high degree of effectiveness. There are two approaches to implementing a distribution test. The first involves calculating the sum and the sum of the squares of the samples during flight, and calculating the mean and standard deviation of the parameter at the end of the flight[10]. These are then compared to an expected range of values to determine if either the mean or standard deviation is out of range. An exceedance is indicative of a fault condition. The second approach involves developing a histogram by dividing the range of the parameter into sections and counting the number of samples falling into each section. An estimate of the mean and standard deviation can then be inferred from the histogram which is processed in the manner described above.

- **Voting Test** This test is feasible when dual or triple redundant transducers are available on an aircraft so that inputs from separate transducers can be compared. If the difference exceeds a pre-established threshold, a fault condition is indicated. With dual redundant sensors, a fault can be detected but not localized. Triply redundant sensors support both fault detection and localization. Our experience has been that most aircraft do not support redundant sensors.

- **Status Word Test** This test is unlike those previously described in that it is utilized only on data received over the aircraft's 1553 multiplexed data bus rather than analog transducer inputs. Most, but not all, digital parameters undergo some degree of parameter reasonableness test

[10] This approach generally requires hardware support of floating point calculations to execute in real time, which tends to increase recurring costs of the recorder electronics. Therefore, this is not the favored method of implementation.

prior to receipt by the structural data recorder. The status of these tests, and hence the parameter validity, is indicated in one or more status words which accompany the parameter value. This test simply checks the status word to determine if some other subsystem in the avionics suite has detected a fault condition. Typically this test is applied in conjunction with the parameter reasonableness tests described above. A variation of this test which is employed on "all-digital" aircraft involves checking to determine that a parameter which is expected to be received over the aircraft MUX bus has, in fact, been received. In the event that it has not, the recorder responds as though the parameter has been received with a status word indicating invalid data.

Invalid Data Disposition When any one of the nine tests just described determines that potentially erroneous data is received, several options are available. These include:

- Record the data as received, but flag the suspect portions for later evaluation.

- Record the last valid sample of data in lieu of the suspect sample.

In our experience, different users have expressed different desires with regard to the management of suspect data. One argument in favor of recording the suspect data without change is that it allows a subsequent postflight analysis of the data to determine if, in fact, it really is invalid.

Test Applicability As noted previously, certain reasonableness tests tend to be better suited for the detection of particular error types than others. A summary of which types of errors are most likely to be detected by each of the recommended tests is presented in Table 2. The applicability of the Status Word Test is not addressed since that would depend upon tests performed elsewhere in the avionics suite and not in the signal data recorder.

Implementation Status A number of the parameter reasonableness tests previously described are either presently operational or, alternatively, are in various stages of implementation. Table 3 summarizes the status of these efforts for the Smiths Industries Standard Flight Data Recorder and Crash Survivable Flight Data Recorder. Presumably, variants of these tests have been implemented on third generation recorders developed by other manufacturers.

Fault Processing

In order for the parameter reasonableness tests to be useful, their results must be processed in some fashion, recorded, and made available to appropriate personnel for maintenance action as necessary. Ideally, the processing will remove as many false indications of transducer failure as possible without significantly reducing the capability of the recorder to accurately identify genu-

Table 2
Parameter Reasonableness Test Applicability

Error Type	Erratic	Spike	Chatter	Calibration	Frozen	Reversed
Activity				X	X	
Chatter		X	X			
Known Condition				X	X	X
Limit	X	X		X		X
Rate of Change	X	X	X			
Correlation				X	X	X
Distribution			X	X	X	X
Voting	X	X	X	X	X	X

Table 3
Parameter Reasonableness Test Implementation

Test	F-16	C-130	C-141	C-17
Activity Test	X	X	X	X
Chatter Test		X	X	
Known Condition Test	X			
Limit Test	X	X	X	X
Rate of Change Test		X	X	
Correlation Test				
Distribution Test				
Voting Test				
Status Word Test	X			X

inely erroneous parameter inputs. One approach to accomplishing this which has been implemented on SFDR is to require that a parameter repeatedly fail a reasonableness test before its BIT status is recorded as failed. If, at any time during a flight, the parameter passes its reasonableness test, the parameter BIT status is reset to a pass state. This is generally referred to as "filtered" BIT. Additionally, SFDR records a BIT failure history which contains a count of the total number of times that filtered BIT status of a parameter has toggled between pass and failed states. This latter feature allows maintenance personnel to identify intermittent or transient parameter errors.

This next issue which must be addressed is how to best annunciate a BIT failure to operational and maintenance personnel. One school of thought suggests that since the structural data recorder and its dedicated sensors are not flight-critical, it is unnecessary to alert the flight crew of a BIT failure. Rather, the BIT status and BIT Failure History are made available to maintenance personnel when data downloads from the recorder's memory are effected. For SFDR, download is accomplished using a ruggedized portable computer with a special serial interface for communications with the recorder. The download software which executes on this computer gives maintenance personnel a variety of options in reading out, initiating and and recording BIT status and BIT Failure History.

One exception to a policy of not annunciating transducer BIT failures to the flight crew should occur when the recorder is used for collecting mishap data in addition to structures data. In this case, the failure of a dedicated transducer providing inputs for mishap recording should be annunciated to both the flight crew and maintenance personnel so that repairs can be effected as soon as possible. This provides the highest probability of recovering valid mishap data from the recorder in the event that an incident should occur.

Conclusions and Recommendations

Recently, the Air Force grounded all of its A-7Ds and A-7Ks for inspection after an Air Guard maintenance team found small cracks around the lower trailing edge of an A-7D wing. Engineers from LTV, the A-7 manufacturer, attributed the cracks to "a pattern of usage more severe than was originally measured in the A-7D Aircraft Structural Integrity Program (ASIP)... ". In normal Navy and AF Tactical Air Command bombing missions "they took off, flew a half hour at altitude, made a standard combat bomb run, and then returned, probably at a high altitude, to a landing," explained Denis C. Duffy Jr., LTV's advanced A-7 program development manager. "In today's environment, the whole mission is flown at low altitude. In the mid-70s timeframe, we didn't have full low-level ingress and egress."[11]

This occurrence underscores the fact that from time to time the operational utilization of an aircraft evolves, and the existing baseline operational spectrum is no longer valid. The *raison d'etre* for L/ESS monitoring is to support early detection of these mission changes so that a new baseline spectrum can be developed and accurate structural analysis and predictions can continue.

[11] Aerospace Daily, 10/25/89

To the extent that a significant fraction of the total recorded L/ESS data is invalid, erroneous, or suspect, this objective cannot be fully realized.

Clearly, airborne parameter reasonableness testing is an important step towards improving the quality of recorded structures data through early detection of malfunctioning transducers. To achieve the full potential of this new technology, it is recommended that each new structures recorder implement parameter reasonableness testing to the greatest extent practicable. During the development of the recorder software, a close working relationship is required between the recorder contractor, the airframe manufacturer, and ASIMIS in order to define the most appropriate tests and parameter limits. During flight testing, the effectiveness of the reasonableness tests should be assessed with test limits and thresholds altered as necessary to provide the most effective detection of invalid data. Finally, after the recorder enters service, it is suggested that the effectiveness of the parameter reasonableness testing be assessed for a time, and, as necessary, changes in the algorithms, thresholds, and limits be implemented.

In closing, it is emphasized that, by itself, parameter reasonableness testing does not improve the quality of structural data. That data quality is largely a function of the inherent reliability and effectiveness of maintenance performed on the transducers which instrument the airframe. Parameter reasonableness testing can, however, provide much more timely notification of suspect transducers to organizational level maintenance personnel than is currently possible. In the near term, it is doubtful that an airborne capability will be developed which *completely* obviates the need for any postflight data quality analysis. It is reasonable, however, to expect that with relatively modest effort and minimal impact on system costs, parameter reasonableness testing can ultimately address the majority of commonly occurring transducer failures.

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